Module 12

FOR B1.3/B1.4 CERTIFICATION

HELICOPTER AERODYNAMICS, STRUCTURES AND SYSTEMS

Aviation Maintenance Technician Certification Series

- Rotary Wing Aerodynamics
- Flight Control Systems
- Blade Tracking and Vibration Analysis
- Transmission
- Airframe Structures
- Air Conditioning
- Instruments/Avionic Systems
- Electrical Power
- Equipment and Furnishings
- Fire Protection
- Fuel Systems
- Hydraulic Power
- Ice and Rain Protection
- Landing Gear
- Lights
- Pneumatic/Vacuum
- Integrated Modular Avionics
- On Board Maintenance Systems
- Information Systems

COMPLIANT WITH EASA PART 66/147

Version 001

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Aviation Maintenance Technician Certification Series

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EASA

EASA PART 66/147

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WELCOME

The publishers of this Aviation Maintenance Technician Certification Series welcome you to the world of aviation maintenance. As you move towards EASA certification, you are required to gain suitable knowledge and experience in your chosen area. Qualification on basic subjects for each aircraft maintenance license category or subcategory is accomplished in accordance with the following matrix. Where applicable, subjects are indicated by an "X" in the column below the license heading.

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We wish you good luck and success in your studies and in your aviation career!

REVISION LOG

VERSION	EFFECTIVE DATE	DESCRIPTION OF CHANGE
001	2022 03	Module Creation and Release

MODULE EDITIONS AND UPDATES

ATB EASA Modules are in a constant state of review for quality, regulatory updates, and new technologies. This book's edition is given in the revision log above. Update notices will be available Online at www.actechbooks.com/revisions.html

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FORWARD

PART-66 and the Acceptable Means of Compliance (AMC) and Guidance Material (GM) of the European Aviation Safety Agency (EASA), Appendix 1 establishes the Basic Knowledge Requirements for those seeking an aircraft maintenance license. The information in this Module of the Aviation Maintenance Technical Certification Series published by Aircraft Technical Book Company meets or exceeds the breadth and depth of knowledge subject matter referenced in Appendix 1 of the Implementing Rules. However, the order of the material presented is at the discretion of the editor in an effort to convey the required knowledge in the most sequential and comprehensible manner. Knowledge levels required for Category A1, B1, B2, and B3 aircraft maintenance licenses remain unchanged from those listed in Appendix 1 Basic Knowledge Requirements. Tables from Appendix 1 Basic Knowledge Requirements are reproduced at the beginning of each module in the series and again at the beginning of each Sub-Module.

How numbers are written in this book:
This book uses the International Civil Aviation Organization (ICAO) standard of writing numbers. This method displays large numbers by adding a space between each group of 3 digits. This is opposed to the American method which uses commas and the European method which uses periods. For example, the number one million is expressed as so:

ICAO Standard	1 000 000
European Standard	1.000.000
American Standard	1,000,000

SI Units:
The International System of Units (SI) developed and maintained by the General Conference of Weights and Measures (CGPM) shall be used as the standard system of units of measurement for all aspects of international civil aviation air and ground operations.

Prefixes:
The prefixes and symbols listed in the table below shall be used to form names and symbols of the decimal multiples and submultiples of International System of Units (SI) units.

MULTIPLICATION FACTOR	PREFIX	SYMBOL
1 000 000 000 000 000 000 = 10^{18}	exa	E
1 000 000 000 000 000 = 10^{15}	peta	P
1 000 000 000 000 = 10^{12}	tera	T
1 000 000 000 = 10^{9}	giga	G
1 000 000 = 10^{6}	mega	M
1 000 = 10^{3}	kilo	k
100 = 10^{2}	hecto	h
10 = 10^{1}	deca	da
0,1 = 10^{-1}	deci	d
0.01 = 10^{-2}	centi	c
0,001 = 10^{-3}	milli	m
0,000 001 = 10^{-6}	micro	µ
0,000 000 001 = 10^{-9}	nano	n
0,000 000 000 001 = 10^{-12}	pico	p
0,000 000 000 000 001 = 10^{-15}	femto	f
0,000 000 000 000 000 001 = 10^{-18}	atto	a

International System of Units (SI) Prefixes

EASA LICENSE CATEGORY CHART

	Module Number and Title	A1 Airplane Turbine	B1.1 Airplane Turbine	B1.2 Airplane Piston	B1.3 Helicopter Turbine	B1.4 Helicopter Piston	B2 Avionics
1	Mathematics	X	X	X	X	X	X
2	Physics	X	X	X	X	X	X
3	Electrical Fundamentals	X	X	X	X	X	X
4	Electronic Fundamentals		X	X	X	X	X
5	Digital Techniques / Electronic Instrument Systems	X	X	X	X	X	X
6	Materials and Hardware	X	X	X	X	X	X
7A	Maintenance Practices	X	X	X	X	X	X
8	Basic Aerodynamics	X	X	X	X	X	X
9A	Human Factors	X	X	X	X	X	X
10	Aviation Legislation	X	X	X	X	X	X
11A	Turbine Aeroplane Aerodynamics, Structures and Systems	X	X				
11B	Piston Aeroplane Aerodynamics, Structures and Systems			X			
12	Helicopter Aerodynamics, Structures and Systems				X	X	
13	Aircraft Aerodynamics, Structures and Systems						X
14	Propulsion						X
15	Gas Turbine Engine	X	X		X		
16	Piston Engine			X		X	
17A	Propeller	X	X	X			

GENERAL KNOWLEDGE REQUIREMENTS

MODULE 12 SYLLABUS AS OUTLINED IN PART-66, APPENDIX 1

Level 1
A familiarization with the principal elements of the subject.
Objectives:
a. The applicant should be familiar with the basic elements of the subject.
b. The applicant should be able to give a simple description of the whole subject, using common words and examples.
c. The applicant should be able to use typical terms.

Level 2
A general knowledge of the theoretical and practical aspects of the subject and an ability to apply that knowledge.
Objectives:
a. The applicant should be able to understand the theoretical fundamentals of the subject.
b. The applicant should be able to give a general description of the subject using, as appropriate, typical examples.
c. The applicant should be able to use mathematical formula in conjunction with physical laws describing the subject.
d. The applicant should be able to read and understand sketches, drawings and schematics describing the subject.
e. The applicant should be able to apply his knowledge in a practical manner using detailed procedures.

Level 3
A detailed knowledge of the theoretical and practical aspects of the subject and a capacity to combine and apply the separate elements of knowledge in a logical and comprehensive manner.
Objectives:
a. The applicant should know the theory of the subject and interrelationships with other subjects.
b. The applicant should be able to give a detailed description of the subject using theoretical fundamentals and specific examples.
c. The applicant should understand and be able to use mathematical formula related to the subject.
d. The applicant should be able to read, understand and prepare sketches, simple drawings and schematics describing the subject.
e. The applicant should be able to apply his knowledge in a practical manner using manufacturer's instructions.
f. The applicant should be able to interpret results from various sources and measurements and apply corrective action where appropriate.

PART-66 - APPENDIX I
BASIC KNOWLEDGE REQUIREMENTS

	LEVELS B1.3 B1.4
12.1 - Theory of Flight – Rotary Wing Aerodynamics	
Terminology;	2
Effects of gyroscopic precession;	
Torque reaction and directional control;	
Dissymmetry of lift, Blade tip stall;	
Translating tendency and its correction;	
Coriolis effect and compensation;	
Vortex ring state, power settling, overpitching;	
Auto-rotation;	
Ground effect.	
12.2 - Flight Control Systems	
Cyclic control;	3
Collective control;	
Swashplate;	
Yaw control: Anti Torque Control, Tail rotor, bleed air;	
Main Rotor Head: Design and Operation features;	
Blade Dampers: Function and construction;	
Rotor Blades: Main and tail rotor blade construction and attachment;	
Trim control, fixed and adjustable stabilizers;	
System operation: manual, hydraulic, electrical and fly-by-wire;	
Artificial feel;	
Balancing and rigging.	
12.3 - Blade Tracking and Vibration Analysis	
Rotor alignment;	3
Main and tail rotor tracking;	
Static and dynamic balancing;	
Vibration types, vibration reduction methods;	
Ground resonance.	
12.4 - Transmission	
Gear boxes, main and tail rotors;	3
Clutches, free wheel units and rotor brake;	
Tail rotor drive shafts, flexible couplings, bearings, vibration dampers and bearing hangers.	
12.5 - Airframe Structures	
(a) Airworthiness requirements for structural strength;	2
Structural classification, primary, secondary and tertiary;	
Fail safe, safe life, damage tolerance concepts;	
Zonal and station identification systems;	
Stress, strain, bending, compression, shear, torsion, tension, hoop stress, fatigue;	
Drains and ventilation provisions;	
System installation provisions;	
Lightning strike protection provision;	

(b) Construction methods of: stressed skin fuselage, formers, stringers, longerons, bulkheads, frames, doublers, struts, ties, beams, floor structures, reinforcement, methods of skinning and anti corrosive protection. 2
Pylon, stabilizer and undercarriage attachments;
Seat installation;
Doors: construction, mechanisms, operation and safety devices;
Windows and windscreen construction;
Fuel storage;
Firewalls;
Engine mounts;
Structure assembly techniques: riveting, bolting, bonding;
Methods of surface protection, such as chromating, anodizing, painting;
Surface cleaning.
Airframe symmetry: methods of alignment and symmetry checks.

12.6 - Air Conditioning (ATA 21)

12.6.1 - Air supply
Sources of air supply including engine bleed and ground cart. 2

12.6.2 - Air conditioning
Air conditioning systems; 3
Distribution systems;
Flow and temperature control systems;
Protection and warning devices.

12.7 - Instruments / Avionic Systems

12.7.1 - Instrument Systems (ATA 31)
Pitot static: altimeter, air speed indicator, vertical speed indicator; 2
Gyroscopic: artificial horizon, attitude director, direction indicator, horizontal situation indicator, turn and slip indicator, turn coordinator;
Compasses: direct reading, remote reading;
Vibration indicating systems - HUMS;
Glass cockpit;
Other aircraft system indication.

12.7.2 - Avionic Systems
Fundamentals of system layouts and operation of: 1
Auto Flight (ATA 22);
Communications (ATA 23);
Navigation Systems (ATA 34).

	LEVELS B1.3 B1.4
12.8 - Electrical Power (ATA 24)	
Batteries Installation and Operation; DC power generation, AC power generation; Emergency power generation; Voltage regulation, Circuit protection. Power distribution; Inverters, transformers, rectifiers; External/Ground power.	3
12.9 - Equipment and Furnishings (ATA 25)	
(a) Emergency equipment requirements; Seats, harnesses and belts; Lifting systems;	2
(b) Emergency flotation systems; Cabin layout, cargo retention; Equipment layout; Cabin Furnishing Installation.	1
12.10 - Fire Protection (ATA 26)	
Fire and smoke detection and warning systems; Fire extinguishing systems; System tests.	3
12.11 - Fuel Systems (ATA 28)	
System layout; Fuel tanks; Supply systems; Dumping, venting and draining; Cross-feed and transfer; Indications and warnings; Refueling and defuelling.	3
12.12 - Hydraulic Power (ATA 29)	
System layout; Hydraulic fluids; Hydraulic reservoirs and accumulators; Pressure generation: electric, mechanical, pneumatic; Emergency pressure generation; Filters; Pressure Control; Power distribution; Indication and warning systems; Interface with other systems.	3

12.13 - Ice And Rain Protection (ATA 30)

Ice formation, classification and detection; Anti-icing and De-icing systems: electrical, hot air and chemical; Rain repellent and removal; Probe and drain heating; Wiper system.	3

12.14 - Landing Gear (ATA 32)

Construction, shock absorbing; Extension and retraction systems: normal and emergency; Indications and warning; Wheels, Tires, brakes; Steering; Air-ground sensing; Skids, floats.	3

12.15 - Lights (ATA 33)

Light sources; External: navigation, landing, taxiing, ice; Internal: cabin, cockpit, cargo; Emergency.	3

12.16 - Pneumatic/Vacuum (ATA 36)

High pressure systems; Medium pressure systems; System layout; Sources: engine/APU, compressors, reservoirs, ground supply; Pressure control; Distribution; Indications and warnings; Interfaces with other systems.	3

12.17 - Integrated Modular Avionics (ATA 42)

Functions that may be typically integrated in the Integrated Modular Avionic (IMA) modules are, among others: Bleed Management, Air Pressure Control, Air Ventilation and Control, Avionics and Cockpit Ventilation Control, Temperature Control, Air Traffic Communication, Avionics Communication Router, Electrical Load Management, Circuit Breaker Monitoring, Electrical System BITE, Fuel Management, Braking Control, Steering Control, Landing Gear Extension and Retraction, Tire Pressure Indication, Oleo Pressure Indication, Brake Temperature Monitoring, etc. Core System; Network Components.	2

12.18 - On Board Maintenance Systems (ATA 45) 2

Central maintenance computers;
Data loading system;
Electronic library system;
Printing;
Structure monitoring (damage tolerance monitoring).

12.19 - Information Systems (ATA 46) 2

Information Systems consist of the units and components which furnish a means of storing, updating and retrieving digital information traditionally provided on paper, microfilm or microfiche. They include units that are dedicated to the information storage and retrieval function such as the electronic library mass storage and controller. They do not include units or components installed for other uses and shared with other systems, such as the flight deck printer or general use display.

Typical examples include:
Air Traffic and Information Management
Systems and Network Server Systems.
Aircraft General Information System;
Flight Deck Information System;
Maintenance Information System;
Passenger Cabin Information System;
Miscellaneous Information System.

HELICOPTER AERODYNAMICS STRUCTURES AND SYSTEMS

SUB-MODULE 01
THEORY OF FLIGHT - ROTARY WING AERODYNAMICS

SUB-MODULE 02
FLIGHT CONTROL SYSTEMS

CONTENTS

SUB-MODULE 06

AIR CONDITIONING (ATA 21)

CONTENTS

SUB-MODULE 07

INSTRUMENTS/AVIONIC SYSTEMS

SUB-MODULE 08
ELECTRICAL POWER (ATA 24)

CONTENTS

SUB-MODULE 09
EQUIPMENT AND FURNISHINGS (ATA 25)

SUB-MODULE 10
FIRE PROTECTION (ATA 26)

SUB-MODULE 11

FUEL SYSTEMS (ATA 28)

SUB-MODULE 12
HYDRAULIC POWER (ATA 29)

SUB-MODULE 15
LIGHTS (ATA 33)

SUB-MODULE 16
PNEUMATIC/VACUUM (ATA 36)

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SUB-MODULE 17
INTEGRATED MODULAR AVIONICS (ATA 42)

SUB-MODULE 18
ON BOARD MAINTENANCE SYSTEMS (ATA 45)

SUB-MODULE 19
INFORMATION SYSTEMS (ATA 46)

HELICOPTER AERODYNAMICS, STRUCTURES AND SYSTEMS

THEORY OF FLIGHT
ROTARY WING AERODYNAMICS

SUB-MODULE 01

PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY – B1.3 B1.4

Sub-Module 01

THEORY OF FLIGHT - ROTARY WING AERODYNAMICS

Knowledge Requirements

12.1 - Theory of Flight – Rotary Wing Aerodynamics	
Terminology;	2
Effects of gyroscopic precession;	
Torque reaction and directional control;	
Dissymmetry of lift, blade tip stall;	
Translating tendency and its correction;	
Coriolis effect and compensation;	
Vortex ring state, power settling, overpitching;	
Auto-rotation;	
Ground effect.	

12.1 - THEORY OF FLIGHT, ROTARY WING AERODYNAMICS

HISTORICAL PERSPECTIVE

The earliest recorded evidence of a rotating wing being used to generate flight dates back to 400 BC, with children in China rotating a stick attached to an airfoil, causing it to generate lift and fly up and out of their hands. (*Figure 1-1*)

This application was a toy, with no thought being given to developing a practical use such as lifting people or objects off the ground. No further development took place until the 1400s, when Leonardo Da Vinci created sketches of a device known as the aerial screw (*Figure 1-2*). The large rotating screw shaped device would force air downward and create lift based on Newton's Third Law of Motion (action and reaction). Da Vinci had a vision of this device lifting objects off the ground, but it never became a reality.

During the 1700s and 1800s a number of inventors continued to develop machines that were intended to lift vertically off the ground and fly, some using the Chinese toy design and some using Da Vinci's design. Some were successful in getting off the ground, but none carried people and none were put to any practical use. Thomas Edison is one of the people who tried to develop a machine that was capable of vertical flight, but he was not successful. Edison did patent a vertical takeoff flying machine in 1910. (*Figure 1-3*) One thing he came to realize was that a machine capable of vertical flight, that could actually carry a useful amount of weight, would require an engine that was powerful and also light in weight. An engine meeting this requirement did not exist until the early 1900s, and even then the power to weight ratio was not very good.

In addition to the problem of not having a powerful lightweight engine, another serious problem encountered by the early designers was the variation in lift generated by the rotating airfoils (blades) as the aircraft starts to move forward. When the aircraft has lifted off the ground but is not moving horizontally, known as being in a hover, the rotating blades all experience the same velocity of air and therefore create the same amount of lift. When the aircraft (rotorcraft) starts moving forward, the blade moving toward the direction of flight has the velocity of the relative wind added to the velocity

Figure 1-1. Chinese Toy.

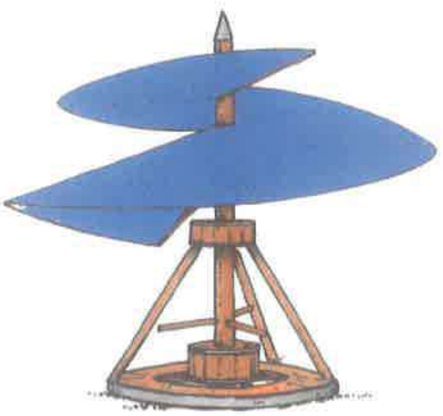


Figure 1-2. Da Vinci Aerial Screw.

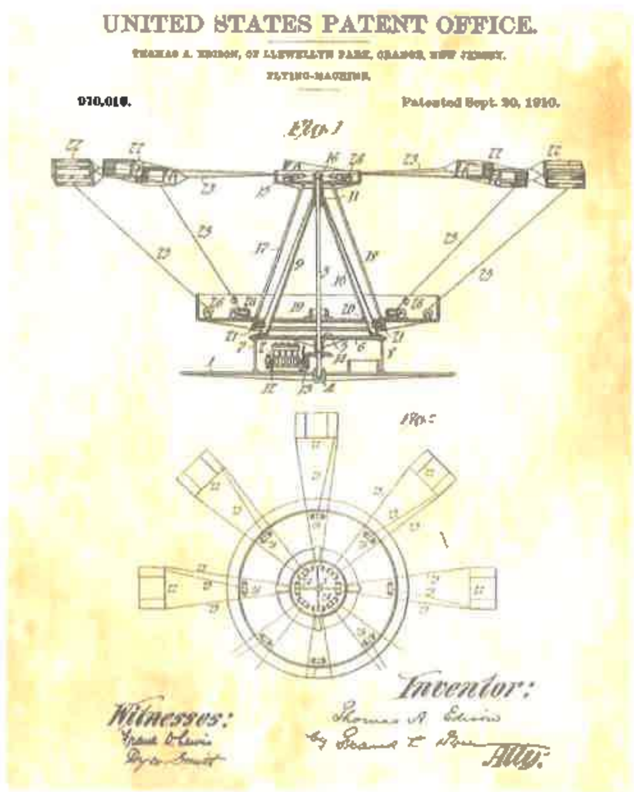


Figure 1-3. Thomas Edison flying machine patent.

created by the rotation of the blade. The blade moving away from the direction of flight has the velocity of the relative wind subtracted from the velocity created by the rotation of the blade. (***Figure 1-4***) This causes the advancing blade to create more lift and the retreating blade to create less lift, potentially causing the rotorcraft to roll and ultimately crash. Many early attempts at forward flight resulted in an accident. Designers had to figure out a way for the blades to automatically change their angle as they transitioned from advancing to retreating, so the lift on the blades would be the same.

The early designers also had to figure out how to make the rotorcraft move in a given direction, without having to resort to the use of a propeller. In the early 1920s designers came up with the idea of what we now refer to as cyclic pitch, which involves changing the pitch of the blades as they pass through various portions of the 360 degrees of rotation. If the blades are at a greater pitch as they pass by the back of the rotorcraft, they will generate more lift and act to propel the rotorcraft forward. The principle of autorotation, where the blades are driven by the force of the air moving through them, was also developed and demonstrated in the early 1920s.

An American inventor named Arthur Young worked with model rotorcraft in the late 1920s, and he developed and patented something known as the stabilizer bar (***Figure 1-5***), which adds stability to the rotating blades when internal and external forces are acting on them. He eventually joined the Bell Aircraft company in 1941, and was given a healthy budget to design and build two working rotorcraft (helicopters). In six months the first Bell Model 1 was built, and then the Bell Model 30 to be followed by the Bell 47. (***Figure 1-5***) The Bell 47 was the first helicopter certified for civilian use in the United States.

During World War II, Germany used helicopters in small numbers for observation, transport and medical evacuation. The Allied Forces also used helicopters, a model known as the R-4 developed by Igor Sikorsky (***Figure 1-6***). The R-4 was primarily used for search and rescue. The German helicopter used a twin main rotor design, similar to what is seen in ***Figure 1-7***, while the Sikorsky design used a single main rotor with an antitorque rotor mounted on the tail boom. The Sikorsky R-4 was the first mass-produced helicopter, with an initial production run of over one hundred.

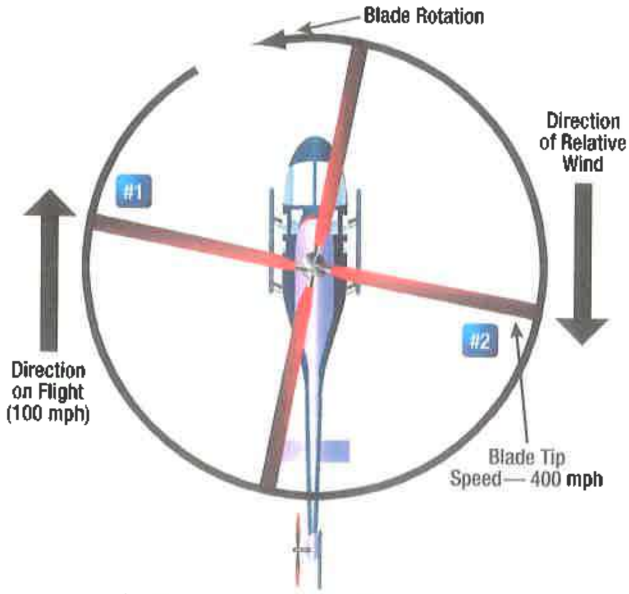


Figure 1-4. Variation in lift with forward flight.

Figure 1-5. Bell 47 helicopter.

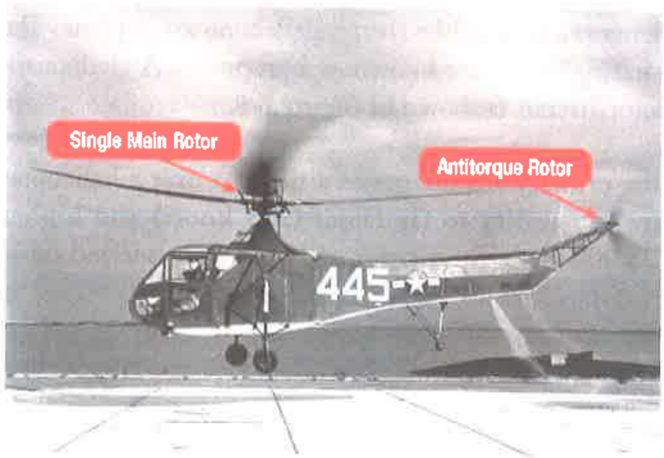


Figure 1-6. Sikorsky R-4 helicopter.

Starting in the 1950s, helicopters began to transition from being powered by the heavier piston engine to the lighter and more powerful turboshaft gas turbine engine. The first helicopter to fly with turbine engine power was a Kaman K-225, which was originally built utilizing a piston engine to power its twin main rotors. (***Figure 1-7***) The U.S. Navy convinced Charles Kaman, the founder of the Kaman helicopter company, to convert a K-225 over to turboshaft power using a Boeing T50 engine. The first turboshaft powered production helicopter was the Sud Aviation (now Aerospatiale) Alouette II. (***Figure 1-8***)

In order to be able to take off vertically, the helicopter's main rotors must be able to create more lift (thrust) than the weight of the aircraft, so the benefit of the light and powerful turboshaft engine is enormous. Although some piston engine powered helicopters are still produced today, the turbine engine powers the majority of current helicopters.

Different types of rotorcraft have been designed and produced over the years, for both civilian and military application, but the one that has dominated the aviation community and continues to dominate is the helicopter. A new type of civil aircraft, with helicopter type capability, is on the horizon. It is like a military aircraft that is currently operating, known as the V-22 Osprey, that takes off like a helicopter and flies like an airplane. It is known as a tiltrotor aircraft. On the tip of each wing there is an engine driving a rotating set of blades. When the engines are in a vertical position, the blades are driven so they act like the main rotor blades on a helicopter, and the aircraft can take off vertically. The engines are able to rotate, with the spinning blades eventually ending up being positioned like the propellers on an airplane. The rotating blades are known as a proprotor. A civilian tilt rotor aircraft is shown in ***Figure 1-9***.

The primary advantages of a tiltrotor over a helicopter are the ability to fly faster (275 knots) and higher (25 000 feet). The aircraft will have a pressurized cabin and supplemental oxygen system, and be equipped to fly into known icing conditions. The drive systems for the two proprotors will be cross-connected with a shaft running through the wing, so if one engine fails the second engine will be able to drive both rotor systems.

INTRODUCTION TO THEORY OF FLIGHT

When looking at an airplane and a helicopter (rotorcraft) sitting next to each other, it would appear that they are two entirely different types of aircraft that operate on different principles. Even though the helicopter can do things the traditional airplane can't, like take off vertically, fly sideways or backwards, and be stationary in flight (hover), both of these aircraft are capable of flight because of wings. The wings on the airplane are fixed while the wings on the helicopter rotate, which is why the helicopter is also referred to as a rotorcraft or rotary wing aircraft.

The wings identified in the previous paragraph are known as airfoils. By definition, an airfoil is an object that generates an action/reaction effect as it moves

Figure 1-7. Kaman K-225 twin main rotor helicopter.

Figure 1-8. Aerospatiale Alouette II helicopter.

Figure 1-9. Civilian tiltrotor aircraft.

through the air, in accordance with Newton's Third Law of Motion. Whether the wing is fixed and attached to another device that is moving through the air, like an airplane, or the wing is attached to a shaft that is rotating on a helicopter, the end result is the same. As the wing moves through the air it generates a lifting force, or thrust, which can be measured in units of newtons. When this force is equal to or greater than the aircraft's weight, it is able to support it in flight.

The rotating airfoil on a helicopter is referred to as a blade, synonymous with another rotating airfoil known as a propeller blade. An example of an airfoil is shown in ***Figure 1-10***, with some of the terminology used to describe it being shown.

The leading edge of the helicopter rotor blade always faces the direction the blade is moving, but since the rotor blades are spinning, the leading edge does not necessarily face the direction the helicopter is moving. As shown in ***Figure 1-4***, the rotating blades are moving in the general direction of flight one half of the time and away from the general direction of flight the other half.

The rotor blade shown in ***Figure 1-10*** has more curvature on the top, known as the camber, than it does on the bottom. Just like the wing of an airplane, this would cause the velocity over the top of the airfoil to be greater than on the bottom, and therefore the pressure on the bottom will be greater than on the top (based on Bernoulli's Principle). This increased pressure on the bottom generates what is referred to as lift. Many helicopter rotor blades are made with the top and bottom camber the same, and are known as symmetrical airfoils. These airfoils control how much pressure difference there is between the top and bottom, and therefore lift, by controlling the angle of attack (***Figure 1-10***).

The ideal situation is to have an equal amount of lift be created along the entire length of the helicopter rotor blade. If the force of lift is noticeably different along the blade's length, the blade will try to bend and it will be subjected to forces that could cause it to fail. Because the rotor blades are spinning, the tip will always be at the greatest velocity because it is moving through the greatest circumference of circle. The inner part of the blade, known as the root end, is moving through the smallest circumference and therefore has the lowest velocity. Due to the difference in air velocity, the

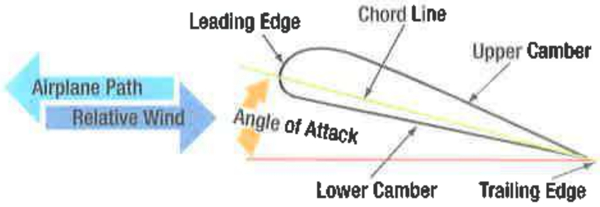


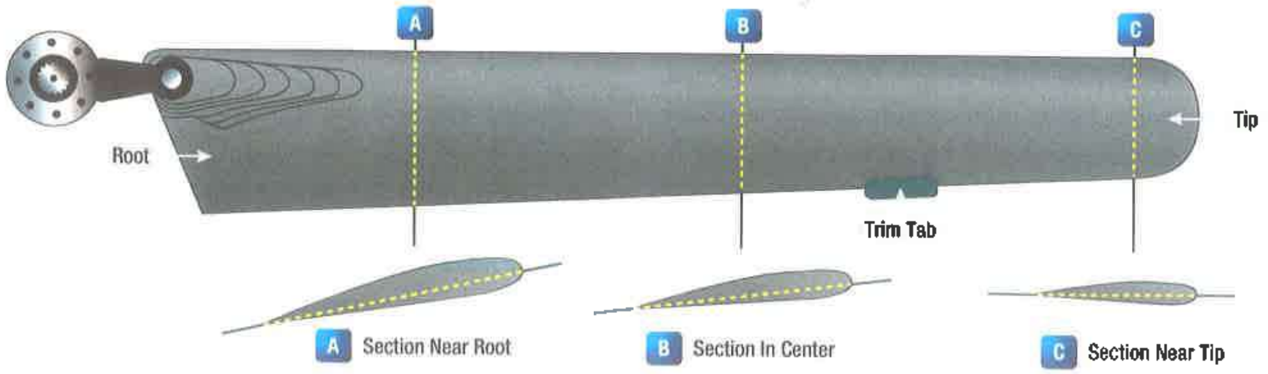
Figure 1-10. Helicopter rotor blade terminology.

blade tips would try to develop the most lift and the inner parts of the blade would produce the least. To compensate for this, the blades are manufactured with a twist that causes the blade angle to be lower at the tip and greater at positions closer to the root end. By matching the velocity of the blade position to the proper blade angle, the workload along the length of the blade is evened out. The twist in the rotor blades can be seen in ***Figure 1-11A***. In ***Figure 1-11B*** the increased blade angle closer to the root can be seen at position "A" and the lesser blade angle closer to the tip at position "C".

TERMINOLOGY

Rotorcraft: A rotorcraft is a type of aircraft that uses a rotating wing to generate the force (lift) necessary to support it while it is in flight. The type of rotorcraft known as a helicopter also uses the rotating wing to generate the force (thrust) necessary to propel it horizontally. Two other types of rotorcraft are the gyroplane and the gyrodyne. The gyroplane's main rotor is not engine driven, and it uses a conventional propeller for forward flight. The main rotor spins as a result of the air moving over the blades as the gyroplane moves forward. The gyrodyne's main rotor is engine driven, but its forward motion is accomplished by using a conventional propeller or the thrust from a turbine engine.


Figure 1-11 (A). Helicopter main rotor blade twist.



Note: "More nose-down" tilt to blade section closer to tip.

Figure 1-11 (B). Rotor blade angle and thickness.

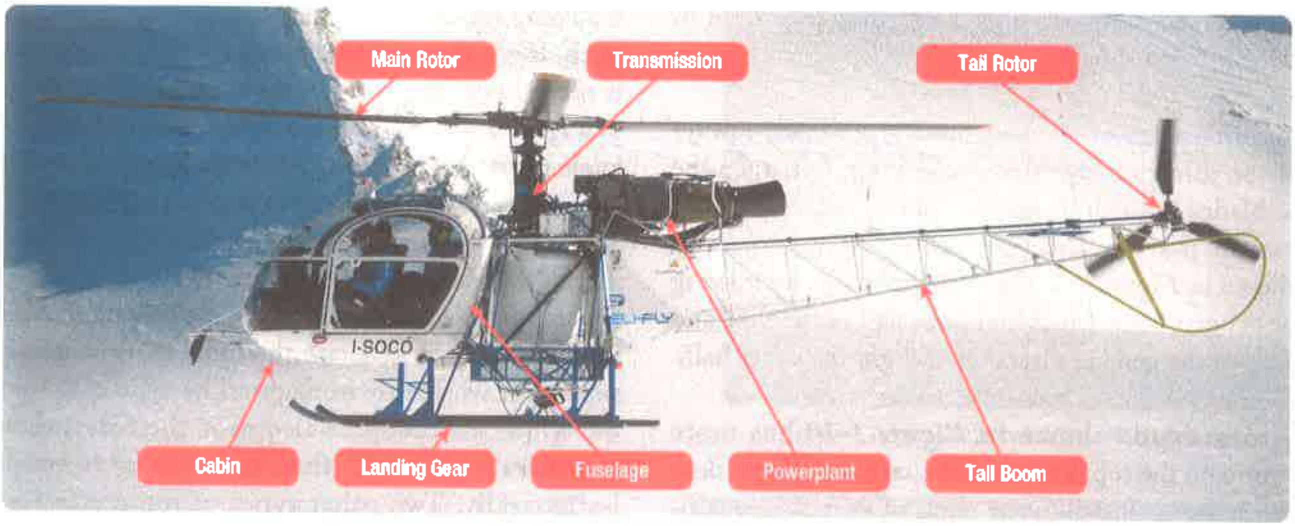


Figure 1-12. Principle parts of a helicopter.

Cabin: The cabin of a rotorcraft, like that of an airplane, houses the pilots and the passengers. The cabin controls seen by the helicopter pilot include the throttle connected to the powerplant, a cyclic pitch lever connected to the main rotor blades, a collective pitch lever connected to the main rotor blades, and two floor mounted pedals that control the tail rotor.

Landing Gear: The landing gear of a rotorcraft, like that of an airplane, supports it when it is not in flight. The landing gear can be skids, as shown in ***Figure 1-12***, floats, like those used on a seaplane, or traditional wheels and tires like those used on most airplanes.

Fuselage: The fuselage is what forms the main body of the rotorcraft, and provides it with the necessary structural integrity. The fuselage can be enclosed by being covered in sheet metal, like the Sikorsky R-4 shown in ***Figure 1-6***, or it can be mostly open tubing like the helicopter shown in ***Figure 1-12***.

Powerplant: The powerplant is the source of energy used to make flight possible. For the helicopter, it drives the main rotor(s), and when applicable the tail rotor. For a gyroplane it drives a conventional propeller and for a gyrodyne it drives the rotors and acts as an energy source for whatever provides horizontal motion. Some rotorcraft are powered by a piston engine, but the majority utilize a turboshaft gas turbine engine. (***Figure 1-12***)

Tail Boom: The tail boom connects the fuselage of the rotorcraft to the most aft location, where the tail rotor is positioned. On a type of helicopter known as a NOTAR (no tail rotor), the tail boom acts as a passageway for air to flow through on its way to being ejected out the back to counteract the torque of the main rotor.

Main Rotor: The main rotor of a rotorcraft is the rotating wings. The rotating wings are airfoils, and they are referred to as rotor blades. On a helicopter and gyrodyne they are driven by the powerplant, and on a gyroplane they are free spinning and driven by the action of the air flowing through them.

Fully Articulated Main Rotor: This type of main rotor has blades that are hinged, so they are able to pivot up and down or fore and aft. This allows the blades to respond to different aerodynamic and mechanical forces that are acting on them, and change their position as they rotate through a full 360 degrees. When the blades pivot up and down it is referred to as blade flap, and when they pivot fore and aft it is referred to as lead/lag. The blades can also change their pitch angle.

Semi-Rigid Main Rotor: This type of main rotor has the blades firmly attached to a rotating hub, with the blades and hub able to teeter like a seesaw. As the blades experience different aerodynamic and mechanical forces acting on them, the entire hub and blade assembly is able to pivot, effectively changing how the blades see the relative wind during their 360 degrees of rotation. The blades can also change their pitch angle.

Rigid Main Rotor: This type of main rotor has the blades firmly attached to a rotating hub that does not pivot, and the blades are able to change their pitch angle. The blades are extremely strong, but they are also flexible. When subjected to aerodynamic and mechanical forces, the blades are able to bend and therefore change the way they see the relative wind during their 360 degrees of rotation.

Tandem Main Rotor: A rotor system that has two main rotors, one located at the front of the helicopter and one located in the back. The use of a tandem rotor cancels out the torque reaction and does away with the need for a tail rotor.

Coaxial Main Rotor: A rotor system that has two main rotors, with one mounted above the other and rotating on concentric shafts with the same axis of rotation. (***Figure 1-13***) The two main rotors rotate in opposite directions, so each main rotor cancels out the torque of the other and no tail rotor is needed. Because of the two counter-rotating main rotors, there is always an advancing blade on each side of the helicopter, meaning the force of lift is equal on both sides.

Transmission: The transmission is driven by the rotorcraft's powerplant, and a shaft coming out of the transmission drives the main rotor. When a rotorcraft has a tail rotor, it is also driven by a shaft coming from the transmission. The RPM of the rotorcraft's powerplant is much too high to drive the main rotor directly, so the transmission acts as a reduction gearbox.

Tail Rotor: The tail rotor consists of two or more airfoils (blades), which act like a propeller to create thrust at the back of the rotorcraft. The tail rotor is also known as the antitorque rotor. When the main rotor blades are driven by the rotorcraft's powerplant, for example in a clockwise direction, there is an equal and opposite force that will try to rotate the aircraft in a counterclockwise direction. This equal and opposite force is known as torque, and the tail rotor counteracts the torque. The thrust from the tail rotor is controlled by the pilot moving the two pedals on the floor of the rotorcraft. (***Figure 1-14***)

Collective Pitch Control: When the pilot lifts up on the collective pitch control (***Figure 1-14***), the blade angle (pitch) of all the main rotor blades increases equally and all the blades create an equal amount of lift. Once the force of lift becomes greater than the weight of the helicopter, the helicopter takes off vertically.

Figure 1-13. Coaxial main rotor helicopter.

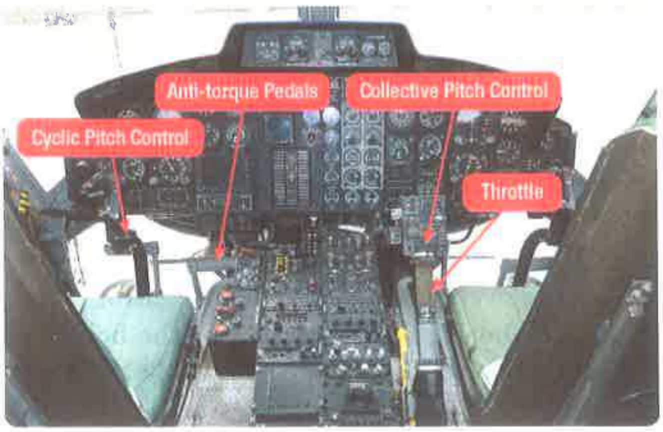


Figure 1-14. Helicopter Flight Deck Controls.

Throttle: The throttle is located on the end of the collective pitch lever (***Figure 1-14***), and operates with a twist of the hand. As the collective pitch lever is lifted up, the throttle rotates to an increased fuel flow position, causing the engine to create more power.

Cyclic Pitch Control: The cyclic pitch control is what causes the helicopter to move horizontally. (***Figure 1-14***) When the cyclic pitch lever is moved forward, the blade angle of the main rotor blades will be greater as they pass by the rear of the helicopter, increasing the lift in this portion of the full rotation. The increased lift in the rear propels the helicopter forward.

Antitorque Pedals: The antitorque pedals (***Figure 1-14***) control the blade angle (pitch) of the blades in the helicopter tail rotor. As the pitch of the blades is increased or decreased, the amount of air they move changes with a corresponding increase or decrease in thrust. When additional engine power is sent to the helicopter's main rotor blades, the pitch of the tail rotor blades has to increase to create more thrust and counteract the torque.

EFFECTS OF GYROSCOPIC PRECESSION

When talking about the aerodynamics of a helicopter main rotor system, it is necessary to first identify which direction the blades rotate when viewed from the top looking down. Unless it is stated otherwise, the concepts presented in this text will be based on the rotor blades turning counterclockwise when viewed from the top. For most helicopters manufactured in the United States, the United Kingdom, Italy, Germany and Japan, the main rotor blades turn counterclockwise. For most helicopters manufactured in France and Russia, the main rotor blades turn clockwise.

When the main rotor blades of a helicopter are spinning they act like a gyroscope, and they experience a characteristic of gyroscopes known as precession. Precession occurs when a force is applied to a rotating object, such as a helicopter main rotor, and the effect is not fully realized until the object has rotated another 90 degrees. When the cyclic pitch control is used to produce horizontal movement of the helicopter (***Figure 1-14***), pitch change rods increase the blade angle on one side and decrease the blade angle on the other side. In order to move forward, there needs to be more lift toward the back of the helicopter and less lift toward the front. Looking at ***Figure 1-15***, the cyclic pitch input to increase blade angle would occur slightly past position #1, and the blade would be at its maximum rise (called flapping up) slightly past position #2. The cyclic pitch input to decrease blade angle would occur slightly past position #3, and the blade would be at its maximum fall (called flapping down) slightly past position #4.

If the cyclic pitch change inputs took place at exactly positions One and Three, the blades would be at maximum flap-up at position Two and maximum flap-down at position Four. This would cause the helicopter to be propelled forward because blade position two is at the rear of the helicopter, and the helicopter would also bank to the right because blade position Two is to the left of the centerline. In ***Figure 1-16*** the tilt of the main rotor can be seen, indicating the helicopter is in forward flight.

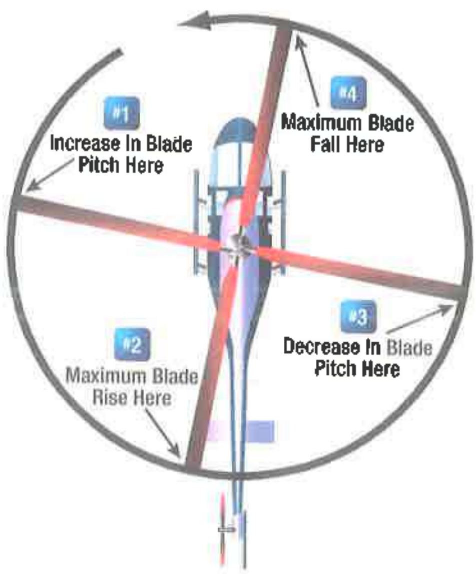


Figure 1-15. Gyroscopic Precession effect on a helicopter main rotor.

Figure 1-16. Tilt of main rotor during forward flight (Sikorsky H-60).

TORQUE REACTION AND DIRECTIONAL CONTROL

TORQUE REACTION

The horsepower of the engine, which powers the helicopter's transmission and ultimately the main rotor shaft, drives the blades of a helicopter's main rotor. According to Sir Isaac Newton's Third Law, for every action there is an equal and opposite reaction. When the output power at the main rotor shaft causes the blades to rotate in a counterclockwise direction (the action), there is an equal and opposite reaction that tries to make the helicopter rotate the opposite way. This equal and opposite reaction is known as torque, and the helicopter must have a means of counteracting it or the helicopter will spin out of control when power is supplied to the main rotor blades.

For a helicopter that has a single main rotor, which is what the majority of today's helicopters have, thrust developed by a tail rotor is the most common method used to counteract torque. The tail rotor is also known as the antitorque rotor. The tail rotor is a set of variable pitch rotating blades located at the end of a tail boom, and driven by a shaft coming from the helicopter's transmission. (***Figure 1-12***) The pilot controls the pitch of the blades by using the pedals located on the floor in front of their seat. (***Figure 1-14***)

When a helicopter pilot is ready to take off vertically, they rotate the throttle on the end of the collective pitch control and simultaneously lift up on the collective pitch control. This increases the engine's power and increases the blade angle of the main rotor blades all at the same time. The increase in blade angle of the main rotor blades creates the lift necessary to support the helicopter, but the force necessary to rotate the blades also creates an enormous amount of torque. To counteract the torque, the pilot will also need to move the floor-mounted pedals that control the blade angle of the tail rotor blades. (***Figure 1-14***)

Depending on the size of the helicopter, between 5 and 25 percent of the engine's power may be needed to drive the helicopter's tail rotor system. This means there is less power available to drive the main rotor that provides the lift for flight, so a limitation is imposed on how much weight the helicopter can carry. Helicopters typically have a vertical fin, much like the vertical stabilizer used on an airplane, and this fin can help compensate for the torque of the main rotor when the helicopter is in forward flight.

A second method of counteracting the torque on a single main rotor helicopter is a technique called the "No Tail Rotor" system, or NOTAR. This system uses a high volume of air that is ejected out of the end of the tail boom, which creates the thrust necessary to counteract the torque. This system will be covered in greater detail in chapter two. Helicopters that have two main rotors spinning in opposite directions do not need a supplemental system to counteract torque, because the two rotors spinning in opposite directions cancel out each other's torque.

Torque is measured in units of newton meters or pound feet, and it is the product of the force being applied (newtons or pounds) multiplied by the distance (meters or feet) from the applied force to the center of the rotating shaft. In ***Figure 1-17***, the torque acting on the main rotor system is shown as 6 779 newton meters, and it is acting in a clockwise direction (opposite of blade rotation). To counteract this torque, the force (thrust) created by the tail rotor, when multiplied by the distance the tail rotor is from the center of the main rotor, must equal 6 779 newton meters. The torque of 6 779 newton meters, divided by the distance shown of 6.1 meters, shows that 1 111 newtons of force is needed from the tail rotor. As shown by the large arrow, the thrust would act in the same direction the main rotor turns because torque acts in the opposite direction.

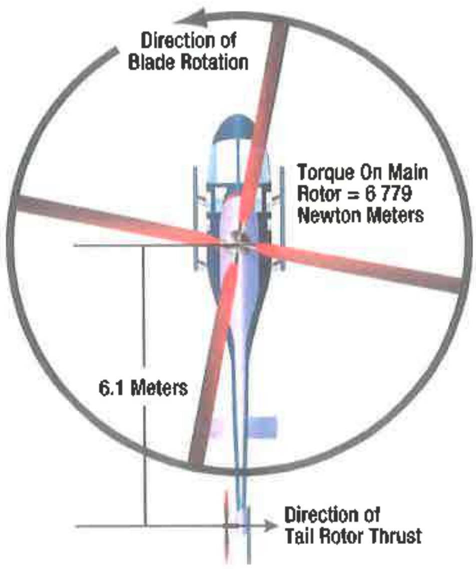


Figure 1-17. Main rotor torque and tail rotor thrust.

If the numbers shown in ***Figure 1-17*** were converted to feet for the distance and pound feet for the torque, the torque would be 5 000 pound feet and the distance would be 20 feet. Using these numbers, the tail rotor would need to create a force of 250 pounds to counteract the torque of the main rotor. The formula for torque, measured in newton meters, is:

Torque = Horsepower × 7 122 ÷ RPM

If the turboshaft engine for a helicopter is supplying 500 horsepower to the main rotor shaft, and the main rotor blades are spinning at 450 RPM, the torque on the main rotor is:

Torque = 500 × 7 122 ÷ 450
Torque = 7 913 Newton meters
To solve for torque in units of pound feet, the engine horsepower is multiplied by 5 252

DIRECTIONAL CONTROL

For a single main rotor helicopter, control around the vertical axis is handled by the tail rotor (antitorque rotor) or from the fan's airflow on a NOTAR type helicopter. Like in an airplane, rotation around this axis is known as yaw. The pilot controls yaw by pushing on the antitorque pedals located on the cabin floor, in the same way the airplane pilot controls yaw by pushing on the rudder pedals. The antitorque pedals can be seen in ***Figure 1-14***.

The tail rotor is used to control the heading of the helicopter while hovering or when making hovering turns, as well as counteracting the torque of the main rotor. Hovering turns are commonly referred to as "pedal turns." At speeds above translational lift, the pedals are used to compensate for torque to put the helicopter in longitudinal trim so that coordinated flight can be maintained. The cyclic control is used to change heading by making a banked turn to the desired direction.

The thrust of the tail rotor depends on the pitch angle of the tail rotor blades. This pitch angle can be positive, negative, or zero. A positive pitch angle tends to move the tail to the right, which matches the direction of blade rotation and is opposite to the torque effect. As the pitch angle approaches zero degrees, there is not enough thrust to counteract the torque and the tail starts moving to the left. At zero degrees of pitch or a negative pitch angle, the tail of the helicopter will move rapidly to the left and cause the fuselage to spin. The maximum positive pitch angle of the tail rotor is greater than the maximum negative pitch angle available. This is because the primary purpose of the tail rotor is to counteract the torque of the main rotor, which is done with positive pitch. The capability for tail rotors to produce thrust to the left (negative pitch angle) is necessary because during autorotation the drag of the transmission tends to yaw the nose to the left, or in the same direction the main rotor is turning.

From the neutral position, applying right pedal causes the nose of the helicopter to yaw right and the tail to swing to the left. (***View A in Figure 1-18***) Pressing on the left pedal has the opposite effect: the nose of the helicopter yaws to the left and the tail swings right. (***View C in Figure 1-18***) When the antitorque pedals are in the neutral position (***View B in Figure 1-18***), the tail rotor has a medium positive pitch angle. In medium positive pitch, the tail rotor thrust approximately equals the torque of the main rotor during cruise flight, so the helicopter maintains a constant heading in level flight.

A vertical fin or stabilizer is used in many single-rotor helicopters to help aid in heading control. The fin is designed to optimize directional stability in flight with a zero tail rotor thrust setting. The size of the fin is crucial to this design. If the surface is too large, the air from the tail rotor may be blocked, heading control could be more difficult at slower airspeeds and in a hover the fin could act as a weathervane.

DISSYMMETRY OF LIFT, BLADE TIP STALL

VERTICAL FLIGHT

When a helicopter initially takes off vertically, and has not yet initiated forward motion, it is considered to be in a hover. Hovering is actually an element of vertical flight. Increasing the blade angle of the rotor blades (pitch) while keeping their rotation speed constant generates additional lift and the helicopter ascends. Decreasing the pitch causes the helicopter to descend. In a no-wind condition in which lift and thrust are less than weight and drag, the helicopter descends vertically. If lift and thrust are greater than weight and drag, the helicopter ascends vertically.

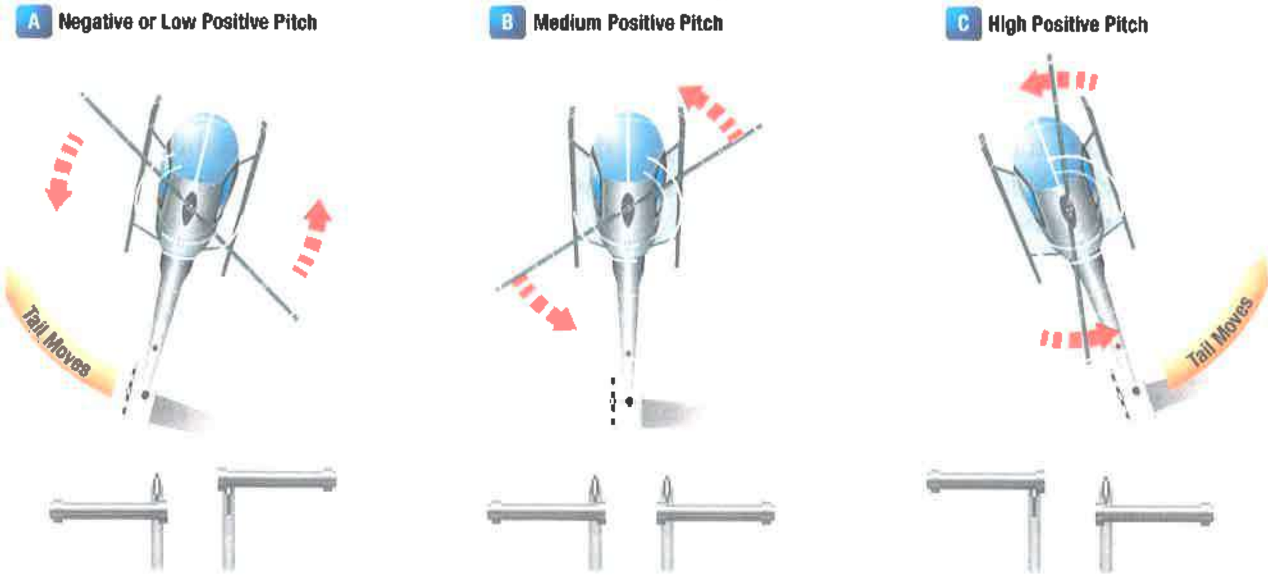


Figure 1-18. Tail rotor pitch angle and helicopter movement in relation to pedal position.

FORWARD FLIGHT

In steady forward flight, with no change in airspeed or vertical speed, the four forces of lift, thrust, drag, and weight must be in balance. Once the main rotor system is tilted forward using cyclic pitch, the total lift-thrust force is also tilted forward. This resultant lift-thrust force can be resolved into two components—lift acting vertically upward and thrust acting horizontally in the direction of flight. In addition to lift and thrust, there is weight (the downward acting force) and drag (the force opposing the motion of an airfoil through the air). (*Figure 1-19*)

In straight and level forward flight at a constant airspeed, lift equals weight and thrust equals drag. If lift exceeds weight, the helicopter accelerates vertically

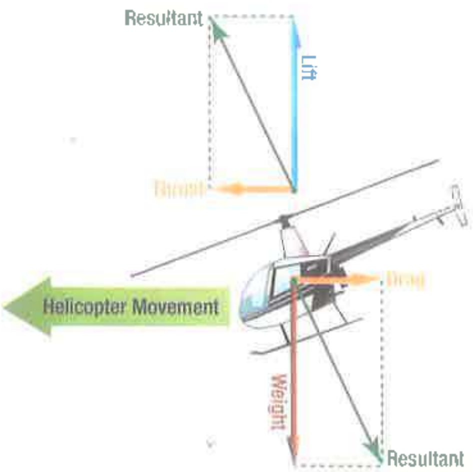


Figure 1-19. Transition from a hover to forward flight with lift converted to thrust.

until the forces are in balance; if thrust is less than drag, the helicopter slows down until the forces are in balance. As the helicopter moves forward, it begins to lose altitude because lift is lost as thrust is diverted forward. However, as the helicopter begins to accelerate from a hover, the rotor system becomes more efficient because it starts experiencing air that is less turbulent. The result is excess power over that which is required to hover. Continued acceleration causes an even larger increase in airflow.

TRANSLATIONAL LIFT

Improved rotor efficiency resulting from directional flight is called translational lift. The efficiency of the hovering rotor system is greatly improved with each knot of incoming wind gained by horizontal movement of the aircraft or by surface wind. As the incoming wind produced by aircraft movement or surface wind enters the rotor system, turbulence and vortices are left behind and the flow of air becomes more horizontal. In addition, the tail rotor becomes more aerodynamically efficient during the transition from hover to forward flight.

Effective Translational Lift (ETL)

While transitioning to forward flight at about 16 to 24 knots, the helicopter goes through Effective Translational Lift (ETL).

As mentioned earlier in the discussion on translational lift, the rotor blades become more efficient as forward airspeed increases. Between 16 and 24 knots, the rotor system completely outruns the recirculation of old vortices and begins to work in relatively undisturbed air. The flow of air through the rotor system is more horizontal; therefore, induced flow and induced drag are reduced. The angle of attack on the blades is effectively increased, which makes the rotor system operate more efficiently. This increased efficiency continues with increased airspeed until the best climb airspeed is reached, and total drag is at its lowest point.

Airflow In Forward Flight

Airflow across the rotor system in forward flight varies from airflow at a hover. In forward flight, air flows opposite the aircraft's flight path. The velocity of this airflow equals the helicopter's forward speed. Because the rotor blades turn in a circular pattern, the velocity of airflow across a blade depends on the position of the blade in the plane of rotation at a given instant, its rotational velocity, and airspeed of the helicopter. Therefore, the airflow meeting each blade varies continuously as the blade rotates. The highest velocity of airflow occurs over the right side (3 o'clock position) of the helicopter (advancing blade in a rotor system that turns counterclockwise) and decreases to rotational velocity over the nose. It continues to decrease until the lowest velocity of airflow occurs over the left side (9 o'clock position) of the helicopter (retreating blade). As the blade continues to rotate, velocity of the airflow then increases to rotational velocity over the tail. It continues to increase until the blade is back at the 3 o'clock position.

The advancing blade in ***Figure 1-20***, position A, moves in the same direction as the helicopter. The velocity of the air meeting this blade equals rotational velocity of the blade plus wind velocity resulting from forward airspeed. The retreating blade (position C) moves in a flow of air moving in the opposite direction of the helicopter. The velocity of airflow meeting this blade equals rotational velocity of the blade minus wind velocity resulting from forward airspeed. The blades (positions B and D) over the nose and tail move essentially at right angles to the airflow created by forward airspeed, so the velocity of airflow meeting these blades equals the rotational velocity. This results in a change to the velocity of airflow all across the rotor disk and causes the blades to create different amounts of lift in the full 360 degrees of rotation.

Dissymmetry Of Lift

Dissymmetry of lift is the differential (unequal) lift between advancing and retreating halves of the rotor disk caused by the different wind flow velocity across each half. This difference in lift would cause the helicopter to be uncontrollable in any situation other than hovering in a calm wind. There must be a means of compensating, correcting, or eliminating this unequal lift to attain symmetry of lift. When the helicopter moves through the air, the relative airflow through the main rotor disk is different on the advancing side than on the retreating side. The relative wind encountered by the advancing blade is increased by the forward speed of the helicopter, while the relative wind speed acting on the retreating blade is reduced by the helicopter's forward airspeed. Therefore, as a result of the relative wind speed, the advancing blade side of the rotor disk can produce more lift than the retreating blade side. (***Figure 1-20***)

If this condition were allowed to exist, a helicopter with a counterclockwise main rotor blade rotation would roll to the left because of the difference in lift. In reality,

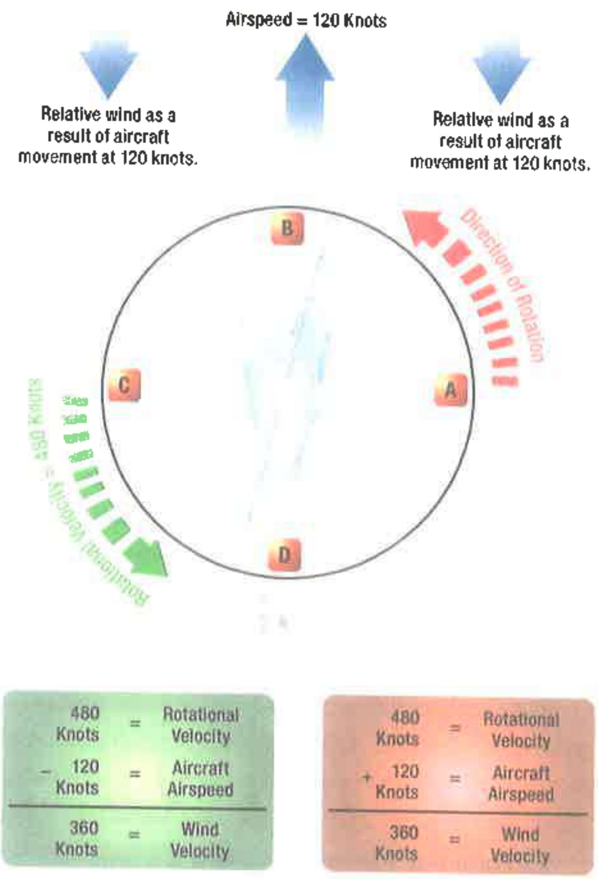


Figure 1-20. Velocity of airflow over the blades in forward flight.

the main rotor blades flap and feather automatically to equalize lift across the rotor disk. Articulated rotor systems, usually with three or more blades, incorporate a horizontal hinge (flapping hinge) to allow the individual rotor blades to move, or flap up and down as they rotate. A semi-rigid rotor system (two blades) utilizes a teetering hinge, which allows the blades to flap as a unit. When one blade flaps up, the other blade flaps down.

As shown in ***Figure 1-21***, as the rotor blade reaches the advancing side of the rotor disk (A), it reaches its maximum up flap velocity. When the blade flaps upward, the angle between the chord line and the resultant relative wind decreases. This decreases the angle of attack, which reduces the amount of lift produced by the blade. At position (C), the rotor blade is now at its maximum down flapping velocity. Due to down flapping, the angle between the chord line and the resultant relative wind increases. This increases the angle of attack and thus the amount of lift produced by the blade.

Blade Tip Stall

In forward flight, the relative airflow through the main rotor disk is different on the advancing and retreating side. The relative airflow over the advancing side is higher due to the forward speed of the helicopter, while the relative airflow on the retreating side is lower. This dissymmetry of lift increases as forward speed increases.

To generate the same amount of lift across the rotor disk, the advancing blade flaps up while the retreating blade flaps down. This causes the angle of attack to decrease on the advancing blade, which reduces lift, and increase on the retreating blade, which increases lift. At some point as the forward speed increases, the low blade speed on the retreating blade, and its high angle of attack cause a stall and loss of lift.

Retreating blade stall is a major factor in limiting a helicopter's Never-Exceed Speed (VNE) and its development can be felt by a low frequency vibration, pitching up of the nose, and a roll in the direction of the retreating blade. High weight, low rotor RPM, high-density altitude, turbulence, and steep abrupt turns are all conducive to retreating blade stall at high forward airspeeds. As altitude is increased, higher blade angles are required to maintain lift at a given airspeed. Thus, retreating blade stall is encountered at a lower forward airspeed at altitude. Most manufacturers publish charts and graphs showing a VNE decrease with altitude.

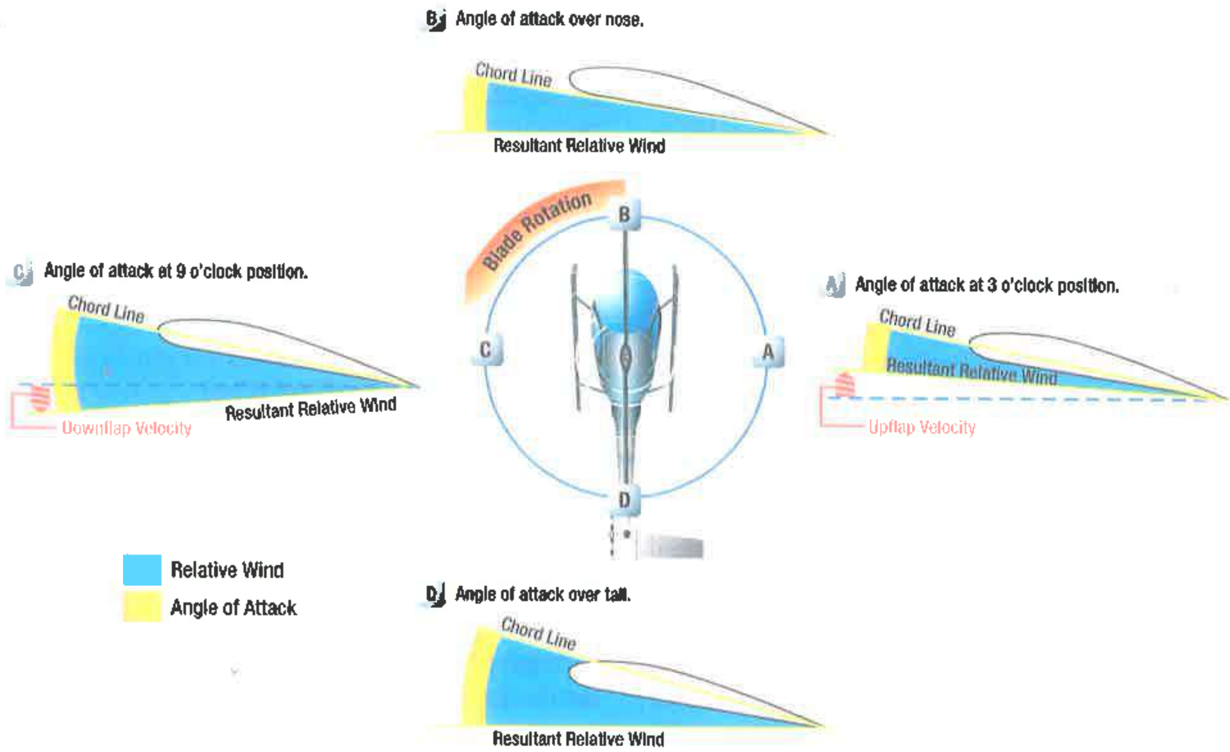


Figure 1-21. Equalized lift across the main rotor disk, caused by upward and downward flapping of blades.

Covered earlier in this chapter was the concept of rotational velocity at various positions along the length of the main rotor blade, and the fact that the tip spins fastest. Because the inner portions of the blade are rotating slower than the outer portions (less circumference of circle), on a retreating blade the inner portions lose the ability to create lift first. This is because at some point the forward speed of the helicopter will equal the aft rotating speed of the blade, leaving the blade with no effective airflow and therefore no lift. This leaves the outer portion of the blade with the responsibility to create most of the lift, and therefore it must operate with the highest angle of attack. The high angle of attack near the tip of the retreating blade can cause it to stall. This is why retreating blade stall is also referred to as blade tip stall.

When recovering from a retreating blade stall condition, moving the cyclic aft only worsens the stall as aft cyclic produces a flare effect, thus increasing the angle of attack. Pushing forward on the cyclic also deepens the stall as the angle of attack on the retreating blade is increased. Correct recovery from retreating blade stall requires the collective pitch to be lowered first, which reduces blade angles and thus angle of attack. Aft cyclic can then be used to slow the helicopter.

TRANSLATING TENDENCY AND CORRECTION

During hovering flight, a single main rotor helicopter tends to move in the direction of tail rotor thrust. This lateral or sideward movement is called translating tendency or drift. (*Figure 1-22*) To counteract this tendency, one or more of the following features may be used for a counterclockwise rotating main rotor.

- The main transmission can be mounted at a slight angle to the left (when viewed from behind) so that the rotor mast has a built-in tilt to oppose the tail rotor thrust.
- Flight controls can be rigged so that the rotor disk is tilted to the left slightly when the cyclic is centered.

Whichever method is used, the tip-path plane is tilted slightly to the left when the helicopter is in a hover. If the transmission is mounted so the rotor shaft is vertical with respect to the fuselage, the helicopter "hangs" left side low in a hover. The opposite is true for rotor systems turning clockwise when viewed from above. The helicopter fuselage will also be tilted when the tail rotor

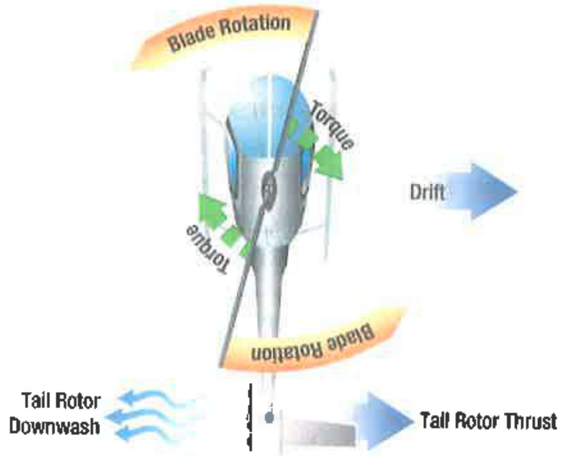


Figure 1-22. Translation tendency causing helicopter to drift to the right.

is below the main rotor disk and supplying antitorque thrust. The fuselage tilt is caused by the imperfect balance of the tail rotor thrust against the main rotor torque in the same plane. The helicopter tilts due to two separate forces, the main rotor disk tilts to neutralize the translating tendency and the lower tail rotor thrust below the plane of the torque action.

In forward flight, the tail rotor continues to push to the right, and the helicopter makes a small angle with the wind when the rotors are level and the turn and bank indicator ball is in the middle. This is called inherent sideslip. For some larger helicopters, the vertical fin or stabilizer is often designed with the tail rotor mounted on them to correct this sideslip and to eliminate some of the tilting in a hover. Also, by mounting the tail rotor on top of the vertical fin or pylon, the antitorque is more in line with or closer to the horizontal plane of torque, resulting in less airframe (or body) lean from the tail rotor. Having the tail rotor higher off the ground reduces the risk of objects coming in contact with the blades, but at the cost of increased weight and complexity.

CORIOLIS EFFECT AND COMPENSATION

CONING

In order for a helicopter to generate lift, the rotor blades must be turning. Rotor system rotation causes the blades to move through the air, creating a relative wind component without having to move the airframe through the air as with an airplane or glider. Depending on the motion of the blades and helicopter airframe, many factors cause the relative wind direction to vary. The rotation of the rotor system creates centripetal acceleration, which causes a centrifugal force to throw the blades straight outward from the main rotor hub. The faster the rotation, the greater the centrifugal force; and the slower the rotation, the smaller the centrifugal force. The maximum centrifugal force generated is determined by the maximum operating rotor Revolutions Per Minute (RPM) and the diameter of the rotor system.

The formula for Centrifugal Force (Cf) is:

$$Cf = (\text{Weight}/\text{Gravity}) \times \text{Velocity}^2 \div \text{Radius}$$

The weight of the blade would be measured in kilograms or pounds, gravity would be 9.8 meters/second^2 or 32.2 feet per second^2, velocity would be in meters per second or feet per second (at the blade's Center of Gravity), and the radius of the rotor system would be in meters or feet (at the blade's Center of Gravity). On the Bell Jet Ranger helicopter, the blade weighs 45.4 kilograms (100 pounds), the velocity at the blade CG is 105 meters per second (345 feet per second), and the rotor system radius at the blade CG is 2.5 meters (8.25 feet). Based on these numbers, the centrifugal force on the blade would be:

$$Cf = 45.4/9.8 \times 1\,052 \div 2.5$$
$$Cf = 20\,430 \text{ kilograms}$$

The calculation above shows there is a tremendous amount of force trying to rip the helicopter's rotor blade out of the hub. This is the most dominant force affecting the helicopter's main rotor system, and because of it the strength of the blade and the method of blade attachment is critical. A person performing inspections on a helicopter must be aware of the forces acting on the blades and use every means available to ensure they are airworthy.

As lift on the blades is increased during takeoff, the two major forces acting on the blades are centrifugal force acting outward and lift acting upward. The result of these two forces is that the blades assume a conical path instead of remaining in a plane of rotation perpendicular to the mast. This can be seen in any helicopter when it takes off, with the rotor disk changing from flat to a slight cone shape. (***Figure 1-23***) If the rotor RPM is allowed to go too low (below the minimum power-on rotor RPM, for example), the centrifugal force becomes smaller and the coning angle becomes much larger.

Figure 1-23. Coning of main rotor during takeoff.

Coriolis Effect

The Coriolis Effect is also referred to as the law of conservation of angular momentum. It states that the value of angular momentum of a rotating body does not change unless an external force is applied. In other words, a rotating body continues to rotate with the same rotational velocity until some external force is applied to change the speed of rotation. Angular momentum is the moment of inertia (mass times distance from the center of rotation squared) multiplied by the speed of rotation.

Changes in angular velocity, known as angular acceleration and deceleration, take place as the mass of a rotating body is moved closer to or farther away from the axis of rotation. The speed of the rotating mass varies proportionately with the square of the radius. An excellent example of this principle in action is a figure skater performing a spin on ice skates. The skater begins rotation on one foot, with the other leg and both arms extended. The rotation of the skater's body is relatively slow. When the skater draws both arms and the second leg inward, the moment of inertia (mass times radius squared) becomes much smaller and the skater starts rotating significantly faster. Because the angular momentum will remain the same (no external force applied), the angular velocity must increase.

The rotor blades rotating about the rotor hub possess angular momentum. As the rotor begins to cone due to increased lift on takeoff or to G-loading maneuvers, the diameter of the rotor disk shrinks. Due to conservation of angular momentum, the blades continue to travel of the same speed even though the blade tips have a shorter distance to travel due to reduced disk diameter. The action results in an increase in rotor RPM that causes a slight increase in lift. Most pilots arrest this increase of RPM with an increase in collective pitch. This increase in blade RPM lift is somewhat negated by the slightly smaller disk area as the blades cone upward.

VORTEX RING STATE, POWER SETTLING, OVERPITCHING

VORTEX RING STATE/POWER SETTLING

Vortex ring state describes an aerodynamic condition in which a helicopter may be in a vertical descent with 20 percent up to maximum power applied, and little or no climb performance. The term "settling with power" comes from the fact that the helicopter keeps settling (descending) even though full engine power is applied.

In a normal Out-of-Ground-Effect (OGE) hover, the helicopter is able to remain stationary by propelling a large mass of air down through the main rotor. Some of the air is recirculated near the tips of the blades, curling up from the bottom of the rotor system and rejoining the air entering the rotor from the top. This phenomenon is common to all airfoils and is known as tip vortices. Tip vortices generate drag and degrade airfoil efficiency. As long as the tip vortices are small, their only effect is a small loss in rotor efficiency. However, when the helicopter begins to descend vertically, it settles into its own downwash and this greatly enlarges the tip vortices. In this vortex ring state, most of the power developed by the engine is wasted in circulating the air in a doughnut pattern around the rotor.

In addition, the helicopter may descend at a rate that exceeds the normal downward induced-flow rate of the inner blade sections. As a result, the airflow of the inner blade sections is upward relative to the disk. This produces a secondary vortex ring in addition to the normal tip vortices. The secondary vortex ring is generated about the point on the blade where the airflow changes from up to down. The result is an unsteady turbulent flow over a large area of the disk. Rotor efficiency is lost even though power is still being supplied from the engine. (*Figure 1-24*)

A fully developed vortex ring state is characterized by an unstable condition in which the helicopter experiences uncommanded pitch and roll oscillations, has little or no collective authority, and achieves a descent rate that may approach 6 000 feet per minute if allowed to develop. A vortex ring state may be entered during any maneuver that places the main rotor in a condition of descending in a column of disturbed air and low forward airspeed. Airspeeds that are below translational lift airspeeds are within this region of susceptibility to settling with power aerodynamics. This condition is sometimes seen during quick-stop type maneuvers or during recovery from autorotation.

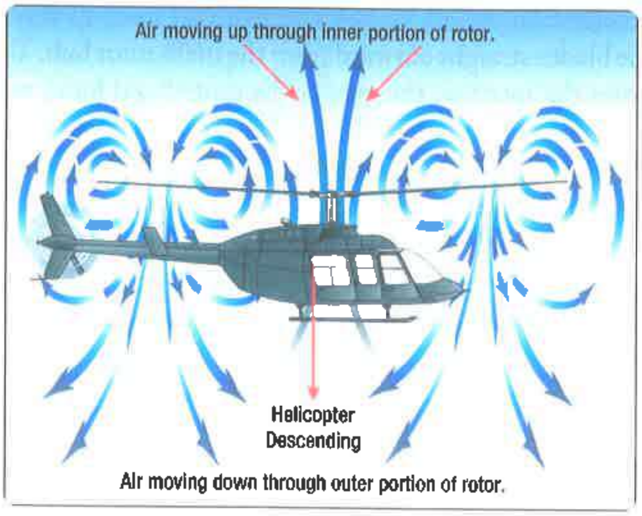


Figure 1-24. Helicopter descending with vortexes forming on the blades.

The following combination of conditions is likely to cause settling in a vortex ring state in any helicopter:

1. A vertical or nearly vertical descent of at least 300 feet per minute. The actual critical rate depends on the gross weight, rotor RPM, density altitude, and other pertinent factors.
2. The rotor system must be using some of the available engine power (20–100 percent).
3. The horizontal velocity must be slower than effective translational lift.

Some of the situations that are conducive to a settling with power condition are:

1. Any hover above ground effect altitude, specifically attempting to hover out of ground effect at altitudes above the hovering ceiling of the helicopter.
2. Attempting to hover out of ground effect without maintaining precise altitude control.
3. Landing on a rooftop helipad when the wind is not aligned with the landing direction.
4. Downwind and steep power approaches in which airspeed is permitted to drop below 10 knots depending on the type of helicopter.

OVERPITCHING

The pitch of a helicopter's main rotor blades, also referred to as the blade angle, is the relationship between the chord line of the airfoil and the plane of rotation for the blade. Overpitching, as the name implies, is when the collective pitch lever is raised to the extreme and blade angle increases to a level that cannot be supported by available engine power. When this happens the RPM of the rotor system will decrease, and with a reduced RPM there will be a reduced centrifugal force. With less centrifugal force and the main rotor blades at a very high blade angle, the blades will rise to a much steeper cone angle. In this situation the helicopter will not be able to maintain altitude and will start descending through very disturbed air, which means a vortex ring state and power settling is getting ready to occur.

Before the helicopter gets to the state that overpitching causes a loss of rotor RPM and the loss of altitude, the pilot needs to reduce the collective pitch so that the available engine power can bring the RPM back to the proper value.

AUTOROTATION

Autorotation is the state of flight where the main rotor system of a helicopter is being turned by the action of air moving up through the rotor rather than engine power driving the rotor. In normal, powered flight, air is drawn into the main rotor system from above and exhausted downward, but during autorotation, air moves up into the rotor system from below as the helicopter descends. Autorotation is possible mechanically by a freewheeling unit, which is a special clutch mechanism that allows the main rotor to continue turning even if the engine is not running. If the engine fails, the freewheeling unit automatically disengages the engine from the main rotor allowing the main rotor to rotate freely. It is the means by which a helicopter can be landed safely in the event of an engine failure; consequently, all helicopters must demonstrate this capability in order to be certified. (***Figure 1-25***)

HOVERING AUTOROTATION

Most autorotations are performed with forward speed. For simplicity, the following aerodynamic explanation is based on a vertical autorotative descent (no forward speed) in still air. Under these conditions, the forces that cause the blades to turn are similar for all blades regardless of their position in the plane of rotation. Therefore, dissymmetry of lift resulting from helicopter airspeed is not a factor.

During vertical autorotation, the rotor disk is divided into three regions (as illustrated in ***Figure 1-26***): (1) driven region, (2) driving region, and (3) stall region. ***Figure 1-27*** shows three blade sections that illustrate force vectors. Part A is the driven region, B and D are points of equilibrium, part C is the driving region, and part E is the stall region. Force vectors are different in each region because rotational relative wind is slower near the blade root and increases continually toward the blade tip. Also, blade twist gives a more positive angle in the driving region than in the driven region. The combination of the inflow up through the rotor with rotational relative wind produces different combinations of aerodynamic force at every point along the blade.

The driven region, also called the propeller region, is nearest the blade tips. Normally, it consists of about 30 percent of the radius. In the driven region, part A of ***Figure 1-27***, the Total Aerodynamic Force (TAF) acts

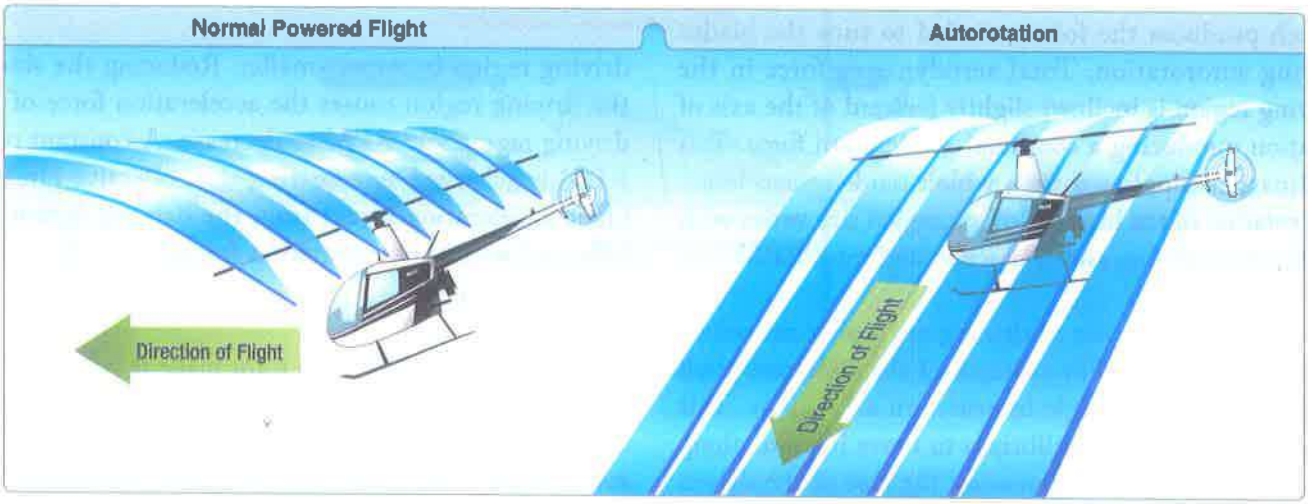


Figure 1-25. Main rotor airflow during normal flight and autorotation.

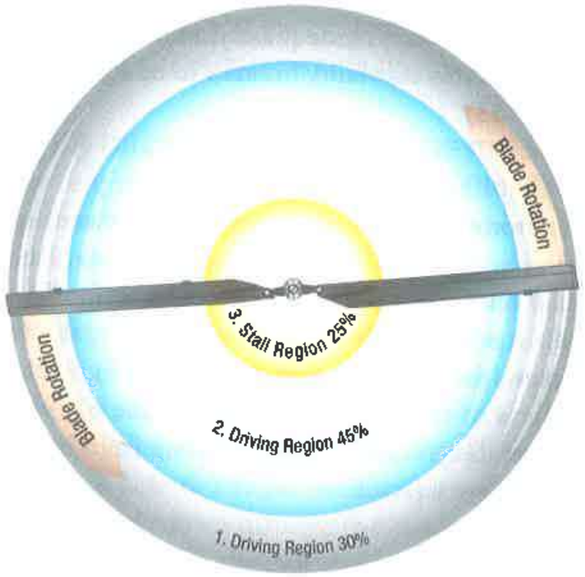


Figure 1-26. Blade regions during an autorotation descent.

behind the axis of rotation, resulting in an overall drag force. The driven region produces some lift, but that lift is offset by drag. The overall result is a deceleration in the rotation of the blade. The size of this region varies with the blade pitch, rate of descent, and rotor RPM. When changing autorotative RPM blade pitch, or rate of descent, the size of the driven region in relation to the other regions also changes.

There are two points of equilibrium on the blade, one between the driven region and the driving region, and one between the driving region and the stall region. At points of equilibrium, TAF is aligned with the axis of rotation. Lift and drag are produced, but the total effect produces neither acceleration nor deceleration. The driving region, or autorotative region, normally lies between 25 to 70 percent of the blade radius. Part C of ***Figure 1-27*** shows the driving region of the blade, which produces the forces needed to turn the blades during autorotation. Total aerodynamic force in the driving region is inclined slightly forward of the axis of rotation, producing a continual acceleration force. This inclination supplies thrust, which tends to accelerate the rotation of the blade. Driving region size varies with blade pitch setting, rate of descent, and rotor RPM.

By controlling the size of this region, a pilot can adjust autorotative RPM. For example, if the collective pitch is raised the pitch angle increases in all regions. This causes the point of equilibrium to move inboard along the blade's span, thus increasing the size of the driven region. The stall region also becomes larger while the driving region becomes smaller. Reducing the size of the driving region causes the acceleration force of the driving region and RPM to decrease. A constant rotor RPM is achieved by adjusting the collective pitch so blade acceleration forces from the driving region are balanced with the deceleration forces from the driven and stall regions.

Figure 1-27. Driven, driving and stall regions during autorotation.

The inner 25 percent of the rotor blade is referred to as the stall region and operates above its maximum angle of attack (stall angle), causing drag, which tends to slow rotation of the blade. Part E of ***Figure 1-27*** depicts the stall region.

GROUND EFFECT

IN GROUND EFFECT (IGE)

Ground effect is the increased efficiency of the rotor system caused by interference of the airflow when near the ground.

The air pressure or density is increased, which acts to decrease the downward velocity of air. Ground effect permits relative wind to be more horizontal, lift vector to be more vertical, and induced drag to be reduced. These conditions allow the rotor system to be more efficient. Maximum ground effect is achieved when hovering over smooth hard surfaces. When hovering over surfaces as tall grass, trees, bushes, rough terrain, and water, maximum ground effect is reduced. Rotor efficiency is increased by ground effect to a height of about one rotor diameter (measured from the ground to the rotor disk) for most helicopters. Since the induced flow velocities are decreased, the angle of attack is increased, which requires a reduced blade pitch angle and a reduction in induced drag. This reduces the power required to hover In Ground Effect. (***Figure 1-28***)

OUT OF GROUND EFFECT (OGE)

The benefit of placing the helicopter near the ground is lost above "In Ground Effect" altitude. Above this altitude, the power required to hover remains nearly constant, given similar conditions (such as wind). Induced flow velocity is increased, resulting in a decrease in angle of attack and a decrease in lift. Under the correct circumstances, this downward flow can become so localized that the helicopter will sink at alarming rates. This effect is called settling with power and was discussed earlier in this chapter. A higher blade pitch angle is required to maintain the same angle of attack that was present when flying an in ground effect hover. The increased pitch angle also creates more drag. This increased pitch angle and drag requires more power to hover OGE than IGE. (***Figure 1-29***)

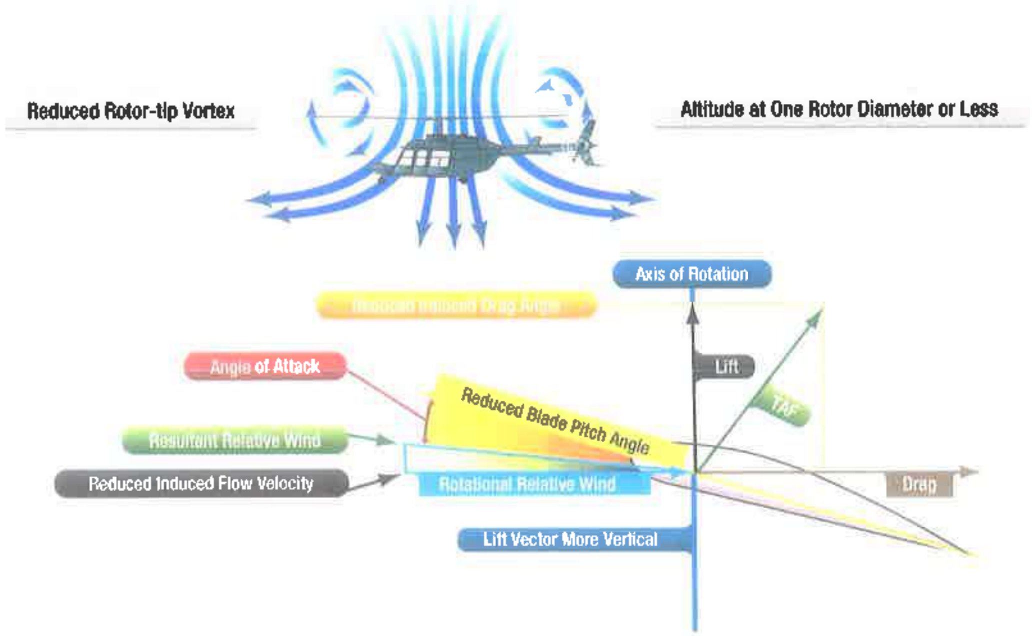


Figure 1-28. Hovering In Ground Effect (IGE).

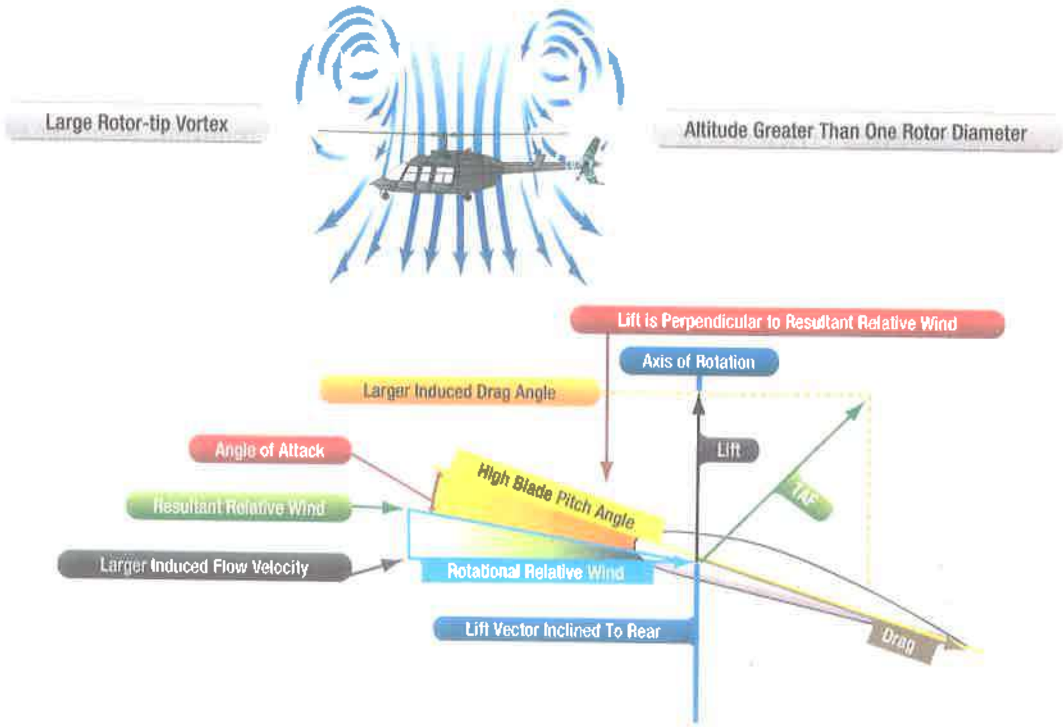


Figure 1-29. Hovering Out of Ground Effect (OGE).

Question: 1-1
Name 4 helicopter configuration types which do not require a tail rotor.

Question: 1-2
In which segment of a main rotor blades rotation is the angle of attack of the blade the highest?

Question: 1-3
In what way do the foot pedals on the flight deck control the amount of thrust generated by the tail rotor?

Question: 1-4
__________ = Horsepower × 7 122 ÷ RPM

Question: 1-5
Why does the efficiency of a the main rotor blades increase as the helicopter reaches translational speeds?

Question: 1-6
In what way would a pilot counter the Coriolis effect?

Question: 1-7
In what way does Vortex Ring State effect the performance of a helicopter?

Question: 1-8
What is the purpose of the freewheeling unit on a helicopter?

ANSWERS

Answer: 1-1
Tandem rotor configuration, coaxial rotor configuration, Tilt rotor configuration, NOTAR configuration.

Answer: 1-2
When in the retreating direction and approximately 90° from the fuselage centerline of the helicopter.

Answer: 1-3
The pedals vary the angle of attack of the tail rotor blades.

Answer: 1-4
Torque.

Answer: 1-5
At translational speeds (16-24 knots) the helicopter begins to move away from the turbulence and vortices generated by the main rotor when in slower flight or a hover.

Answer: 1-6
By slightly increasing the collective pitch, thus countering the increased RPM due to the smaller effective rotor disk size.

Answer: 1-7
A hovering helicopter descends through its own turbulence even though full power is applied.

Answer: 1-8
It allows the main rotor to disengage from the engine in case the engine fails, and so making autorotation possible.

HELICOPTER AERODYNAMICS, STRUCTURES AND SYSTEMS

FLIGHT CONTROL SYSTEMS

SUB-MODULE 02

PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY →	B1.3 B1.4

Sub-Module 02

FLIGHT CONTROL SYSTEMS

Knowledge Requirements

12.2 - Flight Control Systems	
Cyclic control;	3
Collective control;	
Swashplate;	
Yaw control: anti torque control, tail rotor, bleed air;	
Main rotor head: design and operation features;	
Blade dampers: function and construction;	
Rotor blades: main and tail rotor blade construction and attachment;	
Trim control, fixed and adjustable stabilizers;	
System operation: manual, hydraulic, electrical and fly-by-wire;	
Artificial feel;	
Balancing and rigging.	

12.2 - FLIGHT CONTROL SYSTEMS

INTRODUCTION

There are three major controls in a helicopter used by the pilot during flight: the collective pitch control, the cyclic pitch control, and the anti-torque pedals or tail rotor control. Sometimes in addition to these major controls, the pilot must also use the throttle control, which is usually mounted directly to the collective pitch control for controlling engine power. (***Figure 2-1***)

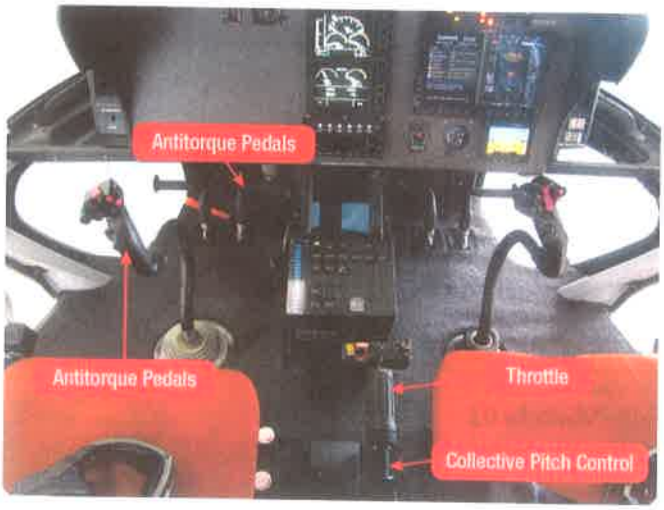


Figure 2-1. Helicopter controls on the flight deck.

The control systems described in this chapter are not limited to the single main rotor type helicopter but are used in one form or another in most helicopter configurations. All examples refer to a counterclockwise main rotor blade rotation, such as viewed from above. If flying a helicopter with a clockwise rotation, left and right references must be reversed, particularly in the areas of rotor blade pitch change, anti-torque pedal movement, and tail rotor thrust.

CYCLIC CONTROL

The cyclic pitch control is usually projected upward from the cockpit floor, between the pilot's legs or in some models between the two pilot's seats. (***Figure 2-2***)


Figure 2-2. Location of cyclic pitch control.

The purpose of the cyclic-pitch control is to cause the tip-path plane of the main rotor to tilt as required to provide for movement of the helicopter in a desired direction: forward, rearward, left, and right.

As discussed in *Sub-Module 01*, the total lift force is always perpendicular to the tip-path plane of the main rotor. The main rotor tilts in the direction called for by the control, and the helicopter moves as directed. When the control stick is in neutral, the helicopter remains stationary in the air (hover). If wind is present, the helicopter drifts in the direction of wind unless sufficient cyclic movement (rotor thrust) is applied to cancel the effect of the wind.

The rotor disk is also subject to the effect of gyroscopic precession (as described in *Sub-Module 01*). To counteract this effect, the mechanical linkages for the cyclic control rods are rigged in such a way that they decrease the pitch angle of each rotor blade at approximately 90° in the point of the rotor's rotation as it reaches the direction of cyclic displacement (***Figure 1-17***) and increases the pitch angle of the rotor blade at approximately 90° after it passes the direction of displacement. An increase in pitch angle increases the angle of attack; a decrease in pitch angle decreases the angle of attack. When the cyclic is moved forward, the angle of attack decreases as the rotor blade passes the right side of the helicopter (with the advancing blade in a rotor system turning counterclockwise) and increases on the left side (on the retreating blade). This results in a maximum downward

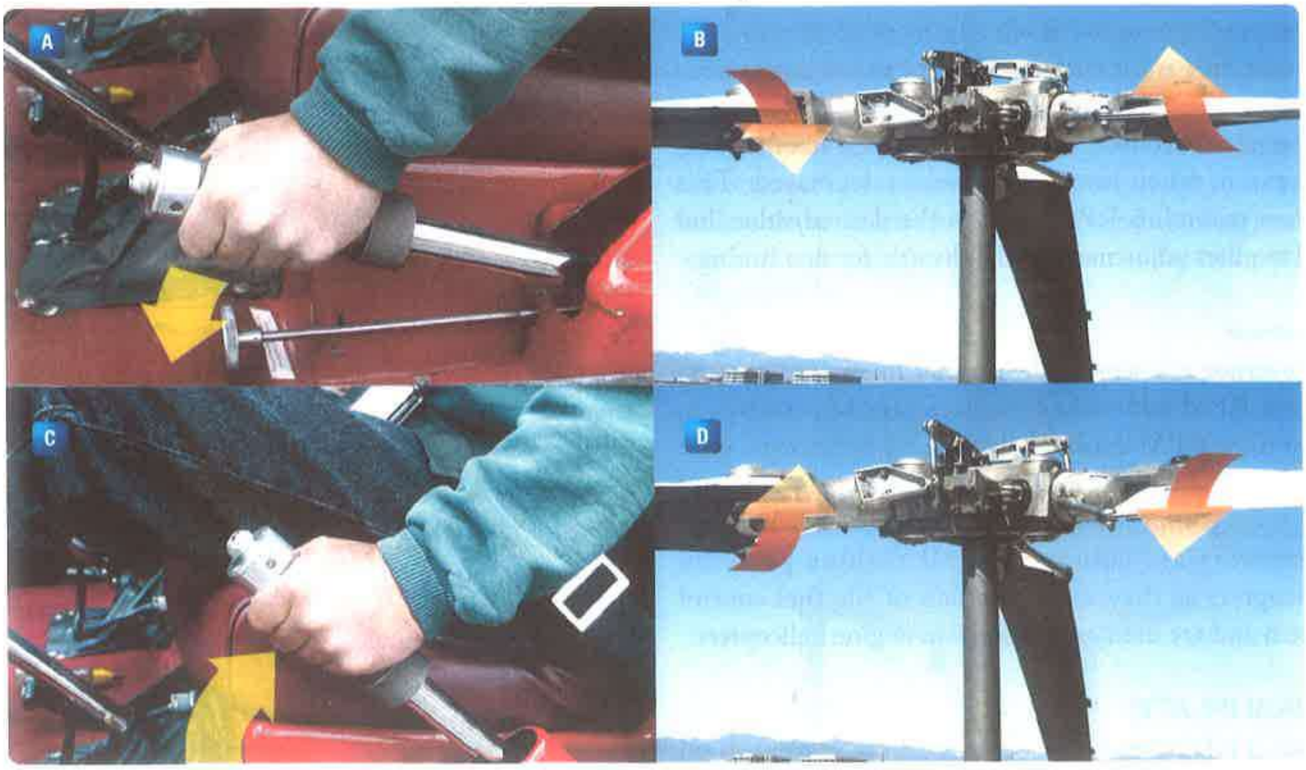


Figure 2-3. Movement of collective pitch control-throttle and effect on rotor blades.

deflection of the rotor blade in front of the helicopter and a maximum upward deflection behind it, causing the rotor disk to tilt forward.

COLLECTIVE CONTROL

The collective pitch control (or simply "collective" or "thrust lever") is located on the left side of the pilot's seat and is operated with the left hand. The collective is used to make changes to the pitch angle of the main rotor blades and acts simultaneously, or "collectively". Through a series of mechanical linkages, as the collective lever is raised, there is a simultaneous and equal increase in pitch angle of all main rotor blades. As it is lowered, there is an equal decrease in pitch angle. The amount of movement in the collective lever determines the amount of blade pitch change. (***Figure 2-3***) An adjustable friction control helps prevent inadvertent collective lever movement.

With a change in pitch angle comes a change in the load on each blade, which affects the speed or Revolutions Per Minute (RPM) of the main rotor. As the pitch angle increases, the blade takes a bigger bite of air that increases the load on the blade, and so the rotor RPM decreases. Decreasing pitch angle decreases the load on the blade, and RPM increases. To maintain a constant rotor RPM, which is essential in helicopter operations, a proportionate change in power is required to compensate for the change in load on the blades. This is accomplished with the throttle control or governor, which automatically adjusts engine power according to the position of the collective.

In ***Figure 2-3A*** the collective pitch control is lowered, which causes the blade angle and the amount of lift created to decrease along with applying a proportional decrease in engine power. (***Figure 2-3B***). In ***Figure 2-3C*** the collective is raised to increase the lift created by the blades ***Figure 2-3D***, and the governor or throttle increases the engine power to maintain the rotor RPM.

Throttle Control

The function of the throttle and the governor is to regulate engine RPM by increasing or decreasing fuel flow towards the engine when the collective is raised or lowered. If the governor is an automatic system, to maintain RPM the throttle must be moved manually with the twist grip. The throttle control is much like a motorcycle throttle and works the same way. Twisting the throttle to the left increases RPM. Twisting to the right decreases RPM. As with any aircraft control, large adjustments of either collective pitch or throttle should be avoided. All corrections should be made using smooth pressure.

Correlator

A correlator is an automated mechanical connection between the collective lever and the engine throttle. When the collective lever is raised, power is automatically increased; when lowered, power is decreased. This system maintains RPM close to the desired value, but still requires adjustment of the throttle for fine tuning.

Governor

A governor is a sensing device that monitors rotor and engine RPM and makes the necessary adjustments to keep rotor RPM constant. In normal operations, once the rotor RPM is set, the governor keeps it constant and there is no need to make any throttle adjustments. Governors are common on all turbine powered helicopters as they are a function of the fuel control system and are used on some piston engine helicopters.

SWASHPLATE

When a pilot moves the collective or cyclic controls on the flight deck, a signal is sent to the main rotor blades to cause them to change their angles. From the flight deck, the signal can originate as the movement of cables and pulleys, push/pull tubes, or even an electronic signal. The command for a change in blade or rotor angle must start from the non rotating fuselage and find its way to the main rotor blades which are rotating. This is a similar challenge faced by an airplane with rotating propellers that are capable of changing propeller blade angle, only more complicated because the helicopter rotor must be able to change multiple angles at once. The device which permits these changes is called a swashplate. (***Figure 2-4***)

The swashplate is made of two circular plates, one on the bottom that is attached to the frame of the helicopter and does not rotate, and one on the top that is attached to the rotating blade assembly. The plates are separated by a set of bearings allowing the top plate to rotate over the bottom plate while physically being in contact with it, so forces can be transferred. ***Figure 2-5*** shows a simplified swash plate as part of a main rotor system.

A swashplate assembly consists of the parts shown in ***Figure 2-6***. The lower swashplate is locked in rotation by the fixed scissor. This scissor is connected between the top of the main gearbox and the lower swashplate. This plate is actuated by the control rods of the cyclic and collective linkages has two possibilities of movement.

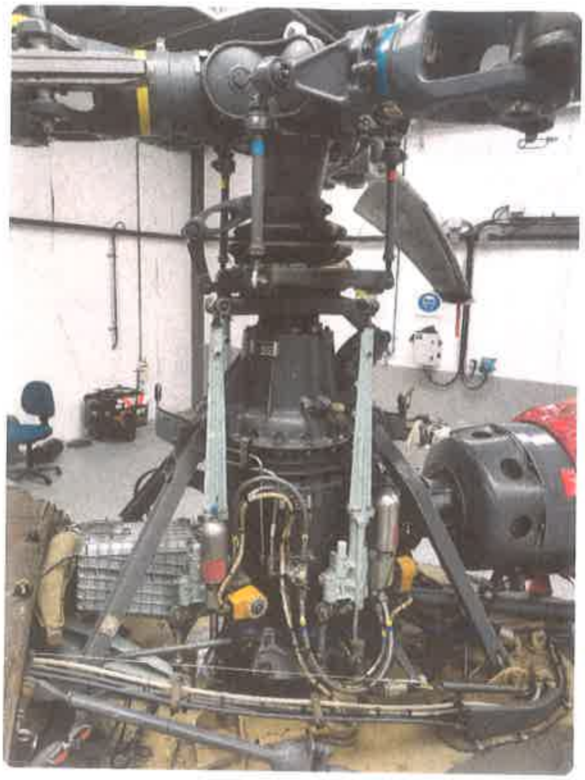


Figure 2-4. Light helicopter swashplate.

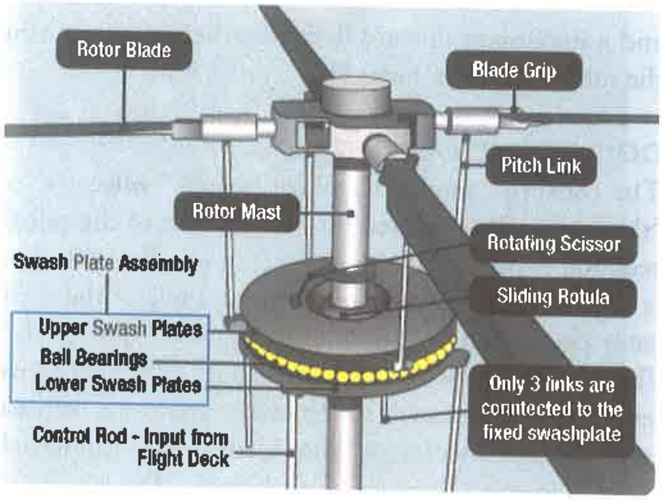


Figure 2-5. Simplified figure of a helicopter swashplate.

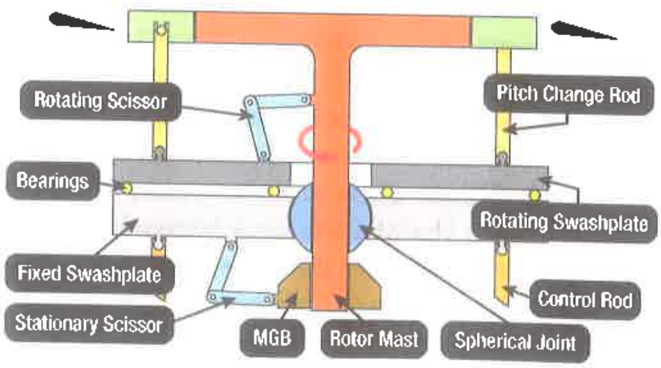


Figure 2-6. Swashplate component.

As each blade rotates through the tip-path plane, the variation of the blade pitch is accomplished through the tilting of a fixed swashplate and its mechanical linkages (pitch angle rods).

The Cyclic Movement And The Swashplate

When the pilot moves the cyclic pitch control, this command sends the lower swashplate around a ball joint and tilts the upper swashplate which remains in parallel with the rotor due to its two bearings. The rotary swashplate now spins in an inclined circle. This means that the pitch change rods are constantly undergoing up and down movements. As they are attached to the rotor they will constantly change the blade pitches between high to low values with each revolution, thus creating high lift at one point on the rotating disc and low at 180° from that point. (***Figure 2-7***)

In other words, the swashplate will tilt the main rotor in the requested direction and therefore the helicopter will also move in this desired direction. To define the plane represented by the swashplate, it is necessary to connect three points which define the different tilting axis. (***Figure 2-8***) These three points are coming from the cyclic stick which give only two pure movements; the longitudinal and the lateral. (***Figure 2-9***)

Longitudinal And Lateral Movement

When the cyclic is moved on a light helicopter, one rod moves the swashplate in longitudinal motion and two rods in lateral movement. (***Figure 2-10***) After changing direction with a bellcrank, the control movement tilts the swashplate around the lateral axis (***Figure 2-11***) to change the angle of attack of the blades, to move the helicopter forward as explained previously. The same is true if the pilot pulls the cyclic stick to back up. The rod will tilt the swashplate in the opposite direction.

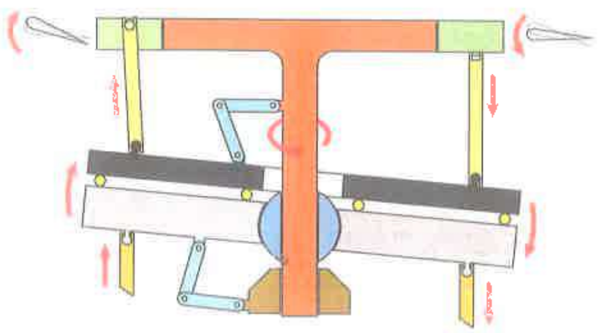


Figure 2-7. Cyclic movement.

To move the helicopter to the left or to the right, lateral rods pass through a transformer system which provides the lateral movement. (***Figure 2-12***) These two outer lines are moving from the same place but in opposite direction. These movements permit, the rods and bellcranks to tilt the swashplate around the longitudinal axis and create the lateral movement of the helicopter. (***Figure 2-13***)

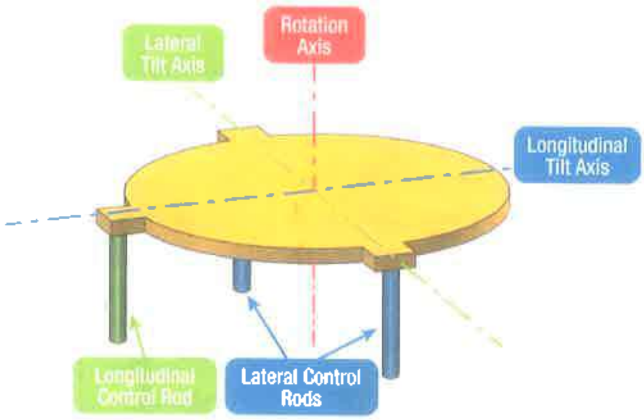


Figure 2-8. Light helicopter axis description.

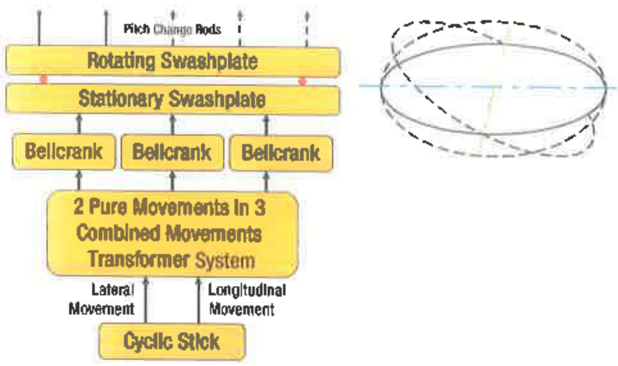


Figure 2-9. Cyclic synoptic.

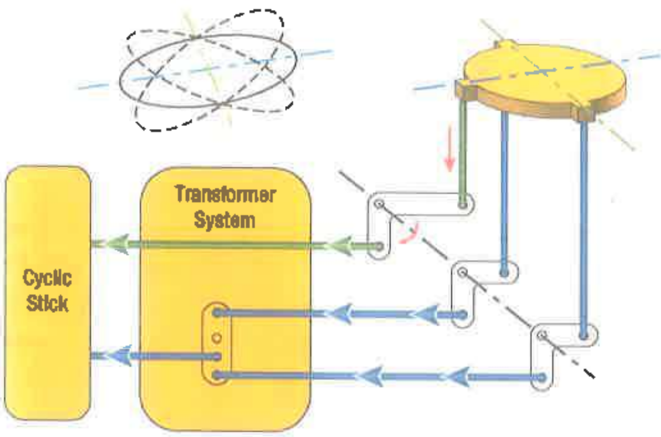


Figure 2-10. Cyclic cinematic.

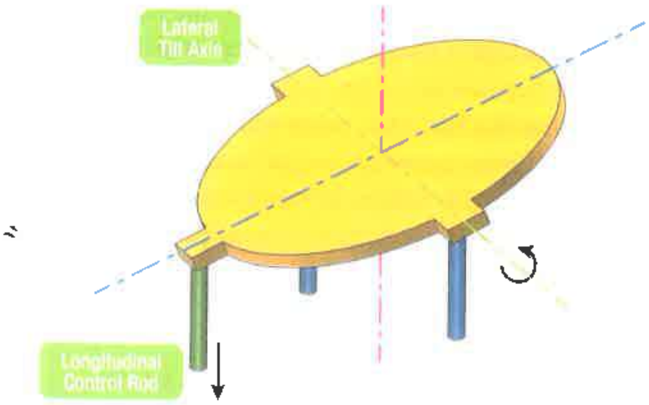


Figure 2-11. Longitudinal swashplate movement.

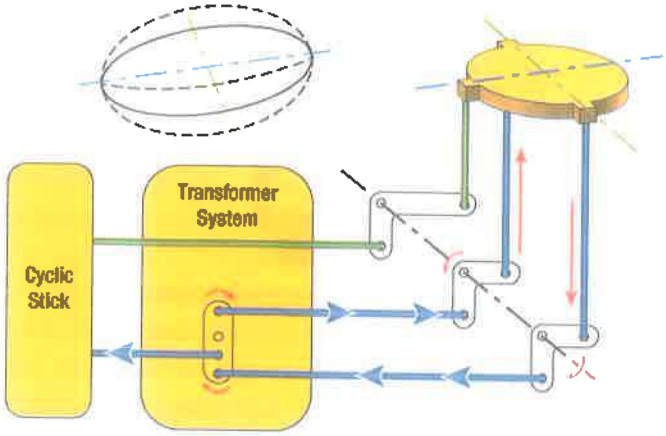


Figure 2-12. Cyclic provided lateral movement to the swashplate.

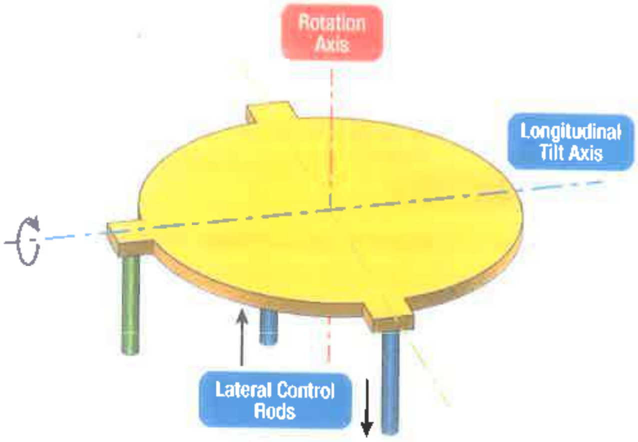


Figure 2-13. Lateral swashplate movement.

To move laterally and longitudinally the cyclic is moved in the diagonal direction and the consequence will be the automatic addition of the two movements. If the control is in neutral, the helicopter will fly and remain in a hover.

The Collective Movement And The Swashplate

The lower swashplate is connected to a ball joint that slides up and down along the rotor mast. When the pilot pulls the collective control, a command is sent to the lower swashplate by three connecting rods as shown in ***Figure 2-5*** as "input from flight deck".

These three rods push the lower swash plate in parallel upwards, which also pushes the upper swashplate and ball joint as one unit. The upper swash plate, through these links, applies force to the rotor blades and so changing the angle of attack of all the blades in the same direction regardless of the inclination of the swashplate. (***Figure 2-14***)

To create this parallel movement, the three rods described above must move in the same direction. To perform this movement, a lever called the mixer, receives the collective pitch order, and moves the three bellcrank's rotation axis in the same direction and for the same value resulting in the variation of lift. (***Figure 2-15 and Figure 2-16***)

This modification in turn, tilts the three cyclic bellcranks which move the control rods of the swashplate. This collective action (same movement, in the same direction,

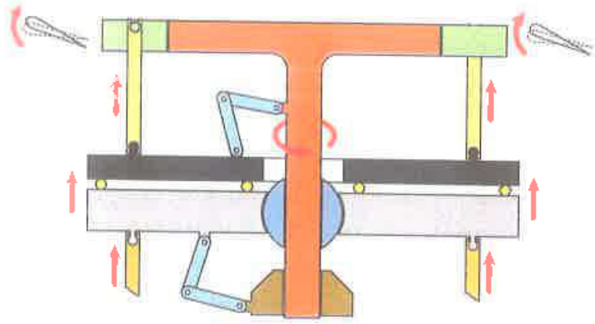


Figure 2-14. Collective movement.

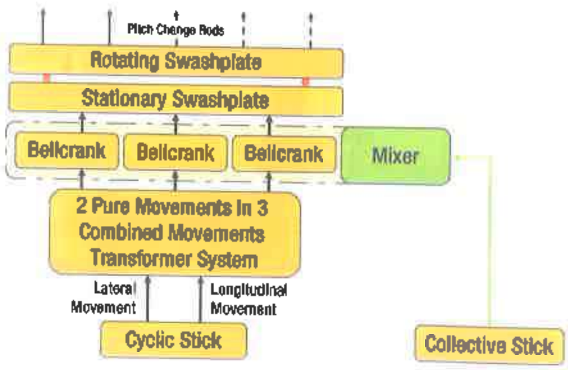


Figure 2-15. Collective synoptic.

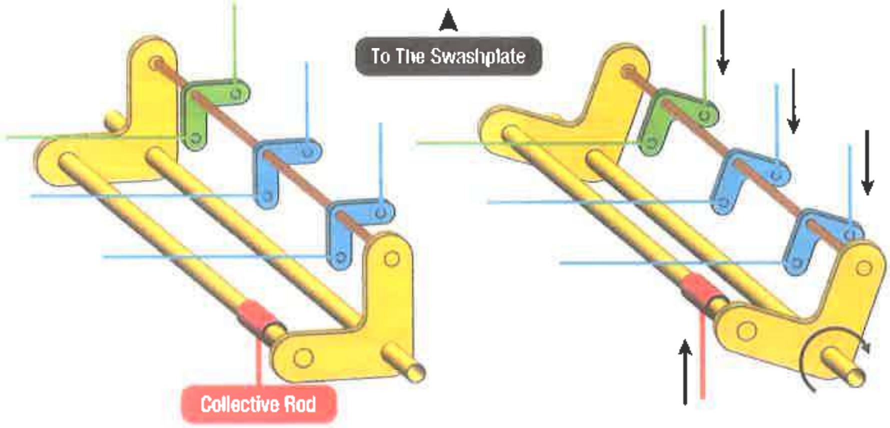


Figure 2-16. Mixer.

at the same time) causes the swashplate to move parallel to its position, modifying the angle of attack of all the blades for the same value and reducing, in this case, the total lift.

It is possible to operate the cyclic stick and the collective stick simultaneously. In that case all the bellcranks move, transmitting a specific movement to the swashplate which acts to control the angle of attack of each blade as the pilot wishes.

YAW CONTROL: ANTI-TORQUE CONTROL, TAIL ROTOR, BLEED AIR

ANTI-TORQUE CONTROL, TAIL ROTOR

Yaw for an aircraft (helicopter or airplane) is a movement that takes place around the vertical axis. The three axes of rotation for a helicopter are shown in ***Figure 2-17***. For a helicopter with a single main rotor yaw control is accomplished by the tail rotor (anti-torque rotor).

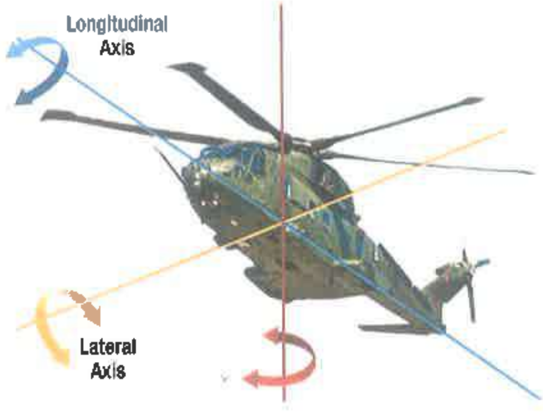


Figure 2-17. Three axes of rotation for a helicopter.

One method to counteract torque is to place a small propeller at the end of the tail boom. Its purpose is to create thrust that acts in the opposite direction of the helicopter's tendency to rotate. The tail rotor force, in newtons, multiplied by the distance from the tail rotor to the main rotor, in meters, creates a torque in newton meters that counteracts the main rotor torque. This concept is shown in ***Figure 1-17*** of *Sub-Module 01*. The main difference is that there is no cyclic control as it is not necessary to modify the direction of the lift but simply to increase or decrease thrust as a collective command. (***Figure 2-18***)

Figure 2-18. Tail rotor description.

Figure 2-19. Movement of right pedal.

The tail rotor consists of a rotor head which supports the two hinges necessary to control the thrust. The collective feathering hinge and the flapping hinge.

The Tail Collective Hinge

The tail collective hinge system is the same, regardless of the number of blades on the tail. To modify the thrust of the tail rotor, it is necessary to be able to change the angle of attack of all the blades at the same time. To create the collective movement on the tail blades, a servo control acts on a platter called a spider. When the spider is in motion, the pitch change rods connected between it and the blades force the latter to move around its feathering hinge.

When the main rotor is turning counterclockwise (viewed from the top), its torque will try to rotate the fuselage clockwise. This makes the nose of the helicopter yaw to the right. When the anti-torque rotor is producing the correct amount of thrust to exactly counteract the torque of the main rotor, the nose of the helicopter will remain steady in the same direction. Making the helicopter yaw to the right is accomplished by pushing on the right anti-torque pedal. (***Figure 2-19***) This action reduces the thrust of the tail rotor (by reducing the blade angle) to less than what is needed to counteract the torque and the helicopter yaws to the right. When the pilot pushes the left anti-torque pedal, increases the blade angle of the tail rotor, the nose of the helicopter will yaw to the left. It is important to understand that the tail rotor does not control the direction in which the helicopter is flying. It only controls the direction in which the fuselage is heading. The direction of flight is controlled by the cyclic pitch system, as previously described.

The Tail Flapping Hinge

When a helicopter is in motion, the translation speed is added or subtracted from the rotation speed of the blades depending on whether it is an advancing or a retreating blade. This problem is the same on the tail rotor. To solve it, a flapping hinge is added. On the tail rotor, the blades are lighter than the main rotor and the movement is easy to control. When the blade rotates in the advancing zone, the pitch is automatically changed by the change rod which reduces the angle of attack. With this reduction in lift, the reaction of the blade is to return to the average flying plan. (***Figure 2-20***)

When the blade retreats, it tends to move closer to the tail structure. To avoid contact, the pitch change rod increases the pitch. The reaction of the blade with the increased pitch is to move away from the structure. (***Figure 2-21***)

Figure 2-20. Advancing zone flapping movement.

The Tail and Pedal Connection

Because the tail rotor is too far from the pedals to connect with connecting rods, the usual solution is to route cables through the tail structure. (***Figure 2-22***) At the end of the tail beam, the movement of the cables controls the servo-control of the tail. When the servo-control extends or retracts, it changes the position of the spider and so changes the blade's angle of attack thus adjusting thrust. (***Figure 2-23***) This cable and pulley system has the additional advantage of being easy to check and repair in the event of damage. Another advantage is the lightness and flexibility of the system required for this task.

Different Solutions

During translational flight, the tail rotor uses 20% of the power supplied by the motor. The remaining 80% is given to the main rotor. Different solutions are used to reduce the power consumption of the tail and restore maximum torque to the main rotor.

Figure 2-21. Retreating zone flapping movement.

Figure 2-23. Cables-rear sector-rod and tail servo-control.

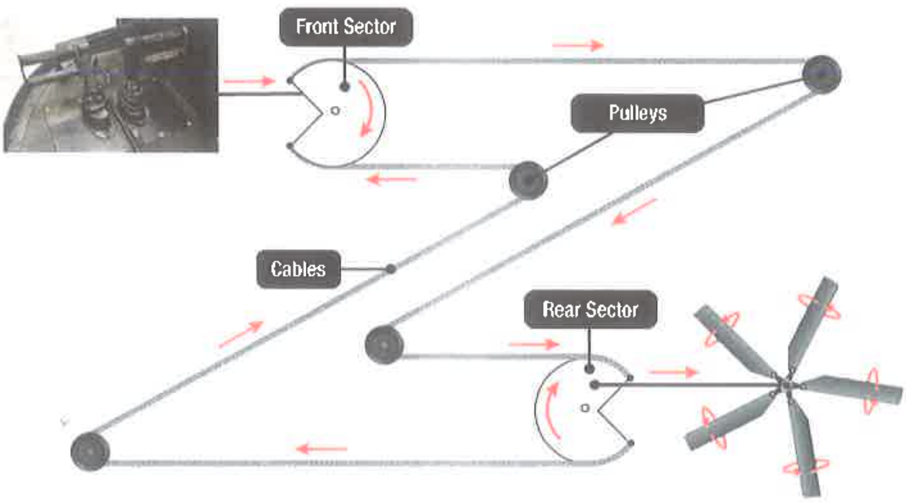


Figure 2-22. Yaw control system.

On a helicopter with two main rotors, known as a tandem configuration, the main rotors are located at each end of the helicopter and there is no tail rotor. The two main rotors spin in opposite directions, with each one counteracting the torque of the other. The Boeing CH-47 Chinook helicopter has this type of main rotor system. (***Figure 2-24***)

Changes in yaw on this helicopter are done with differential cyclic pitch, with the front rotor generating cyclic pitch in the direction the pilot wants the nose to move and the aft rotor generating the opposite cyclic pitch. On helicopters with two main rotors, known as a coaxial configuration, the main rotors are located one above the other and driven by concentric drive shafts. The two main rotors spin in opposite directions, with each one counteracting the torque of the other. The Russian KA-32 helicopter in ***Figure 2-25*** has this type of main rotor system. Changes in yaw on this helicopter are made by increasing the collective pitch of one of the main rotors and decreasing the collective pitch of the other, creating a dissymmetry of torque.

Figure 2-24. Tandem main rotor helicopter.

Figure 2-25. Helicopter with coaxial counter-rotating main rotors.

On helicopters with two main rotors that are side by side with separate shafts, such as the Kaman helicopter shown in *Sub-Module 01* (***Figure 1-7***), opposite cyclic pitch on the two rotors is used to produce yaw movement.

The Fenestron Solution

While the tail rotor is a highly effective solution, there are real dangers in its use. First, the spinning blades are deadly if someone steps into them. Second is the risk of hitting a tree or other object. If the tail rotor is lost, the helicopter becomes uncontrollable and a crash is inevitable. Aerodynamically, this is also a problem. When the helicopter is in forward flight and a vertical pylon is used to support the tail rotor head, (***Figure 2-26***) drag appears.

An alternative solution to the tail rotor, seen in ***Figure 2-26***, is a type of anti-torque rotor known as a Fenestron, or fan-in-tail design as seen in ***Figure 2-27***. Because the rotating blades in this design are enclosed in a shroud, they present less of a hazard to personnel on ground and they create less drag in flight. Another advantage is to simplify the tail rotor assembly, as inside the shroud, the problem of advancing and retreating blade disappears. This direct benefit allows the removal of the flapping hinge.

Another advantage of the Fenestron is weight. The blades are smaller and lighter than the classic blades, and in case of change the cost is cheaper. As stated before, the tail rotor robs the main rotor of a part of its power. With a Fenestron, the special shape of the tail pylon reduces the tail rotor's power consumption. This occurs during forward flight, as the dissymmetric vertical fin

Figure 2-26. Aerospatiale Helicopter Tail Rotor.

(***Figure 2-28***) creates its own lift to oppose the torque, similar to a wing but in vertical position. However, one negative aspect of a Fenestron is the whistling sound in the cabin due to the airflow into the shroud.

Figure 2-27. Fenestron on a Eurocopter Model SA341 Gazelle.

Figure 2-28. Fenestron dissymmetric vertical fin.

The Bleed Air NOTAR System

Another method of counteracting the torque of the main rotor is a technique called the no-tail-rotor system, or NOTAR. (***Figure 2-29***)

This system uses a high volume of air at low pressure, which is generated by a fan installed in the tail boom and driven by the helicopter's engine. The fan forces air into the boom, where a portion exits from slots on the right side of the boom and together with the main rotor downwash, creates a phenomenon called the Coanda effect. The air coming from the slots causes a higher velocity, and therefore lower pressure on that side of the boom. The higher pressure resulting on the left side of the boom partially counteracts the torque of the main rotor. (***Figure 2-30***)

The remainder of the air travels back to a controllable rotating nozzle in the helicopter tail. The air exits the nozzle at a high velocity and creates thrust which additionally counteracts the torque of the main rotor. In a hover this force on the left side of the boom counteracts up to 60% of the main rotor torque, with the remainder being controlled by thrust from the nozzle.

The NOTAR anti-torque system eliminates some of the mechanical problems of a tail rotor, including long driveshafts, suspension bearings, mid-gearboxes, and 90° gearboxes. It is safer for the mechanic due to the absence of rotating tail blades, and in flight, the reduction of drag allows for less stress on the structure and reduced fuel consumption. A final advantage, unlike the Fenestron, is that the NOTAR system is quieter.

Figure 2-29. MD-520 helicopter with no tail rotor.

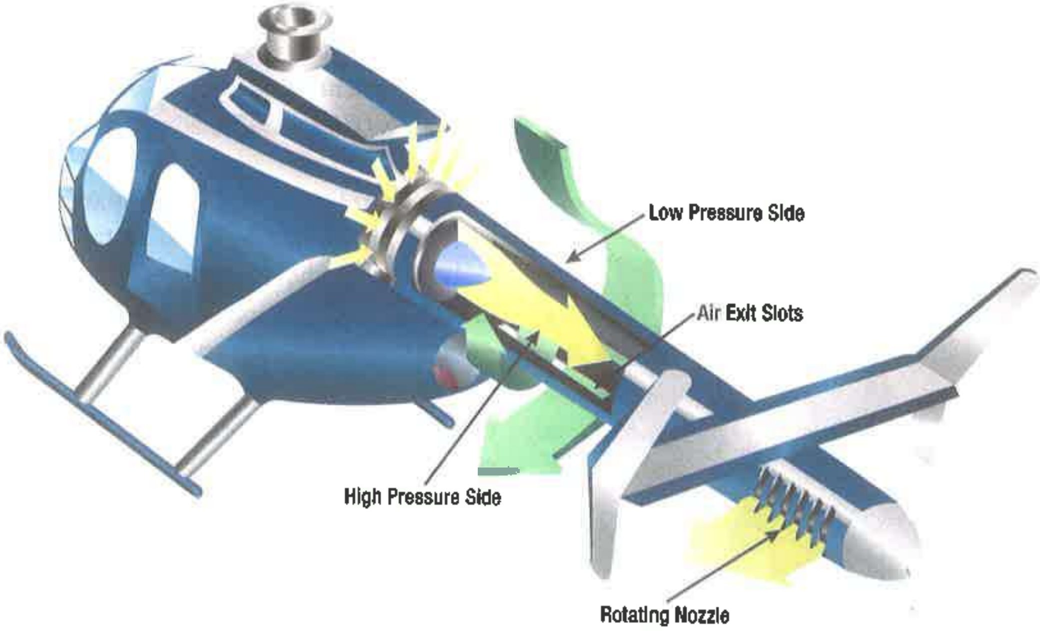


Figure 2-30. Airflow from a variable pitch fan on a NOTAR helicopter.

MAIN ROTOR HEAD: DESIGN AND OPERATION FEATURES

The rotor system on a helicopter consists of a mast, hub, and rotor blades. The mast is a hollow cylindrical metal shaft which extends upwards from and is driven, (and sometimes supported) by the transmission. At the top of the mast is the attachment point for the rotor blades called the hub. The rotor blades are then attached to the hub by a variety of different methods. The blades need three free movements to withstand the various stresses and to control the flight direction. (***Figure 2-31***)

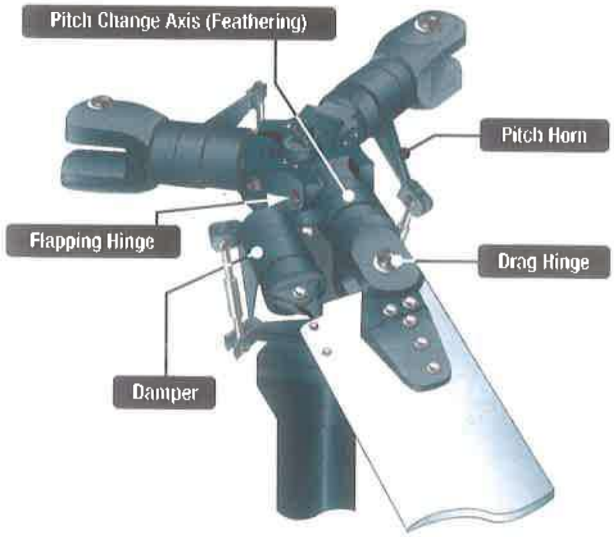


Figure 2-31. Flapping hinge allows up and down blade movement.

The flapping hinge prevents the helicopter from tilting due to the dissymmetry of lift in the advancing and retreating zone of translational flight. The drag hinge withstands the lead/lag movement due to the Coriolis effect. (***Figure 2-32***) The feathering hinge permits changes to the angle of attack to change the direction of the lift.

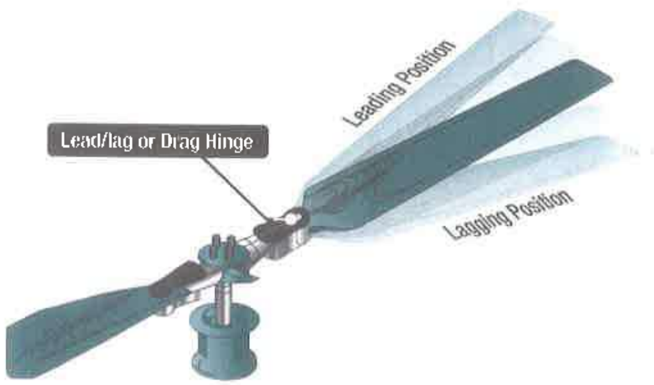


Figure 2-32. Lead-lag hinge allows fore and aft blade movement.

Main rotor systems are classified according to how the main rotor blades are attached and move relative to the main rotor hub. There are three basic classifications: fully articulated, semi-rigid and rigid. Some modern rotor systems, such as the bearingless rotor system, use a combination of these types.

Fully Articulated Rotor Systems

Fully articulated systems allow each blade to lead/lag (move back and forth in a plane), to flap (move up and down about an inboard hinge) independently of the other blades, and to feather, or rotate about the pitch axis to change lift. (The term feather should not be confused with the concept of feather as it applies to constant speed propellers.) Each of these blade motions are related to the others. The three hinges are mechanical assemblies with bearings and lubrication. In ***Figure 2-33***, the drag hinge and the flapping hinge are lubricated with oil by gravity, and a pipe is connected to the feathering hinge to lubricate it.

As the rotor spins, each blade responds to inputs from the collective and cyclic controls. The center of lift on the whole system moves in response to these inputs to effect pitch, roll, and upward motion.

As the lift on a given blade increases, it tends to flap upwards. The flapping hinge for the blade permits this motion and is balanced by the centrifugal force acting on the blade which tries to keep it in the horizontal plane. Under the flapping hinge, ***Figure 2-34***, a mechanical stop is installed to avoid letting the blade rise too high and have uncorrectable loss of lift. At the same time, a lower stop protects the helicopter structure from impact in case the blade's flight would become too low.

Either way, some motion must be allowed. The centrifugal force is nominally constant; however, the flapping force is affected by the severity of in flight maneuvering (rate of climb, forward speed, aircraft gross weight, G forces). As the blade flaps, its center of gravity changes. This changes the local inertia of the blade with respect to the rotor system. It speeds up or slows down with respect to the rest of the blades and with the whole rotor system. This is accommodated by the lead/lag or a drag hinge, shown in ***Figure 2-32***. The blade trying to speed up or slow down can best be understood by thinking about an ice skater performing a spin. As the skater moves their arms in, they spin faster because the inertia changes, but the total energy remains constant. Conversely, as they extend their arms the spin slows down. This is also known as conservation of angular momentum. An in-plane damper (***Figure 2-35***) controls how quickly the lead/lag motion can occur.

Older hinge designs relied on conventional metal bearings. By basic geometry, this precludes a single hinge from being able to handle both flapping and lead/lag movement and is cause for recurring maintenance. Newer rotor systems use elastomeric bearings, which are arrangements of rubber and steel that can permit motion in two axes. This allows two metal bearings to be replaced by a single bearing. The new hinge design requires less maintenance, is easier to inspect, and its wear is gradual and more visible. The metal-to-metal contact of older bearings and the need for lubrication is thus eliminated.

Figure 2-33. Fully Articulated Main Rotor System.

Figure 2-34. Flap stops.

Figure 2-35. Drag hydraulic damper on SA330 puma.

Semi-Rigid Rotor Systems

A semi-rigid rotor system is usually composed of two blades that are rigidly mounted to the main rotor hub. The main rotor hub is free to tilt with respect to the main rotor shaft on what is known as a teetering hinge. This allows the blades to flap together as a unit. As one blade flaps up, the other flaps down. Since there is no vertical drag hinge, lead/lag forces are absorbed and mitigated by the blade bending. The semi-rigid rotor is also capable of feathering, which is made possible by a feathering hinge. (***Figure 2-36***)

The underslung rotor system mitigates the lead/lag forces by mounting the blades slightly lower than the usual plane of rotation, so the lead and lag forces are minimized. As the blades flap upward with increased lift, the centers of pressure of the blades are almost in the same plane as the hub. Whatever stresses are remaining bend the blades for compliance.

If the semi-rigid system is an underslung rotor, the Center of Gravity (CG) is below where it is attached to the mast. This underslung mounting is designed to align the blade's center of mass with a common flapping hinge so that both blade's centers of mass vary equally in distance from the center of rotation during flapping. The rotational speed of the system tends to change, but this

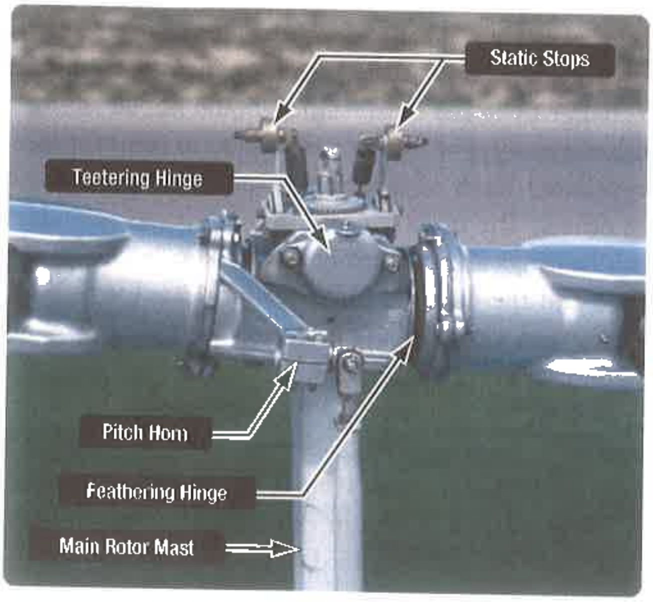


Figure 2-36. Teetering hinge for hub tilt and feathering hinge for pitch change.

is restrained by the inertia of the engine and flexibility of the drive system. Only a moderate amount of stiffening at the blade root is necessary to handle this restriction. Simply put, underslinging effectively eliminates geometric imbalance. ***Figure 2-37*** shows how the centerline of the blades falls below the teetering hinge.

Helicopters with semi-rigid rotors are vulnerable to a condition known as mast bumping which can cause the rotor flap stops to shear the mast. The static stops can be seen in ***Figure 2-36***. The mechanical design of the semi-rigid system requires some physical limit to downward flapping of the blades. Mast bumping is the result of excessive rotor flapping. Each rotor system design has a maximum flapping angle. If flapping exceeds the design value, a static stop will contact the mast. It is the violent contact between the static stop and the mast during flight that causes mast damage or separation. This contact must be avoided at all

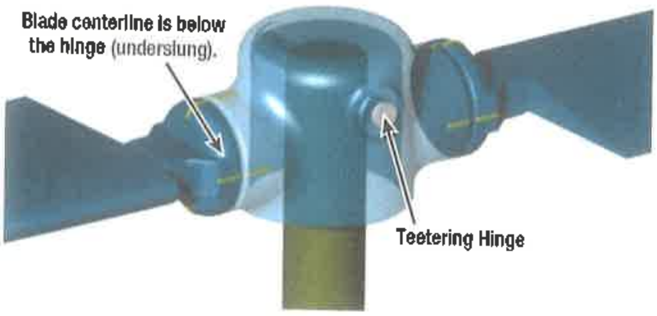


Figure 2-37. Semirigid rotor with underslung design.

costs. Mast bumping is related to how much the blade system flaps. In straight and level flight, blade flapping is minimal, at perhaps 2°. Flapping angles increase slightly with high forward speeds, at low rotor RPM, at high density altitude, at high gross weights, and when encountering turbulence. Maneuvering the aircraft in a side slip or during low speed flight at extreme CG positions can induce larger flapping angles.

Rigid Rotor Systems

The rigid rotor system shown in ***Figure 2-38*** is mechanically simple, but structurally complex as operating loads must be absorbed in bending rather than through hinges. In this system, the blade roots are rigidly attached to the rotor hub. Rigid rotor systems tend to behave like fully articulated systems through aerodynamics but lack flapping or lead/lag hinges. Instead, the blades accommodate these motions by bending. They cannot flap or lead/lag, but can be feathered (blade pitch changes). As advances in helicopter aerodynamics and materials continue to improve, rigid rotor systems may become more common because the system is fundamentally easier to design and offers the best properties of both semi-rigid and fully articulated systems.

The rigid rotor system is very responsive and usually not susceptible to mast bumping like the semi-rigid or articulated systems because the rotor hubs are solidly mounted to the main rotor mast. This allows the rotor and fuselage to move together as one entity and eliminates much of the oscillation usually present in the other rotor systems. Other advantages of the rigid rotor include a reduction in the weight and drag of the rotor hub and a larger flapping arm, which significantly reduces control inputs. Without the complex hinges, the rigid system becomes more reliable and easier to maintain than the other configurations. A disadvantage is the quality of ride in turbulent or gusty air. Because there are no hinges to help absorb these loads, vibrations are felt in the cabin more than with other rotor head designs.

The flexible rotor head, is similar to articulated rotors, with the difference that the seals have been eliminated due to the use of flexible materials which simulate the necessary flap and lead-lag. These materials are elastomers and beams made of titanium or fiberglass.

Bearingless Rotor Systems

There are several variations of the basic three rotor head designs. The bearingless rotor system of the Eurocopter

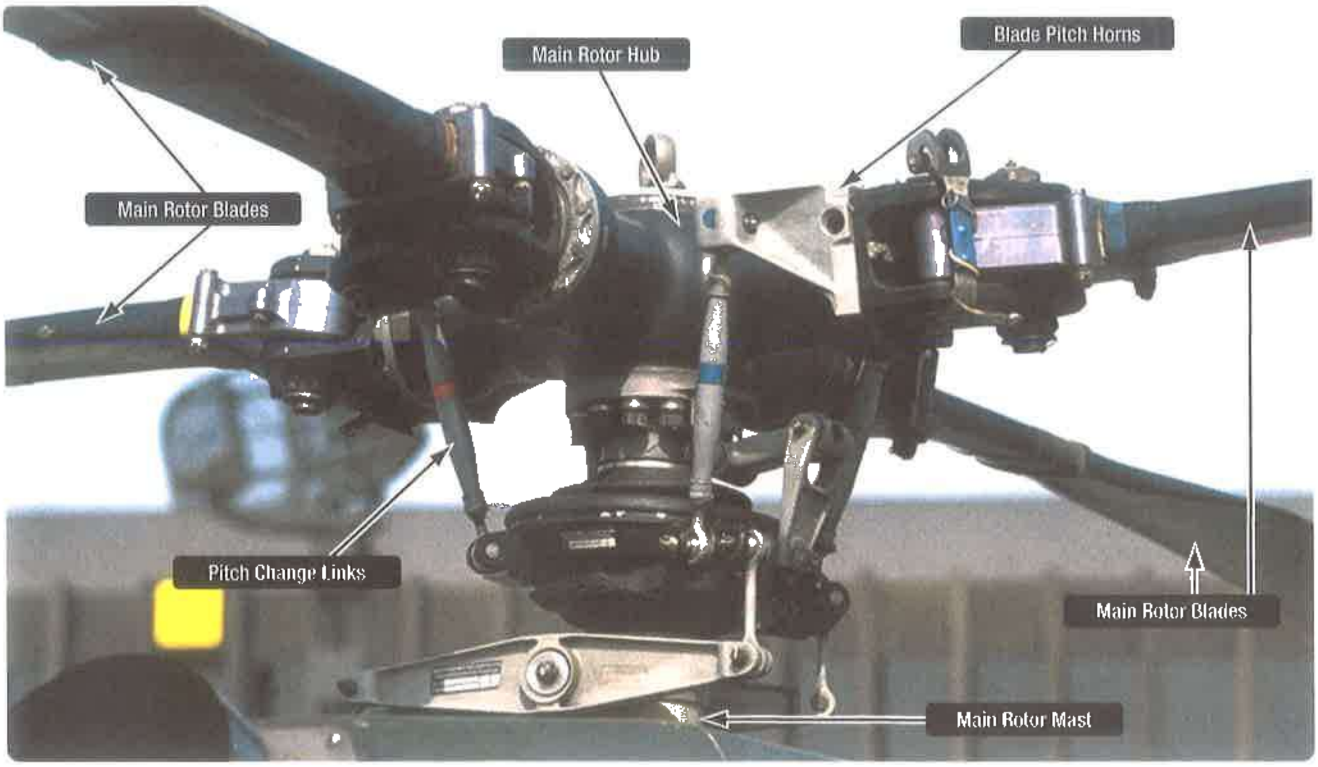


Figure 2-38. Four-blade hingeless (rigid) main rotor.

EC665 Tiger or Comanche, is closely related to the articulated rotor system, but has no bearings or hinges. This design relies on the structure of blades and hub to absorb stresses. The main difference between the rigid system and the bearingless system is that the bearingless system has no feathering bearing. Instead, material inside the cuff is twisted by the action of the pitch change arm. The material inside the blade cuff is known as elastomeric, having the characteristic of being able to deform when a force is applied and then immediately return to its original shape when the force is removed. Nearly all bearingless hubs are made of fiber-composite materials.

BLADE DAMPERS: FUNCTION AND CONSTRUCTION

On a fully articulated rotor system, the blades are hinged to allow for movement up and down (flapping) and fore and aft (lead/lag). As shown in ***Figure 2-31***, these hinges are called the flapping hinge and the drag hinge. To limit how quickly the blade can move forward (lead) or aft (lag) when it experiences a change in angular momentum, a device known as a damper is installed between the blade and the rotor hub. (***Figure 2-39***) When the blade tries to lead or lag, it experiences a resistance to this movement by an internal force provided by the damper. This concept is the same as the internal resistance provided by the shock strut on an airplane or a shock absorber on an automobile. There are three types of dampers, the hydraulic, the multiple disks and elastomeric.

Hydraulic Damper

The hydraulic damper is made up of a cylinder containing a piston, filled with hydraulic fluid that can pass from one side of the piston to the other by flowing through a specific sized orifice. On some dampers the size of the orifice is adjustable and on others it is fixed. By controlling how fast the hydraulic fluid can move from one side of the piston to the other, the damper controls how fast the blade is able to move when a lead or lag condition is called for. A hydraulic damper is shown in ***Figure 2-35***. The hydraulic reserve is in the center of the rotor head. Pipes allow the hydraulics to be transferred to the shock absorber and automatically purge it to ensure that no air enters inside to disturb the action of the shock absorber.

Figure 2-39. Fully articulated main rotor with lead-lag damper.

Multiple Disk Damper

The multiple disk damper is made up of a set of circular disks that are mounted in a cylinder filled with hydraulic fluid. The hydraulic fluid is used for cooling and lubrication. Looking at the disks in pairs, the first disk will be secured to the cylinder and cannot rotate, while the second disk has splines in the middle and can rotate on a shaft that passes through its center. The next pair of disks, and all subsequent pairs, are set up the same way. The shaft that is splined to all the rotating disks is attached to the blade, and when the blade tries to move forward (lead) or aft (lag), it causes the rotating disks to create friction with the stationary disks. The entire stack of disks is spring loaded, and the tension on the spring determines how much resistance must be overcome for the blade to move.

Elastomeric Damper

The elastomeric damper consists of an aluminum cylinder filled with a specialized rubber (elastomeric) type substance. A clevis-type bolt-and-force transfer-plate is embedded in the elastomeric material, with the cylinder attached to the blade and the clevis attached to the rotor hub. (***Figure 2-40***)

Figure 2-40. Elastomeric damper on SA341 Gazelle.

When a blade tries to lead or lag, it applies a force to the elastomeric material that causes it to deform and limit how quickly the movement of the blade can occur. When the force is removed the material returns to its original shape.

This type of shock absorber is used with a flexible blade. This is because the elastomeric shock absorber is not capable of absorbing large movements but is installed to absorb vibrations.

ROTOR BLADES: MAIN AND TAIL ROTOR BLADE CONSTRUCTION AND ATTACHMENT

The profile of a blade can be symmetrical or asymmetrical. In the case of a symmetrical profile, ***Figure 2-41***, the advantage is a low manufacturing cost but low efficiency. To produce the same lift, the asymmetric profile requires a smaller angle of attack, but this has the disadvantage of misaligning the center of gravity and the center of pressure when the angle of attack increases, creating instability and vibrations.

Choosing the shape of a blade depends greatly on the material used to build it. The greater the possibility of using composite, the more complex the shape may become to increase flight capabilities and reduce turbulence. (***Figure 2-42***)

Earlier, a rectangular shape was used for both wooden and metal blades. Nowadays, with the possibility of opposing the stresses and absorbing the deformation, carbon material allows an extremely specific and complex form, becoming very efficient. At the same time, different tip shapes have been developed to limit the vortex effect at the end of the blade by reducing the turbulence from the previous blade in the plane of rotation thus increasing flight performance.

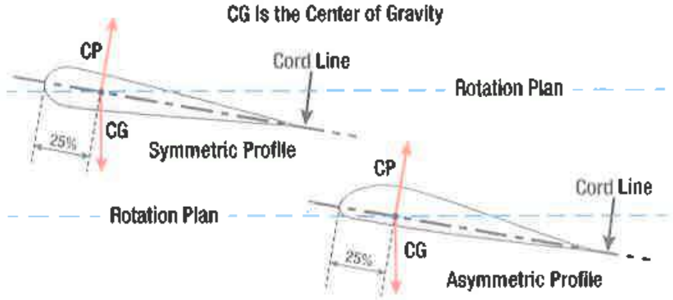


Figure 2-41. Blade profile.

Figure 2-42. Blade form evolution.

Main And Tail Rotor Blade Construction And Attachment

The blades on a helicopter are long and narrow airfoils, referred to as having a high aspect ratio. This design helps minimize the drag that can result from vortices forming at the tip. The blades can be made of wood, steel, aluminum, titanium, or composite materials. Many helicopter blades are symmetrical airfoils, meaning they have the same amount of camber on the top and the bottom, but some helicopters use asymmetrical airfoils. Symmetrical blades are very stable, which helps keep blade twisting and flight control loads to a minimum. This stability is achieved by keeping the center of pressure virtually unchanged as the angle of attack changes. Center of pressure is the imaginary point on the chord line where the resultant aerodynamic forces concentrate. Some airfoils are asymmetrical in design, meaning the upper and lower surface do not have the same camber.

Normally these airfoils would not be as stable, but this can be corrected by bending the trailing edge to produce the same characteristics as symmetrical airfoils. This is called reflexing. Using this type of rotor blade allows the rotor system to operate at higher forward speeds. One of the reasons an asymmetrical rotor blade is not as stable, is that the center of pressure changes with changes in angle of attack. When the center of pressure lifting force is behind the pivot point on a rotor blade, it tends to cause the rotor disc to pitch up. As the angle of attack increases, the center of pressure moves forward. If it moves ahead of the pivot point, the pitch of the rotor disc decreases. Since the angle of attack of the rotor blades is constantly changing during each cycle of rotation, the blades tend to flap, feather, lead, and lag to a greater degree.

Wooden Rotor Blades

Early rotor blades were made of laminated wood and fabric. Various combinations of birch, spruce, pine, and balsa wood were used to get the proper strength and the necessary shape to serve as an efficient rotating wing. A steel core is often placed near the blade leading edge to provide the proper balance and center of mass for the blade. Most of the leading edge of the blade is covered with a stainless-steel cap to protect it from foreign object damage. Metal plates are attached at the root-end of the blade to provide a location for the blade to be attached to the rotating hub.

Most wooden blades have a trim tab attached to the outboard trailing edge that is used to adjust the track of the blade when it is rotating. At the tip of the blade there is a tip pocket where weights can be added for balance purposes. The exterior of the blade is covered with a resin-impregnated fiberglass cloth. Because of the inherent differences that can exist between two pieces of wood, even though they are from the same type of tree, the manufacture of wood blades was done in matched sets that came from the same wood lot number. If one of the blades on a helicopter was damaged beyond repair limits, both blades would need to be replaced to maintain a matched set. One potential problem with the wooden blades is their ability to absorb moisture which changes the blade weight and the center of gravity.

Metal Rotor Blades

In the 1960s rotor blades started to be manufactured out of steel and aluminum, with aluminum being the most popular choice. The blades are of bonded-type construction, with a central box beam spar running the length of the blade and the metal skin wrapped around the spar and bonded to it. The inner portion of the blade is hollow, like the inside of an airplane wing. Similar to the wooden blade, a stainless-steel strip is attached to the leading edge to protect the blade from foreign object damage. Grip plates and reinforcing doublers are added to the root end of the blade to provide the strength needed for attachment to the rotor hub. A trim tab may be used on the outboard trailing edge to adjust the tracking of the blade. Like the wooden blade, the metal blade has a tip pocket where weight can be added for balance purposes.

Another design of metal blade utilizing a box beam spar, has an inner core made from a honeycomb material, instead of being hollow on the inside. The honeycomb core provides support for the blade skin and it dictates the shape of the blade. Some models of this blade had a built-in electronic inspection system that would activate a warning light if a possible crack was detected in the blade. The warning light would lead to a detailed inspection of the blade.

A third type of metal blade utilizes a full-length spar that also incorporates the leading edge of the blade. Instead of the metal spar being wrapped in a metal skin that is bonded to it, individual airfoil sections are built that get bonded to the spar to form the complete blade. This allows complete control over the different blade angles that are needed along the length of the blade. Some models of this blade incorporate a built-in system for detecting cracks. This is done by pressurizing the inside of the blade with an inert gas before sealing up the blade. A pressure sensor is then installed that monitors the pressure of the gas. If the pressure inside the blade starts decreasing because of a leak, the source of the leak could potentially be a crack, and a detailed inspection will determine this.

Helicopter main rotor blades made of metal, typically have a specific service life, identified in hours of service. When a metal blade is subject to flight stresses, causing it to flex and bend, at some point those stresses will take their toll and cause the blade to fail. If a structural problem develops with a rotor blade made of wood, it will generally let the pilot know by a noticeable change in vibration, and something can be done before a catastrophic failure occurs. With a metal blade, a structural problem can become a catastrophic failure very quickly, and the pilot may not have any advance warning. This is one of the reasons why many blades have a built-in system for detecting cracks.

Composite Rotor Blades

Composite blades are made of a variety of materials, such as fiberglass, carbon fiber, Kevlar, glass-fiber reinforced plastics and carbon-fiber reinforced plastics. Some composite blades use a metal spar with composite material being bonded to the spar to form the shape of the blade, while others use composite spars and only metal on the leading edge and on the blade attachment point. One big advantage of the composite blade over the metal blade is their service life, with no specified time in service at which they must be replaced. In addition, the

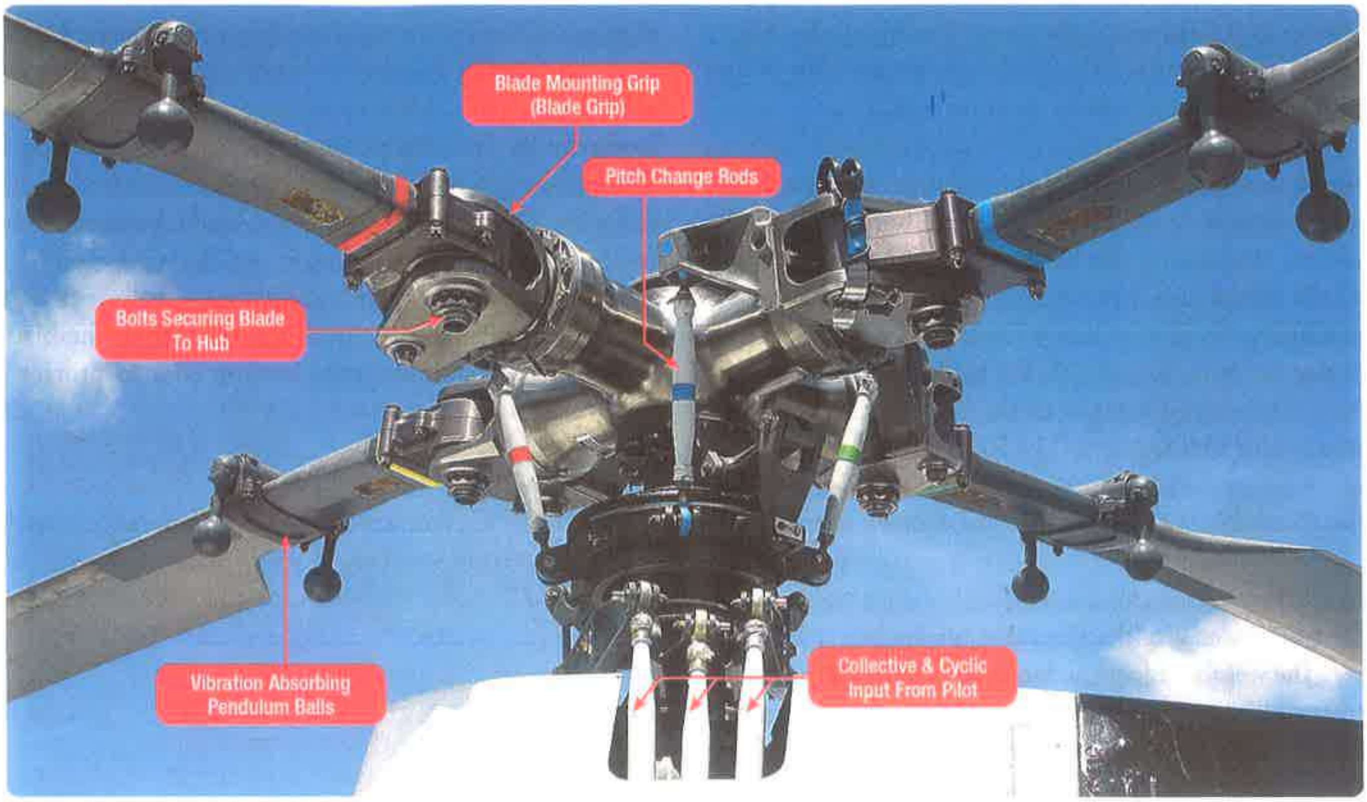


Figure 2-43. Method of blade attachment for Eurocopter Model 145.

composite blades do not have a problem with corrosion as do metal blades.

Blade Attachment

The main rotor blades are attached to the rotor hub in a variety of ways, with three of the common methods being: a blade grip (also called a blade mounting fork), a blade spindle, and a blade flange.

Figure 2-43 shows the main rotor hub and blades from a Eurocopter Model 145. This is a four blade hingeless design. The main rotor blades are made of lightweight construction glass and carbon-fiber reinforced plastics. A blade fitting assembly attached to the blade mounting fork is the connection between the main rotor blade and the main rotor head. Each blade is retained by the main blade bolt and a secondary blade bolt. The main blade bolt withstands most of the centrifugal force trying to throw the blade out of the hub, and is hollow to permit installation of dynamic balance weights. A vibration absorber, installed to the blade neck, reduces the vibrations that mainly occur during low speed and flare maneuvers. The vibration absorber consists of a set of pendulum balls attached to a shaft.

Figure 2-44 shows the grip method of blade attachment for a Bell 206. The larger bolt passing through the end of the blade grip absorbs most of the centrifugal force acting on the blade. The secondary bolt secures the blade so that it does not have free movement fore and aft (lead/lag).

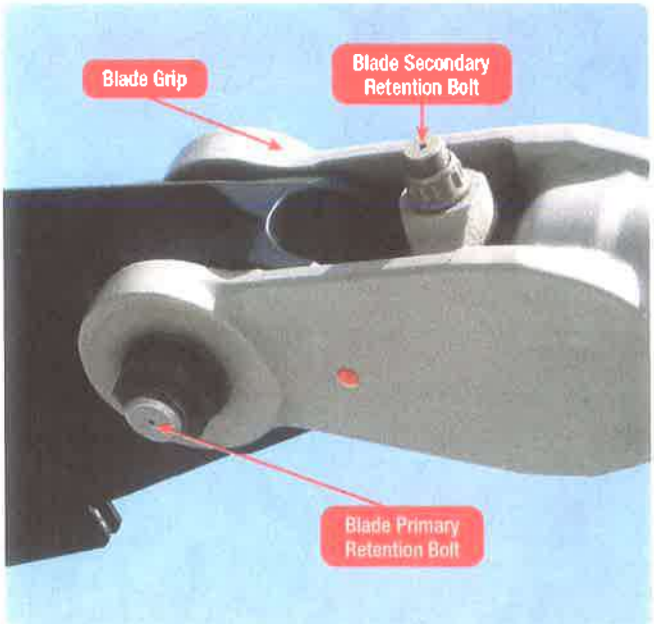


Figure 2-44. Method of blade attachment for Bell 206 helicopter.

Figure 2-45 shows a close-up photo of the blade attachment spindle. Two bolts on either side of the spindle secure the blade to the rotor hub.

Figure 2-46 shows a blade with a flange mount method of attachment on a Sikorsky SH-60. There are ten bolts in total that secure the blade to the extension coming off the rotor hub, providing an exceptionally strong attachment for a very heavy blade (113 Kg or 250 lbs). At normal operating RPM, the blade would be under a force of gravity of 450 Gs, so the blade would act like it weighs 50 850 kilograms (112 500 lbs).

Tail Rotor Blade Construction And Attachment

Blade Construction

The tail blades are like the main blades but reduced in size. The weight reduction due to the smaller size reduces stress on them. The material used for their construction has changed with technological development. Tail rotor blades are also airfoils, manufactured using either aluminum alloy or composite materials like fiberglass, carbon fiber, or fiber reinforced plastics. Some blades are made totally from aluminum alloy, some have an aluminum spar with a composite skin, and some are made totally from composite materials. The exception is the root end of the blade, where it gets attached to the tail rotor hub. The root end of the blade, because of the rotational stresses imposed on it, will be reinforced with a metal plate made from aluminum or sometimes a high strength metal like titanium or a steel alloy. The blade typically has a stainless-steel leading edge to protect it from foreign object damage.

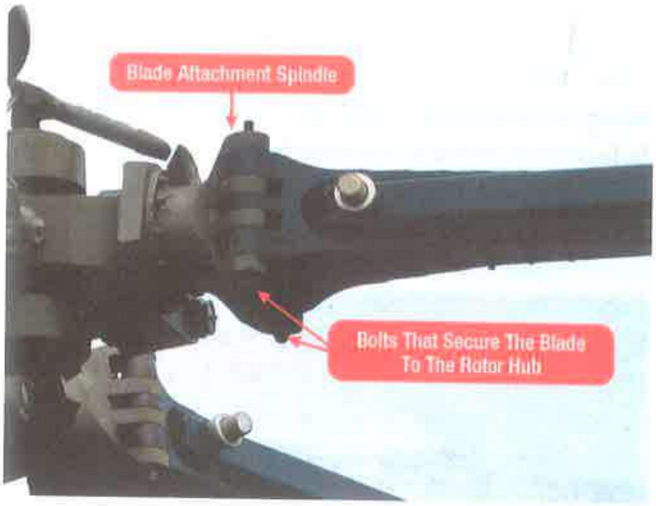


Figure 2-45. Method of blade attachment for Sikorsky CH-54 helicopter.

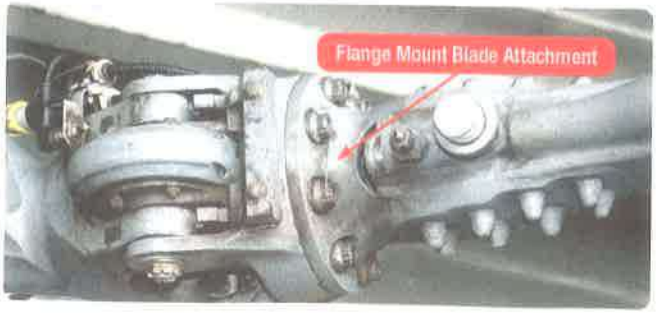


Figure 2-46. Method of blade attachment for Sikorsky SH-60 helicopter.

Blade Attachment

The blades are secured to the hub with either a blade-mounting fork (blade grip) or a flange mount. *Figure 2-47* shows the tail rotor on a Westland Sea Lynx, with two bolts securing each blade to the mounting fork. The reinforced portion at the blade root and the stainless steel leading edge are also visible in *Figure 2-47*.

A mounting fork method of attaching the tail rotor blades is also visible on the Robinson R44 shown in *Figure 2-48*. The blades are an all-metal construction, utilizing an aluminum spar with aluminum skin bonded to it. The tail rotor hub, incorporating the mounting fork for each blade, is a one piece unit. Two bolts secure each of the tail rotor blades, entering the mounting fork on one side and passing through a spherical bearing in the blade before coming out the other side to be secured with a nut. The two spherical bearings can pivot when

Figure 2-47. Tail rotor blade attachment on a Westland Sea Lynx helicopter.

a pitch change command comes from the anti-torque pedals, allowing the blades to increase or decrease their pitch (blade angle).

The two-bladed tail rotor shown in ***Figure 2-48*** has an elastomeric teetering bearing that allows the rotor to tilt in much the same manner that a two blade semi-rigid main rotor does. The elastomeric material, when subjected to a force, can change its shape, and allow movement of the tail rotor hub. When the force is removed, or changed, the material is able to return to its original shape and size. One significant advantage of the elastomeric bearing or hinge is that it does not require any periodic lubrication and is almost maintenance free.

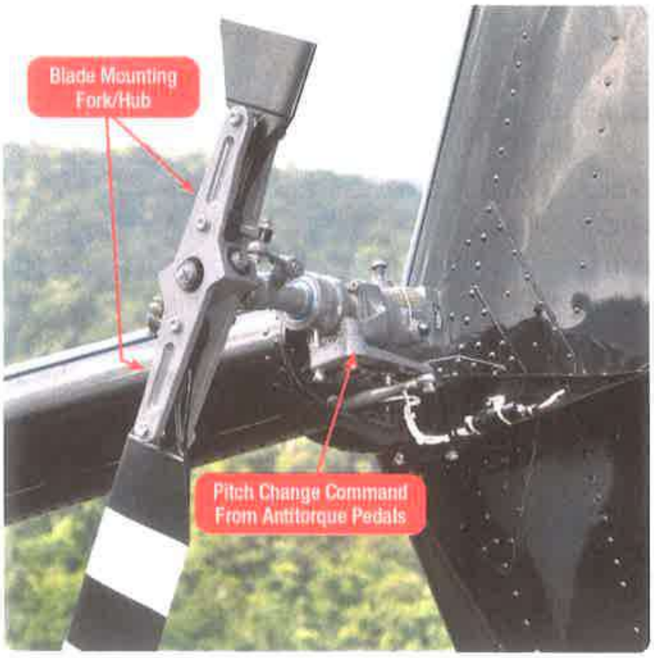


Figure 2-48. Tail rotor blade attachment on a Robinson R-44 helicopter.

Figure 2-49 shows an older tail rotor system from a MI-26 helicopter that utilizes metal bearings and hinges. In the photo grease can be seen splattered on the root end of the blades, which is typical of a system that requires lubrication. Also visible in the photo is the blade attachment fork, the reinforced blade root, the blade-flapping hinge, and the pitch change rod.

Figure 2-50 shows the tail rotor of a Sikorsky CH-53, which uses a flange style of mounting to attach the tail rotor blade to the hub. Also visible in the photo is the blade flap hinge and the pitch change rod.

Each of the tail rotor systems shown in ***Figure 2-47*** through ***Figure 2-50*** can teeter or experience blade flap. Having the ability for this type of blade movement is important because the tail rotor, like the helicopter main rotor, will experience dissymmetry of lift when the helicopter is moving forward. In forward flight, the tail rotor will have blades that are advancing (moving toward the direction of flight) and blades that are

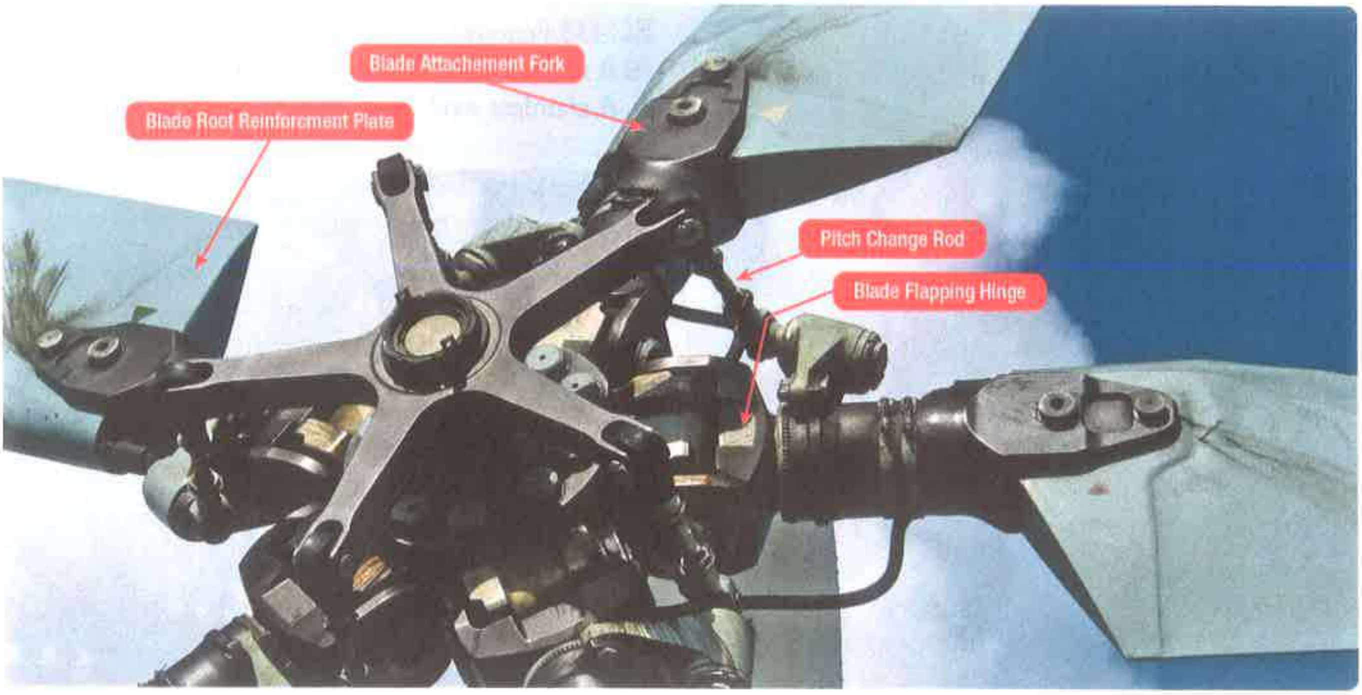


Figure 2-49. Tail rotor assembly on a MIL MI-26 helicopter.

retreating (moving away from the direction of flight). The advancing and retreating blades experience different velocities of airflow, and therefore will generate different amounts of force. In the same manner as the main rotor, the teetering or flapping of the blades changes the angle of attack and evens out the amount of force created by the blades. Even when the helicopter is not moving forward, the velocity of the air coming off the main rotor can interact with the tail rotor and cause dissymmetry of lift.

Fenestron Tail Rotors

The Fenestron-type tail rotor is made up of several airfoil shaped blades (between 8 and 18), all contained in an armored tunnel. Due to their location, the blades are protected from the effects of the helicopter forward movement and from the air pushed down by the main rotor. (***Figure 2-51***) Under these conditions, the flapping hinge is not necessary, which simplifies the design of the tail rotor head. Seen previously, the consequence of the flapping hinge is the creation of the drag hinge due to the Coriolis effect. If it is not necessary to adapt a flapping link, it is also not necessary to add a drag hinge. The only hinge needed is for collective feathering. Another advantage is the reduction in mass, limiting the stress and allowing the use of friction rings to replace bearings that require maintenance.

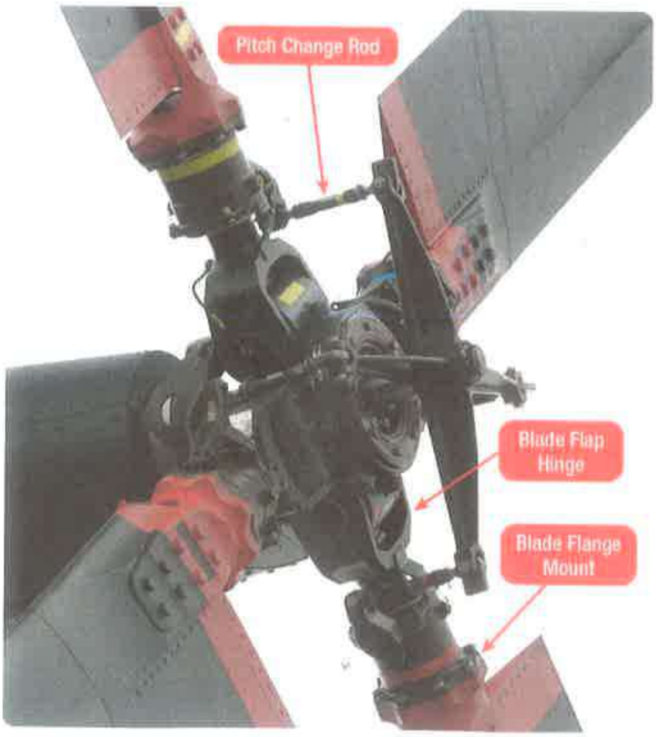


Figure 2-50. Sikorsky CH-53 tail rotor assembly.

In ***Figure 2-51*** the Fenestron is being viewed from the right side and the blades would rotate counterclockwise, bringing in air from the right and accelerating it through the guide vanes and out the left side. This would generate the forces necessary to counteract the torque of the main rotor blades, which are turning counterclockwise when viewed from the top. The guide vanes provide structural support and help to straighten out the swirling air coming off the rotating blades. If a helicopter equipped with a Fenestron, has a main rotor turning clockwise when viewed from the top, the rotating blades will be on the left side of the tail and the stationary guide vanes will be on the right. (***Figure 2-52***)

When looking at the Fenestron system in ***Figure 2-51***, we can see that the ten blades are not evenly spaced from each other. The uneven spacing produces an overlapping of acoustic vibrations, thereby providing a lower noise level in the cabin of the helicopter. It is important to work on the noise due to a Fenestron, as the air channeling with this type of tail rotor tends to produce a hissing noise.

Fenstron Blade Construction

The blades used on Fenestron tail rotors are constructed of either solid aluminum or a composite material made up of carbon-fiber fabric and a foam filling. Solid aluminum blades are used on the Airbus models SA-341, SA-342, EC-120, and EC-135, while the EC-155 uses the composite blades. The composite structure on an EC-155 Fenestron blade is made up of:

- A carbon fabric skin up to the blade root.
- A stainless steel sheet element on the leading edge.

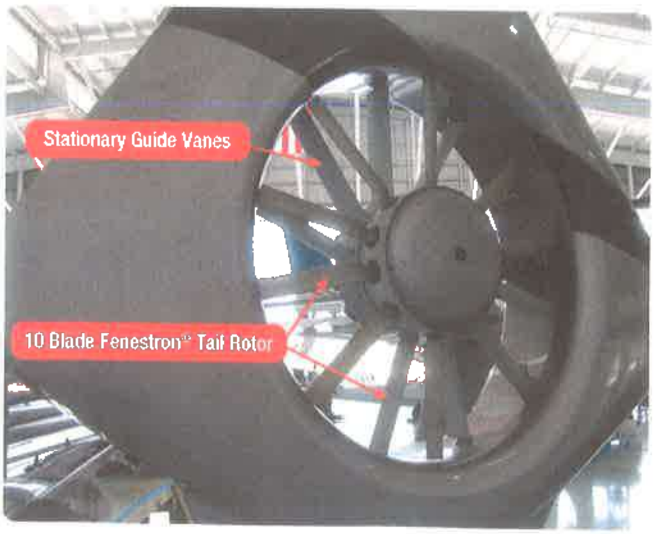


Figure 2-51. Fenestron tail rotor for an EC-135 helicopter.

Figure 2-52. Fenestron on SA341 Gazelle.

- Stainless steel bushes on the end of the blade root and on the sleeve.
- A plain, stainless-steel bush on the suction face of the blade root.
- A stainless steel shouldered bush on the pressure face of the blade root.
- A pitch change control crank pin on the trailing edge of the blade root.

The internal structure of the blade is made up of:
- Carbon fabric plies which act as a spar over the full length of the blade.
- A foam filing in the main section.
- The sleeve which houses the balance weights on the suction face, at the blade root.

Fenestron Blade Attachment

Figure 2-53 shows a Fenestron tail rotor assembly that has been disassembled. In the photo, the blades can be seen passing through the hub and attached to the pitch change ring of the blades. The blades are secured by means of a tension/torsion strap, also known as a torsion link bar and called the spider (silver piece in the center of ***Figure 2-53***). The spider is made up of several thin steel plates with holes at each end allowing them to be attached to the blades. They hold the blades against centrifugal force, while letting them autorotate if the pilot acts on a yaw control.

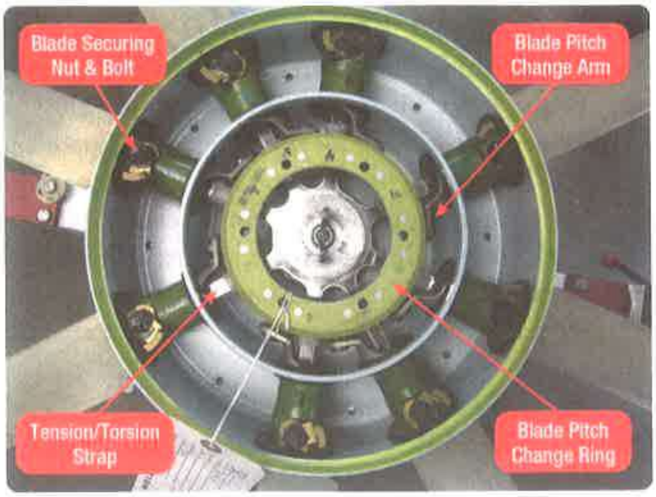


Figure 2-53. The inner workings of a Fenestron type tail rotor.

A close-up view of the method of securing the blade can be seen in ***Figure 2-54***, with the torsion link bar shown spread apart in the lower left.

As the pilot moves the yaw pedals, the blade pitch change ring moves in or out, causing the blades to change pitch collectively through their blade arms. The tension/torsion strap steel construction can hold the blades and twist, allowing the blades to change pitch and stay in the hub.

NOTAR Type Anti-Torque Rotor

An alternative to having rotating blades spinning at the rear of the helicopter is to have a set of rotating blades located within the body, just aft of the cabin. This set of blades is referred to as a fan, and this type of anti-torque system is referred to as NOTAR (No Tail Rotor).

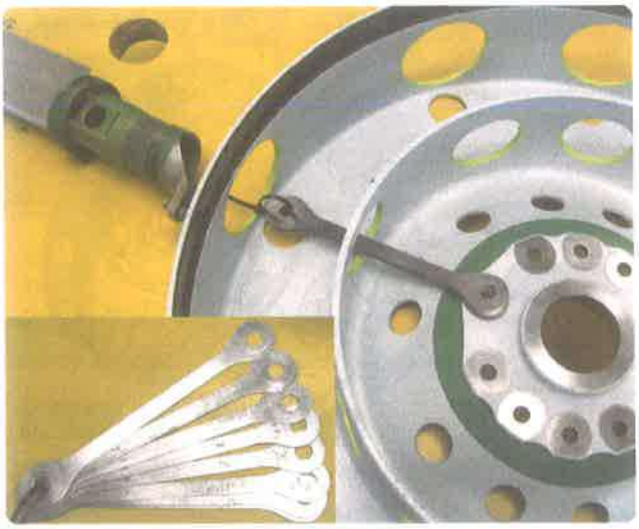


Figure 2-54. Close up view of Fenestron blade attachment.

An anti-torque fan, like on the MD-500, contains thirteen variable pitch blades. The fan provides an anti-torque force by furnishing a variable flow of low-pressure high-volume air through the tail boom and thrusters. During flight the anti-torque pedal position and pressure required to maintain a desired heading varies with changes in main rotor torque, altitude, and airspeed conditions. The variable pitch fan used on the MD-500 can be seen in ***Figure 2-55***.

NOTAR Blade Construction

The original airfoils used in the fan of the NOTAR system were of a hand layup composite construction, consisting of a foam core with a fiberglass/epoxy skin. Newer NOTAR airfoils utilize an injection molded long glass-fiber polypropylene which is a proprietary type of plastic. A substance known as a blowing agent is used when the airfoil is injection molded, causing the polypropylene to cure with a consistent weight and mass distribution. The airfoil is bonded to an aluminum spar and attachment arm that is cast as a single piece.

The polypropylene blades have been certified for a service life of 7 500 hours, and provide a 90% cost savings over the original composite blades. The original blades required 10 hours to produce and had a 20% scrap rate, compared to the new blades requiring 5 minutes to produce and having a 3% scrap rate.

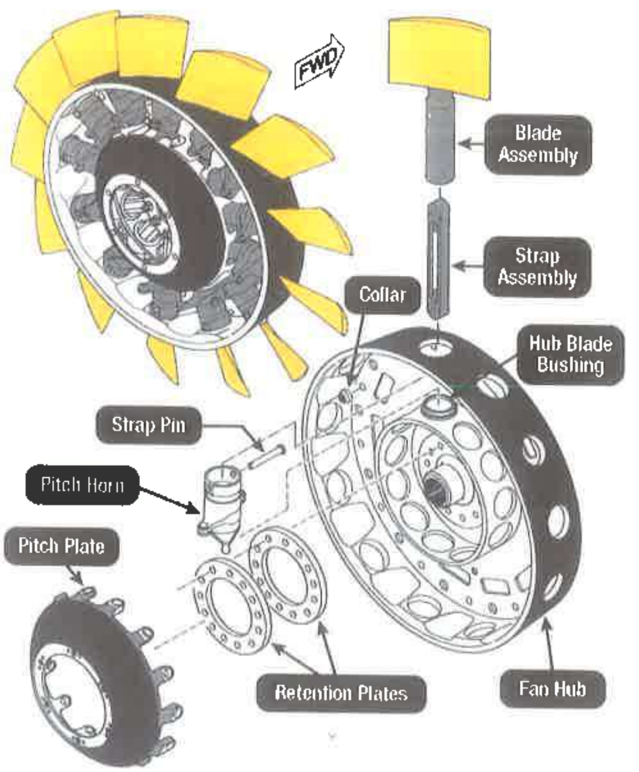


Figure 2-55. Variable pitch fan on NOTAR type helicopter.

NOTAR Blade Attachment

NOTAR blades are attached to the rotating fan hub by being bolted to a strap assembly, (***Figure 2-55***) which in turn is pinned to a pitch horn. The pitch horn for each blade is secured to the pitch plate. When the anti-torque pedals are moved on the flight deck, the pitch plate moves in and out, causing the pitch horns to rotate and the angle of the fan blades to increase or decrease. This assembly is quite the same as the Fenestron assembly. The difference is the function. The Fenestron generates a horizontal lift and the NOTAR fan generates an air flow.

TRIM CONTROL, FIXED AND ADJUSTABLE STABILIZERS

Vertical And Horizontal Stabilizers

Like fixed wing airplanes, most helicopters make use of vertical and horizontal stabilizers which serve a similar purpose with respect to the stability of the aircraft. Unlike the airplane however, the vertical stabilizer on a helicopter does not have a moveable flight control (rudder) attached to it. In a translational flight situation, the surface of the airframe facing the relative wind creates a drag force. The application of this force at the center of gravity with respect to the center of pressure results in a diving moment of the aircraft. (***Figure 2-56***) This moment will be compensated by the weight of the aircraft while flying "nose down", making the flight unpleasant and dangerous, especially for large aircraft. The integration of a stabilizer compensates for this phenomenon and allows translational flight with an attitude close to zero. (***Figure 2-57***)

Most often the horizontal stabilizer also has no moveable flight control attached to it, but a few do have a trailing edge surface that moves when the cyclic pitch control is moved. There are also a few helicopters that allow the entire horizontal surface to move, which in airplane terminology could be viewed as a stabilator. For those helicopters that do have a moveable surface associated with the horizontal stabilizer, it is not directly controlled by the pilot, but rather moves automatically in response to movement of the cyclic or because of sensors that detect the attitude of the helicopter. When a helicopter

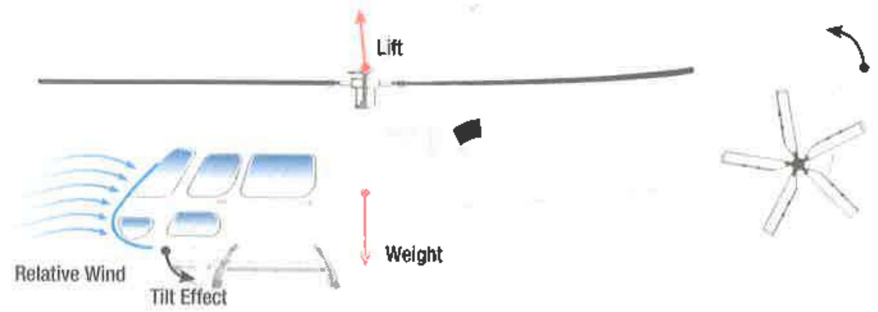


Figure 2-56. Forward flight without stabilizer.

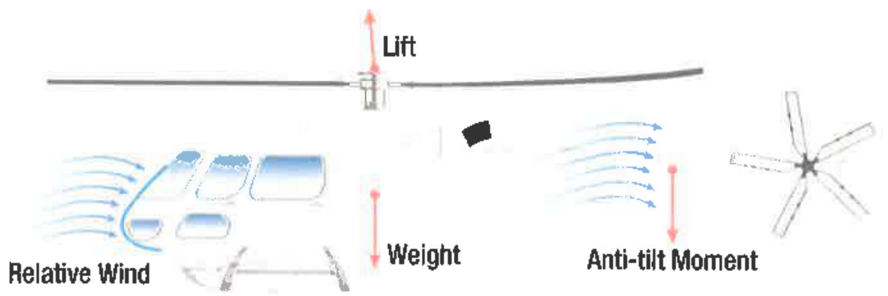


Figure 2-57. Forward flight with stabilizer.

transitions from vertical takeoff to forward flight, the additional lift on the main rotor blades as they pass by the rear of the helicopter cause the nose to be lowered. The horizontal stabilizer generates a downward force to lower the tail and keep the helicopter level.

Figure 2-58. Sikorsky SH-60 helicopter in forward flight.

Figure 2-58 shows a helicopter main rotor tilted up in the back, indicating forward flight, and its moveable horizontal stabilizer is positioned to provide an upward force on the tail to maintain level flight.

In ***Figure 2-59*** the same model helicopter is in a hover, and the moveable horizontal stabilizer has changed position to maintain the proper attitude of the helicopter. If this helicopter were to transition to forward flight by the pilot moving the cyclic forward, the moveable stabilizer would automatically start changing position by raising the trailing edge until it eventually would be in the position shown in ***Figure 2-58***.

Figure 2-59. Sikorsky SH-60 helicopter in a hover.

Most helicopters have a fixed horizontal stabilizer, like the one shown in ***Figure 2-60*** on an Aerospatiale AS-350. As can be seen in the photo, the horizontal stabilizer is an upside-down wing, with the cambered side on the bottom and the relatively flat side on the top. The wing on an airplane is designed to provide an upward force to lift the airplane and support it in flight. When the helicopter is flying in a forward direction, the horizontal stabilizer is designed to provide a downward force to keep the nose of the helicopter up and to provide a level flight attitude.

During forward flight, a vertical stabilizer provides directional stability around the vertical axis (yaw axis). In addition, during forward flight the vertical stabilizer

Figure 2-60. Aerospatiale AS-350 horizontal stabilizer.

helps to counteract the torque produced by the spinning main rotor. To counteract the torque, the vertical stabilizer is offset a few degrees so that it produces a force (lift) that pushes the tail in the opposite direction that torque would try to make it rotate. If the main rotor is spinning counterclockwise when viewed from the top, torque will try to spin the helicopter clockwise, making the tail move to the left. By offsetting the leading edge of the vertical stabilizer a few degrees to the right, a force will be created that tries to move the tail to the right. (***Figure 2-28***) There are strakes on the tail boom, to create side forces to help counteract torque.

SYSTEM OPERATION: MANUAL, HYDRAULIC, ELECTRICAL AND FLY-BY-WIRE

Manual Operation

All manual operations are described in the different sections of this chapter. When the pilot moves a stick (cyclic or collective) the various connecting rods control, through the mixer, the swashplate which tilts or moves up/down. As the swashplate tilts or moves up and down, it changes the angle of attack of each blade to move the helicopter in the desired direction.

Hydraulic Operation

Hydraulic systems and their different functions will be studied in chapter 12. For the flight control system, hydraulics allows the pilot to move a heavy load effortlessly. In fact, the stresses applied to the blades are evaluated in tons. Hydraulic servo controls allow the angle of attack to be changed and prevent the load applied to the main rotor from passing into the flight control. This system is called the irreversibility of hydraulic servo drives.

Electrical Fly-By-Wire Operation

Fly-by-wire technology has long been used on fixed wing aircraft but is only in development on rotorcraft. Manufacturers have been reluctant to challenge safety on rotorcraft which absolutely cannot continue to fly if the flight control system is lost. While a light helicopter can land safely with autorotation, it is more difficult with a heavy helicopter. The engines must be doubled to increase power, but also to increase safety. In the event of an engine failure, the second can eventually replace the loss of power to continue flying and land in an emergency. Hydraulic redundancy on a heavy helicopter is the solution chosen to ensure safety. Two separate circuits transmit hydraulic power to each sensitive system such as the main and rear servo controls, as well as for the landing gear extension. Two hydraulic reservoirs, two pumps and two chambers on each servocontrol separate the risk of total control loss in flight.

While the servo drives are usually actuated by mechanical rods, they can also receive commands electrically. When the pilot moves the sticks, the controls do not move rods and bellcranks but create an electrical signal sent to electrical actuators to move the hydraulic power system. While the complete replacement of the connecting rods is done for a small amount of the helicopter type, electric assistance has long been used to help the pilot stabilize the rotorcraft. Some assistance systems such as the "Stability Augmentation System" (SAS) are commonly used.

Stability Augmentation System (SAS)

Some helicopters incorporate an SAS to help stabilize the helicopter in flight and in a hover. The original purpose and design allowed decreased pilot workload and lessened fatigue. It allowed pilots to place an aircraft at a set attitude to accomplish other tasks or simply stabilize the aircraft for long cross-country flights. SAS systems reduce pilot workload by improving basic aircraft control harmony and decreasing disturbances. These systems are especially useful when the pilot is required to perform other duties, such as sling loading and search-and-rescue operations. Other inputs such as heading, speed, altitude, and navigation information may be supplied to the computer to form a complete autopilot system.

Modern helicopter SAS architecture is generally an attitude based system that accepts attitude source data using a combination of the installed attitude gyro or digital Attitude Heading Reference System (AHRS) inputs, and motion sensors located in the Flight Control Computer (FCC). The FCC commands the servo actuators to apply small corrections to the cyclic as required to maintain attitude. The SAS is designed to maintain the helicopter at the datum to which it has been trimmed. It uses a simple feedback control in which a rate gyro senses pitch rate, for example which on integration, provides a correcting input at the swash plate (if this is the means of rotor control).

SAS Components

The SAS along with the associated autopilot system consists of the following components:

- HeliSAS Control Panel (HCP) – The HCP interfaces with the FCC. This push button panel located in the cockpit within the pilot's reach provides for engagement of the SAS and the desired autopilot mode selections of altitude hold, heading hold, navigation signal tracking, and vertical (approach) navigation features, which are controlled via six push buttons on the panel.
- Flight Control Computer (FCC) – The Flight Control Computer receives inputs from the HCP as well as from the on-board NAV, GPS and Attitude systems and commands the pitch and roll servos to perform the selected autopilot function. Internal to the FCC are three body axis rate sensors, a tri-axial accelerometer, two differential pressure sensors, and one absolute pressure sensor. The rate sensors and accelerometer are used to sense the rotational velocities and acceleration of the SAS in three orthogonal axes. The pressure sensors are used to detect and calculate the indicated airspeed and altitude of the airframe.
- Roll Servo – It receives roll error signals from the FCC and provides roll correction to the cyclic controls through electromagnetic clutches located within the servo.
- Pitch Servo – It receives pitch error signals from the FCC and provides pitch correction to the cyclic controls through electromagnetic clutches located within the servo.
- Attitude Gyro – It sends attitude signals to the FCC. The attitude reference for the FCC can also be provided by a digital AHRS (if provisioned).
- AP DISC Switch – The cyclic-mounted AP DISC switch disengages the SAS mode when pressed. If the autopilot is engaged, pushing the AP DISC button causes the autopilot modes to disengage, while the SAS remains engaged. Pushing the button twice causes the autopilot and SAS to disengage. If only the SAS is engaged, pushing the button once disengages the SAS.
- Cyclic Controls – The SAS installation includes a modification to the cyclic controls to add a momentary trim button and an AP/SAS DISC button. The momentary trim button provides a force-trim-release function to allow the pilot to re-trim to a new pitch or roll attitude.
- Servos – The servos are electromechanical servo-actuators consisting of a DC brushless commutating motor, low ratio gearbox, clutch, and servo position feedback resolvers that control the pitch and roll axes of the helicopter. The servo-actuators are connected to the flight control system in parallel with the basic helicopter control rods and have manual servomotor back-drive capability. The clutches consist of an electromagnetic pressure plate design that disconnects the servo-actuators from the flight control system when the SAS is selected off. Loss of power to the clutches causes them to fail to the open position with the clutch faces separated. The gear ratio between the cyclic stick and the servo-motor is sufficiently low so that the helicopter can be safely flown with the SAS disengaged and the clutch stuck closed (i.e., the pilot can back drive the gearbox and servo-motor with negligible resistance).

SAS Operations

The Helicopter SAS is typically a two-axis attitude hold, attitude command, flight control system. The system has two basic functions: to aid with aircraft stability and autopilot outer loop control modes for altitude hold, heading selection and navigation sensor coupling. The SAS and Autopilot System provide a significant reduction in pilot workload, from takeoff to landing. The SAS mode should be engaged prior to liftoff and disengaged following touchdown. The various autopilot modes can only be engaged when the SAS mode is already engaged and the airspeed is greater than the designated minimum engagement airspeed for the autopilot (example: 44 Knots Indicated Airspeed for a Bell 206).

A pair of servos is coupled to the cyclic through electromagnetic clutches as a means to control the helicopter for a given mode of operation. One servo controls the cyclic about the roll axis, and the other servo controls the cyclic about the pitch axis. These servos are driven by error signals received from the FCC, which in turn receives inputs from AHRS, Heading System (HSI or Electronic Flight Instrument System (EFIS), VHF Navigation Receiver and GPS Navigation Receiver.

The SAS and autopilot (ATT) systems make it possible to fly for indefinite periods, with hands off the cyclic when in ATT mode. Due to the unstable nature of helicopters, the pilot must always be prepared to assume immediate manual control of the cyclic in the event of an automatic SAS disengagement resulting from a system failure. In addition, despite an engaged SAS mode and the main rotor speed governor in the helicopter, the pilot is not relieved of their responsibility to always monitor closely, the helicopter attitude and the main rotor RPM.

ARTIFICIAL FEEL

With cable and connecting rod type flight controls, the pilot feels aerodynamic loads through the control stick. The introduction of wired flight controls and servos in flight control systems eliminates this tactile feedback, on flight controls which are equipped with non-reversible hydraulic servo controls. As a result, the force necessary for the pilot to control these servos is extremely low (of the order of a few newtons).

The pilots therefore do not feel the steering force and their analysis of the behavior of the machine is erroneous. They do not have the sense of direction through the control sticks and do not realize when their controls stop. Their direction lacks fluidity and is too abrupt. Another risk is that without feeling the loads, the pilot no longer feels the behavior and the limits of the helicopter. Therefore, it is important to introduce systems into the process to recreate tactile feedback with artificial feel devices. These devices can be mechanical or hydraulic.

Friction

The displacement of the cyclic or collective stick is quite easy thanks to the hydraulic assistance. Without feeling, the pilot can have wide and rapid movements, acting violently on the system at the risk of irreparable destruction and a crash of the helicopter. An adjustable friction system may simply solve the problem. For example, at the bottom of the cyclic stick, a large nut allows the pilot to increase or decrease resistance by compressing a spring which increases or decreases the pressure of the friction cap on the spherical joint. It then becomes more or less difficult to move the cyclic stick. (*Figure 2-61 and Figure 2-62*)

On the collective pitch, a friction system allows each pilot to adjust the hardness to move the stick, according to his/her wishes. (*Figure 2-63*)

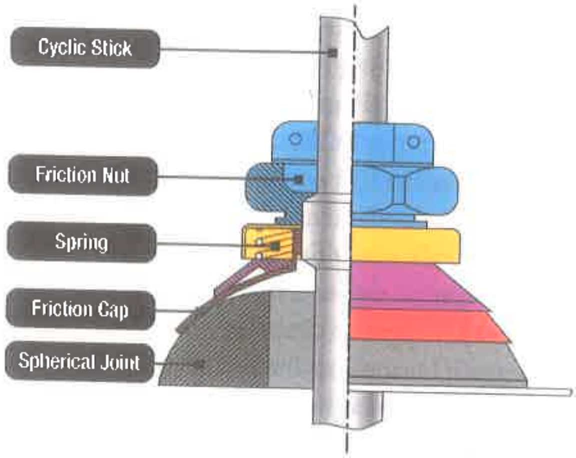


Figure 2-61. Cyclic pitch stick friction.


Figure 2-62. Cyclic pitch stick friction.

Figure 2-63. Collective pitch stick friction.

Elastic Rod

An elastic rod is installed in the flight control system. When the pilot gives an order by moving the rod, a spring is compressed according to the load supplied. The more the spring is compressed, the more difficult it is for the pilot who has the feeling of the applied load and feels the limit of the mechanics. (***Figure 2-64***)

This elastic rod is generally used on the yaw chain and is used as a safety unit as well as an artificial feeling. In the event of a control cable breakage, the resilient rod will automatically reduce the incidence of the tail rotor blades to a determined value, allowing the aircraft to land safely without changing the course on the center line.

Hydraulic Damper

A similar system helps reduce travel motion without limiting the range of motion. A piston with a calibrated hole allows the passage of the hydraulics but with this laminated device, the travel speed is limited. This system is used on the yaw line to avoid destroying the tail beam junction. Due to the fragility of the material, an excessive speed of the pedal movement risks breaking the junction and folding the tail on the structure. When the piston moves to the right, the hydraulic goes from the right chamber to the left chamber through the laminated hole calibrated to determine the speed. The excess of hydraulic fluid due to the chamber volumetric difference, refills the reservoir. (***Figure 2-65***)

In the other direction, hydraulic fluid goes from the left chamber to the right chamber and the lack of volume is sucked from the reservoir. This exchange with the tank allows the system to be permanently purged, avoiding air bubbles which will disrupt proper operation. Connected directly on the pedals lever, the yaw damper is located under the pilot floor. (***Figure 2-66***)

BALANCING AND RIGGING

Balancing and rigging of the flight control system is important to avoid locking and distortion of the rods. Before each flight, pilots and mechanics check the fluidity of the flight control system. The manufacturer must organize the different rods and cranks to reduce, increase or just transmit the movement. It is also necessary to change the direction or double the link as for the lateral movement. Everything is possible with the different cranks, depending on the connection positions on them.

Rods

A rod is a simple aluminum tube with a connecting eye bolt at each end. The connectors are adjustable in length due to the thread allowing the rotation of the body to obtain a very precise installation. (***Figure 2-67***) The position of the connector is locked with a nut tightened

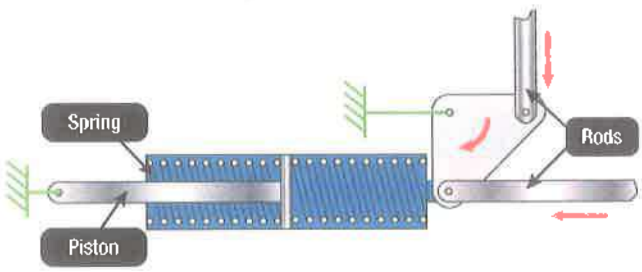


Figure 2-64. Elastic rod.

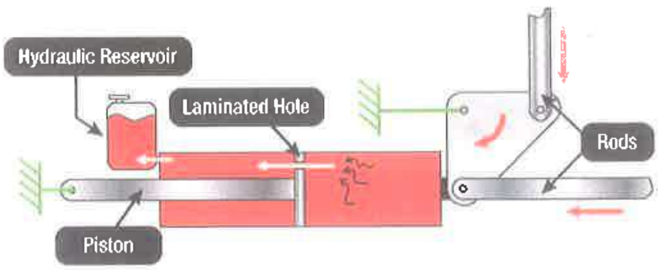


Figure 2-65. Hydraulic yaw damper.

to it, creating a stress on the thread. This nut is in turn locked in rotation with a locking wire to be sure that it will not be loosened with the helicopter vibrations. (*Figure 2-68*)

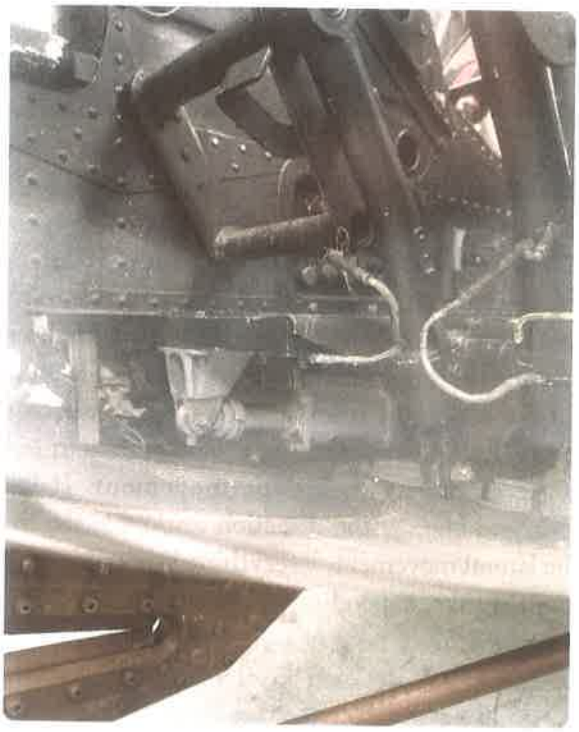


Figure 2-66. Hydraulic yaw damper.

Figure 2-67. Rods and bellcranks on the mixer unit.

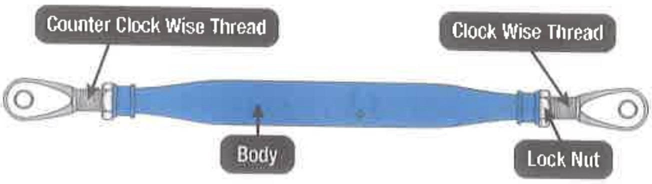


Figure 2-68. Rod description.

Two threads are opposite allowing the length to be adjusted. When the body rotates in one direction, the two connecting bolts reduce the overall dimension. By turning in the opposite direction, the dimension increases allowing the size to be adjusted to the correct length to wedge the flight control rods as explained. (*Figure 2-69*)

Transmission Unit

A bellcrank is a rotating part on which are connected two rods, one inner and one outer. Their shapes depend on the space where they are connected and the stress to be transmitted. In case of high load, they will be bigger or made of a special material to withstand greater loads. Generally, they are before the high load part limited by the servo control and are simply made of aluminum alloy. Four bellcranks can be defined by their different functions.

Relay

To avoid having too long rods with the risk of bending and reduced length, the manufacturer prefers to separate a long rod in two shorter segments. The wish is just to transmit the movement from point A to point B without increasing or reducing the movement. In this condition the inner and the outer rods are connected to the same point (C) on the bellcrank called a relay. (*Figure 2-70*)

Balancing

Sometimes it is necessary to increase or decrease an ordered movement. A small order given by a stick (cyclic or collective) can give a large order. The reverse is also true. In this case, the inner and outer rods are not connected at the same point and the ratio depends on the distance between the rod connections and the relay rotation center. (*Figure 2-71 and Figure 2-72*)

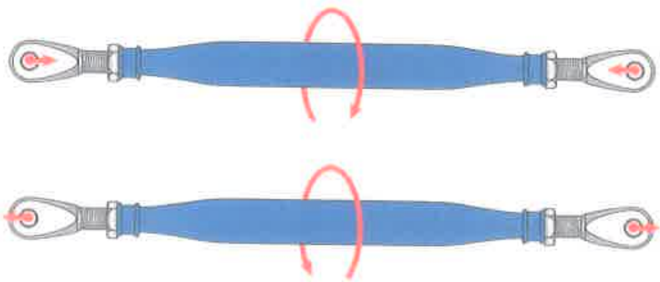


Figure 2-69. Length adjustment.

The last relay is a relay on which three rods are connected to double the movement as on the lateral chain. It is necessary to have three points to control the position of the swashplate. To transform the two pure movements which are the longitudinal movement and the lateral movement, it is necessary to transform a pure movement (generally the lateral movement) into two compound movements which have the same value but not the same direction. (***Figure 2-73***) The center of rotation of the relay is in the middle of the two outer connecting rods.

Bellcrank

The control rods from the cyclic and collective sticks are under the cabin floor and have to reach the top of the helicopter. It is obvious that the rods must change from the horizontal direction to the vertical direction. To effect this change, a lever is inserted in the chain. Like the relay, it is a simple piece of aluminum alloy that can change the range of motion depending on the rod connection. (***Figure 2-74***, ***Figure 2-75*** and ***Figure 2-76***) The best example is the mixer consisting of three cranks and a collective command.

Bellcranks and relays are a reliable component of the flight control system but require a lot of maintenance. The bearings at the end of each rod and at the center of rotation require frequent lubrication and for access require removal of the structural panels. When performing maintenance, it is important that the mechanic does not hang on the aluminum rods due to the risk of bending and thus reducing the size. The consequence is the modification of the original setting, hence requiring the replacement of the connecting rod and the rigging of all the flight controls to be reset to the manufacturer's settings.

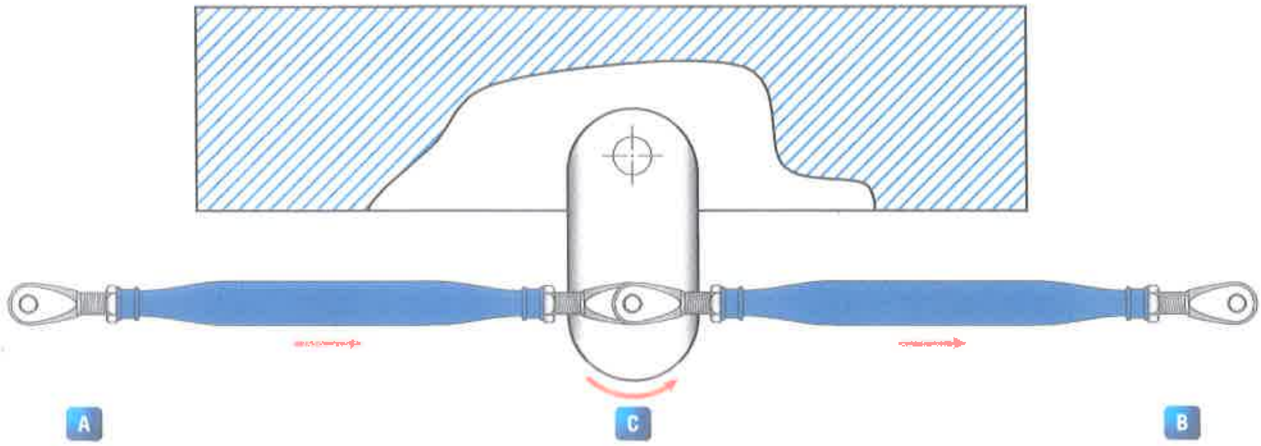


Figure 2-70. Simple relay.

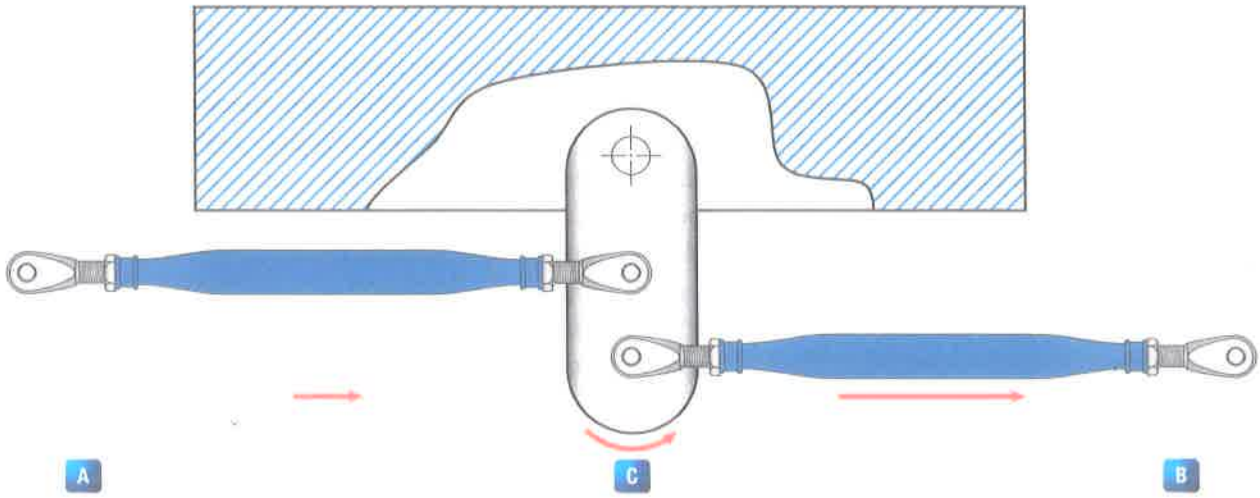


Figure 2-71. Increasing relay.

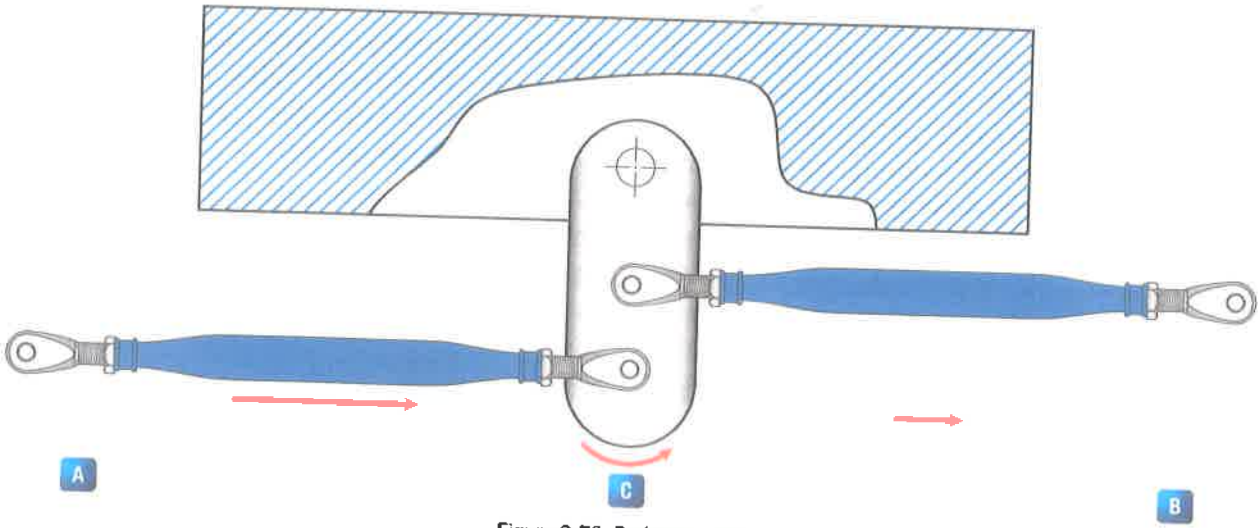


Figure 2-72. Reducing relay.

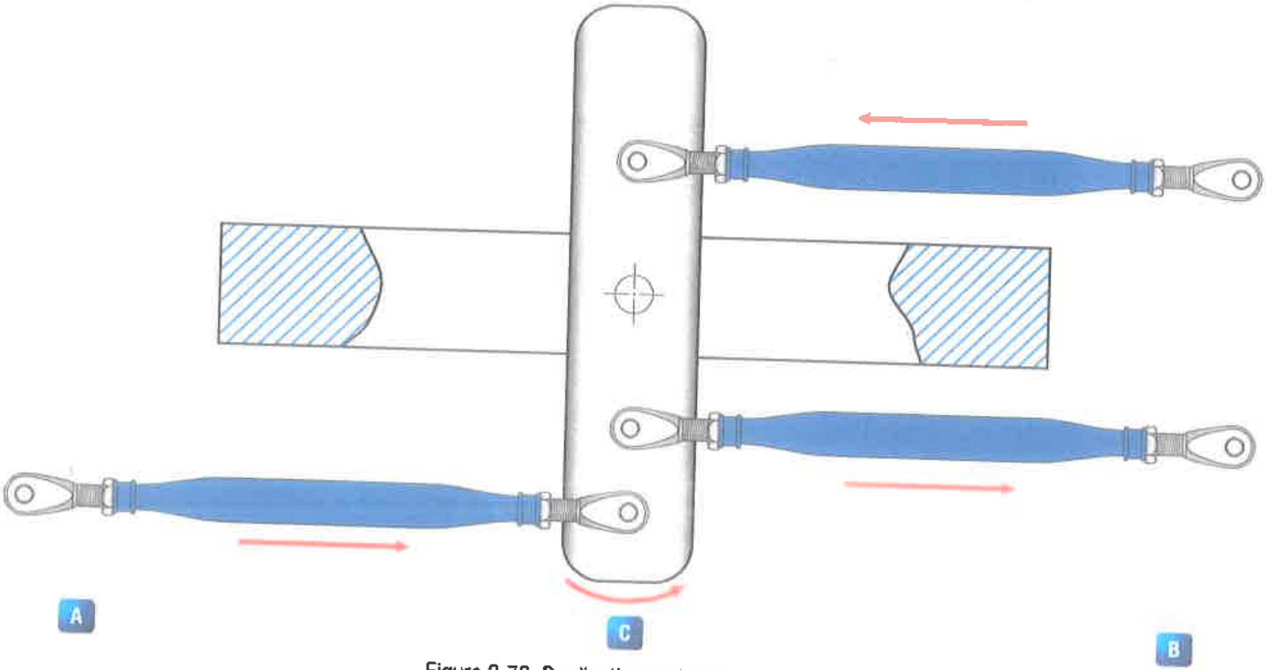


Figure 2-73. Duplicating and reducing relay.

RIGGING

Rigging involves positioning all relays and cranks in the correct position, and setting the rods at the correct length to connect them together. It looks like returning the entire flight control system to neutral. Due to pipes, wires or structural obstacles, the manufacturer decided on the original position of each relay or crank to be sure that during the flight nothing will damage the rods or block the movement preventing the pilot from controlling the helicopter risking a crash. Some special pins, ***Figure 2-77***, allow the relay to be locked in only one position.

After disconnecting all the rods, the mechanic begins to lock the various sticks, bellcranks and relays by inserting pins through the moving part and a reference hole in the structure. (***Figure 2-78***)

Pins are marked with a red flag to make sure none are forgotten after maintenance. The next step will be to connect the different rods to the lever. For this operation, three types of rods exist, the fixed, the semi-adjustable and the adjustable. The fixed rod is a rod adjusted by the manufacturer, the length of which the mechanic does not have authority to modify. It connects a rigged lever to another which has no determined or locked position.

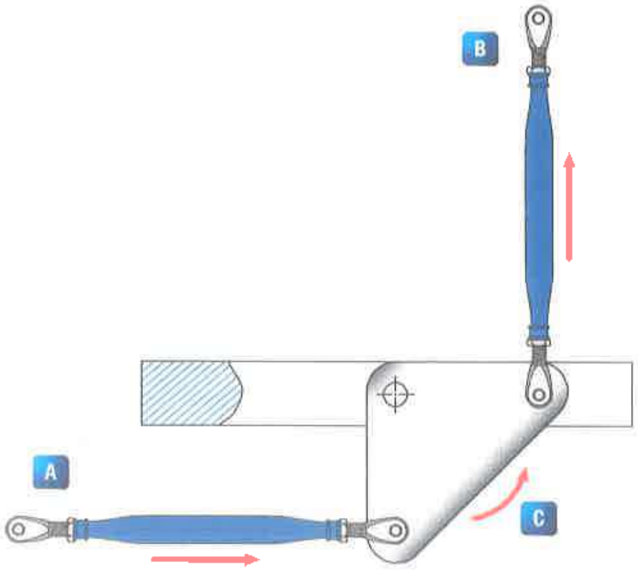


Figure 2-74. Simple bellcrank.

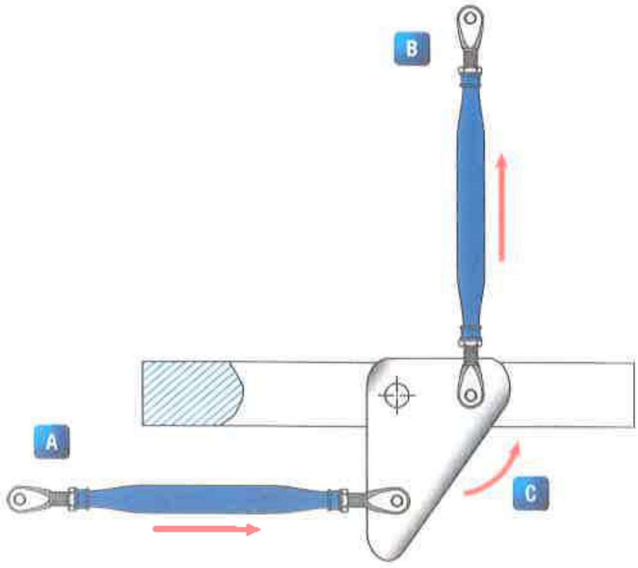


Figure 2-76. Reducing bellcrank.

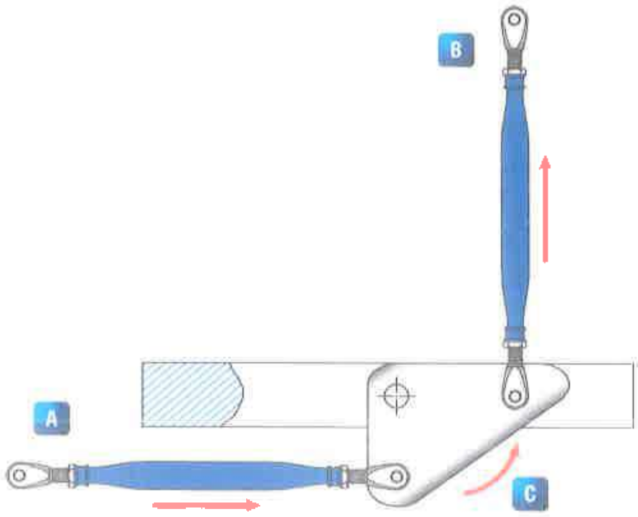


Figure 2-75. Increasing bellcrank.

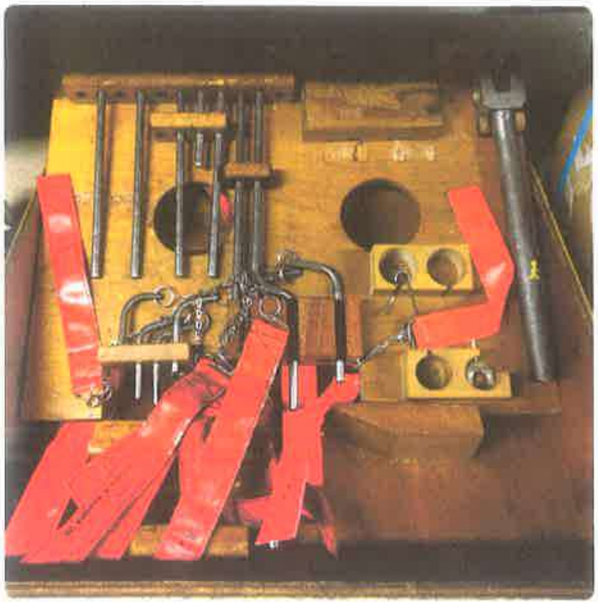


Figure 2-77. Rigging-pin tool kit.

In this condition, the free lever is forced to assume a single position which is safe for chain movement.

From this free/locked lever, a semi-adjustable rod will be installed to connect it to the second attachment point. A semi-adjustable connecting rod is not a connecting rod of which only one of the two connecting bolts can be adjusted, but a connecting rod that the mechanic can adjust during the first installation. After this connection, it may not be allowed to change in length and so the rod becomes like a fixed rod. It may be necessary to adjust it once due to manufacturing tolerances. In fact, it is not technically possible to have the exact same distance between the levers on different helicopters. It is important to remember that to lock two rigging points it is necessary to install a semi-adjustable rod, ***Figure 2-79***, and that in case of free levers and multiple rods between two attachment points, a single rod will be semi-adjustable, and all the others will be fixed rods to determine the position of the levers. (***Figure 2-80***) The position of the semi adjustable rod is defined by the manufacturer.

Physically all rods look the same. To tell them apart, colored rubbers cover the locknuts to make sure the mechanic sees them if he/she wants to change their length. Black is chosen for the fixed rods and white for the semi-adjustable ones. (***Figure 2-81***)

Figure 2-78. Relay locked by a pin.

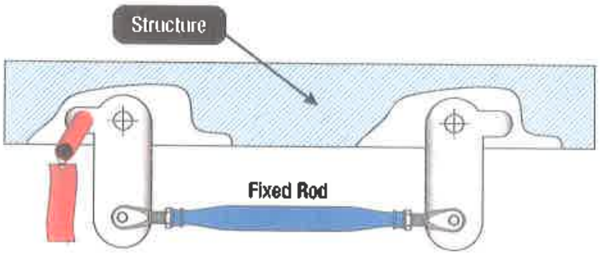


Figure 2-79. Single rod between two rigging points.

Figure 2-81. Fixed and semi adjustable rods.

The last rod, without colored rubbers, is an adjustable rod and its length can be changed with each adjustment of the flight controls. The length is not important, only the result must be considered in accordance with the adjustment tolerances. The pitch change rod is an adjustable rod, and its length is set to balance vibrations. The actuator control rod of the servo controls is also adjustable. Their lengths are defined to position the swashplate according to the positions of the sticks. (*Figure 2-82*)

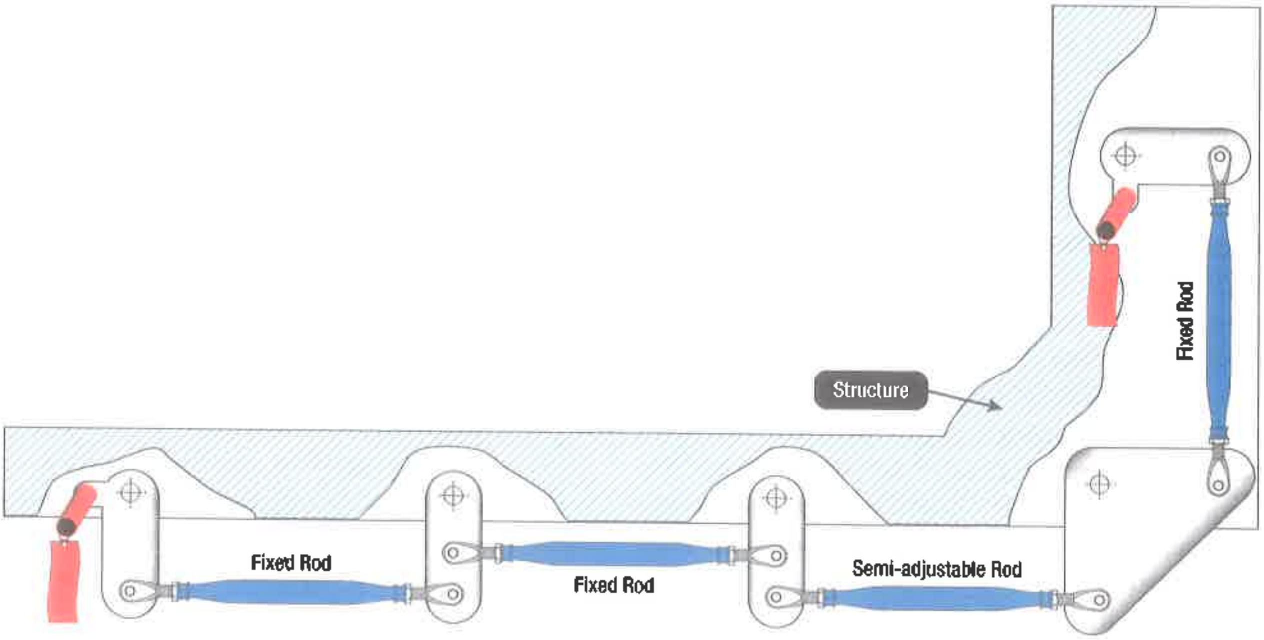


Figure 2-80. Multiple rods and levers between two attachment points.

Figure 2-82. Pitch change rod and actuator control rod of the servocontrols.

Question: 2-1
Which control movement causes the helicopter to transition from a hover to forward flight?

Question: 2-2
In addition to pulling the collective upwards, what else must be done to cause a helicopter to climb?

Question: 2-3
Which are the 6 basic components of a swashplate?

Question: 2-4
In which way does the swashplate on the main rotor differ from on the tail rotor?

Question: 2-5
Name 4 ways in which the main rotor torque can be controlled on a helicopter.

Question: 2-6
What is the primary method of reducing flapping of rotor blades?

Question: 2-7
In which mode of flight is blade flapping not a concern?

Question: 2-8
What force serves to counter and limit the extent of flapping on a turning rotor blade?

ANSWERS

Answer: 2-1
Moving the cyclic forward, moves the main rotor to a nose down position.

Answer: 2-2
Increasing the throttle (which is often done automatically by a correlator).

Answer: 2-3
Upper rotating plate, lower fixed plate, stationary and rotating scissors, control rods, spherical joint, bearings.

Answer: 2-4
The tail rotor does not have a swashplate.

Answer: 2-5
A tail rotor (standard or Fenestron), dual counter rotating main rotors, a NOTAR system, a vertical tail fin.

Answer: 2-6
By changing the angle of attack on each blade as it rotates from an advancing to retreating position.

Answer: 2-7
While hovering when there is no horizontal motion.

Answer: 2-8
Centrifugal force.

SUB-MODULE 03

PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY → B1.3 B1.4

Sub-Module 03

BLADE TRACKING AND VIBRATION ANALYSIS

Knowledge Requirements

12.3 - Blade Tracking and Vibration Analysis	
Rotor alignment;	3
Main and tail rotor tracking;	
Static and dynamic balancing;	
Vibration types, vibration reduction methods;	
Ground resonance.	

12.3 - BLADE TRACKING AND VIBRATION ANALYSIS

INTRODUCTION

The critical component of a helicopter is the rotor system, a complex collection of rotating assemblies that allow flight characteristics unavailable to fixed wing aircraft.

Rotor blades move at rapid speeds around a stationary point. While the helicopter is in the air, these rotors are very vulnerable to deterioration as they travel at different angles to achieve the proper lift. Excessive vibration levels caused by rotating components can lead to their premature wear and failure.

The rotors should be created and categorized into matching sets to function smoothly during their life cycle. Sometimes, they might endure minor damage, depreciate, or undergo erosion. At such times, the rotor becomes unbalanced. Moreover, the main rotor is not the only rotating assembly of concern in a helicopter; there are other critical assemblies such as the tail rotor, drive shafts, and oil cooler fans. Therefore, it is essential to reduce the vibration levels of these components to a minimum to extend the longevity and safety of the helicopter. Rotor tracking and balancing processes allow the maintenance technician to reduce vibration levels. But before balancing any component, the type of vibration causing the imbalance has to be determined.

VIBRATION TYPES

The engine, gear boxes, shafts, the tail and the main rotors themselves create different vibrations which damage the mechanisms and make the flight uncomfortable for the passengers and crew. It is important to understand what a vibration is. In a general sense, anything that modulates back and forth, in and out, or up and down is vibrating. A vibration is a periodic "wiggle" in time; a repeating movement around a position reference.

- The extent of the wiggle is defined by its amplitude.
- The repetition of the wiggle per second is defined by the frequency.

Properties and/or characteristics of vibrations are conceptualized as a sine wave. (*Figure 3-1*)

- The displacement or extent of the movement is defined as the vertical line (the abscissa), and the time or frequency is defined as the horizontal line (the orderly).
- The amplitude defines the movement around the reference point (zero) and the peak-to-peak defines the total displacement.
- The period corresponds to the time in seconds to complete one cycle.
- The frequency is the number of cycles completed in one second.

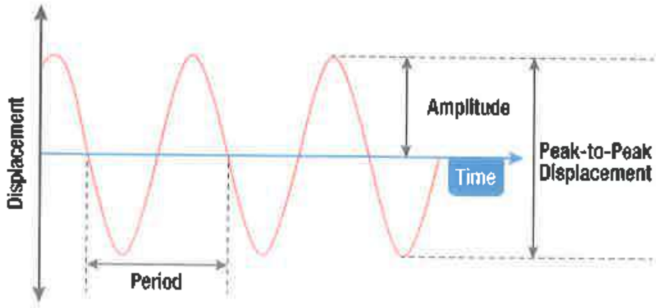


Figure 3-1. The vibration characteristics.

VIBRATION EXAMPLE

During a linear displacement, the mass starts from the equilibrium position and moves to the upper position. This movement defines the amplitude. (*Figure 3-2*) It goes down, crosses the equilibrium position, and continues to the lower one. When it comes back to the equilibrium position a cycle is done and defines the period. (*Figure 3-2*)

During a flexion displacement, the board is at the equilibrium position and flexes to the up position defining the amplitude. (*Figure 3-3*) It then flexes down, crosses the equilibrium position, and continues to the lower position. When it comes back to the equilibrium position a cycle is completed, again defining the period. (*Figure 3-3*)

Each rotating system on the helicopter, including bearings, rotors, or gear boxes can generate its own harmonics, with the total (or combination) of those harmonics within the system being the total vibration. The difference of time between vibrations is the frequency. To find the components generating the harmonics it is necessary to measure them and then to refer to the manufacturer's documentation for comparison with the acceptable reference harmonic.

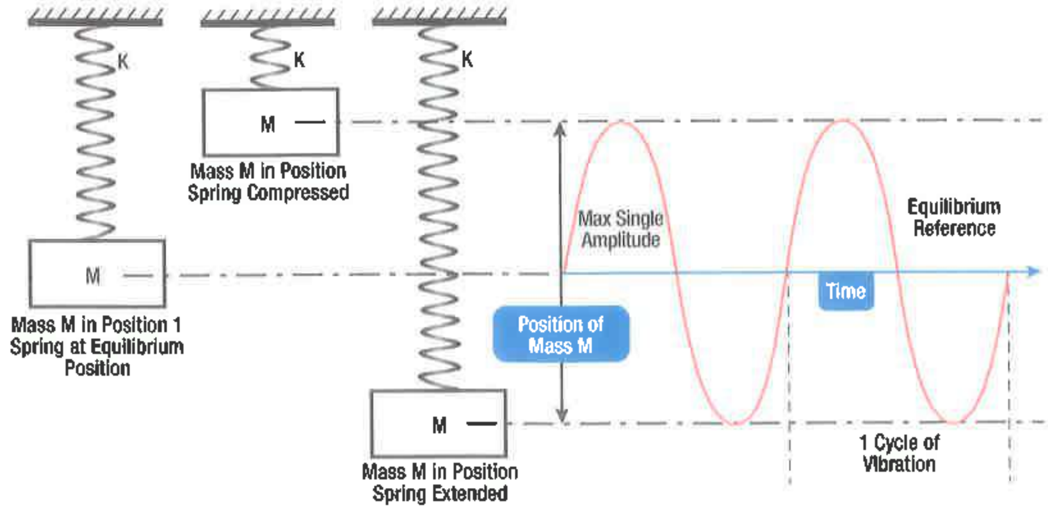


Figure 3-2. Linear vibration representation.

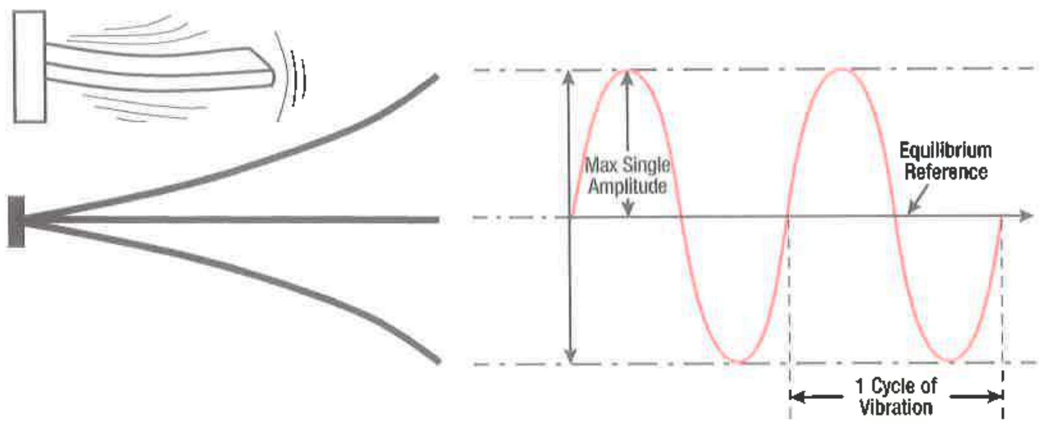


Figure 3-3. Flexion vibration representation.

The manufacturer references the acceptable frequency and the amplitude for each vibration. To control a vibration, after frequency adjustment, some tools are used to measure the amplitude. The factor causing the material to be deteriorated is not the frequency but the amplitude. If the amplitude is more than the acceptable tolerance, it is necessary either to replace the faulty component such as a worn bearing or gear box or to adjust the problem as in the case of a rotor. In aeronautics two types of vibrations are defined, one involving an imbalance or one involving tracking.

IMBALANCE

An imbalance appears in a rotating system in the case of a non-balance of the weight of a rotating object around the rotation axis. The force on the non-balanced point tends to move the system out of its rotation axis and so creates a vibration. The more unbalanced the weight, the more pronounced the amplitude. The frequency of this vibration depends on the rotation speed. When the rotating system is balanced, the system turns exactly around the rotation axis (***Figure 3-4***) and there is no vibration. (***Figure 3-5***)

When the rotating system is unbalanced, the system is not turning exactly around the rotational axis and a vibration appears. (***Figure 3-6 and Figure 3-7***)

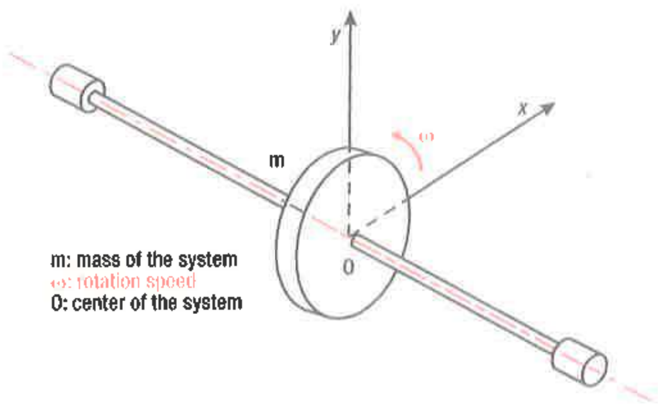


Figure 3-4. A balanced rotating system.

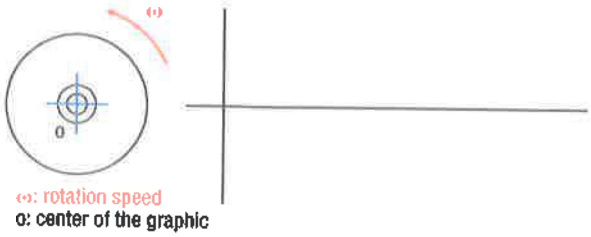


Figure 3-5. A balanced rotating system vibration representation.

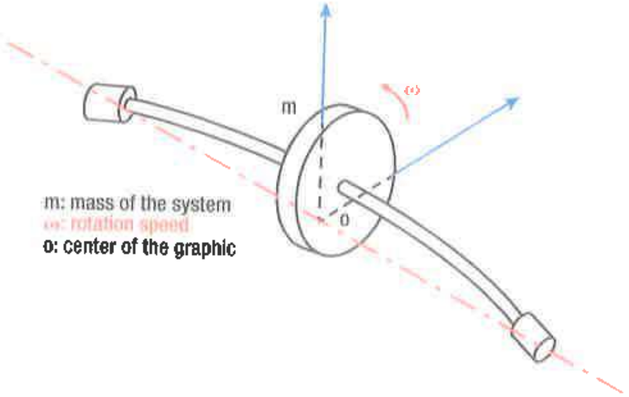


Figure 3-6. An unbalanced rotating system.

TRACK

This vibration caused by track is specific to the rotor. Tracking errors come from the non-alignment of the blades which are not turning in the same rotating plane. Concerning the main rotor, a tracking error occurs when one blade wants to go up and pulls the rotor mast to the top, and another one wants to go down, pulling the rotor mast to the bottom. When all the blades are turning in the same plane, no perturbation disturbs the flight and there is no vibration. (*Figure 3-8*)

Figure 3-9 shows how when one or more blades are not turning in the identical plane, an axial movement appears and creates a vibration. This vibration is perpendicular to the rotating plane or in the rotation axis. To fix this problem it is necessary to adjust the flight of each blade to be sure that they are all turning in the same rotational plane. When this error concerns the main rotor it is a vertical vibration. If for the tail rotor, the vibration is horizontal.

GROUND RESONANCE

The phenomenon of ground resonance should not be overlooked. Its consequences can range from a simple discomfort to complete destruction of the helicopter. The particularity of this vibration is that it is the combination of two or more vibrations, with each having the same harmonic and in the same direction. In this

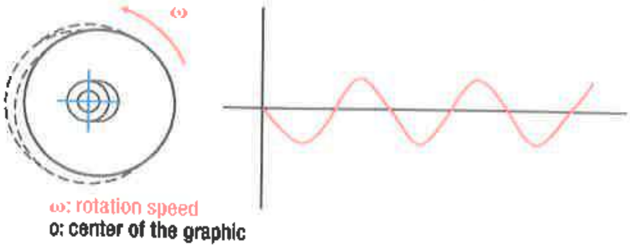


Figure 3-7. An unbalanced rotating system vibration representation.

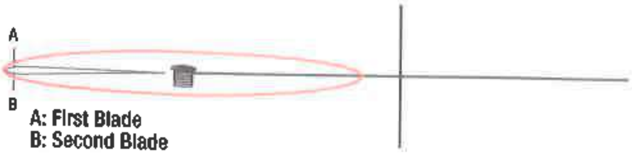


Figure 3-8. Track representation when all the blades are turning in the same plane.

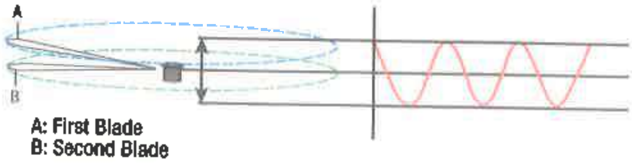


Figure 3-9. Track representation when all the blades are NOT turning in the same plane.

special case, the vibrations will amplify each other and generate a dangerous amplitude. When on the ground, it occurs when a helicopter vibration is phased with the landing gear system acting like a shock absorber.

The down movement of the track compresses the gear. When the up movement of the track then occurs, if the damper expansion is in phase with it, this up movement is amplified. (*Figure 3-10*)

If nothing is done to stop this amplification, there is a risk that the helicopter will start to rebound on the ground and could be destroyed by this violent oscillation.

An example of vibration amplification is a diver on a diving board. His interest is to amplify the movement of the board by synchronization of his jump. (*Figure 3-11*)

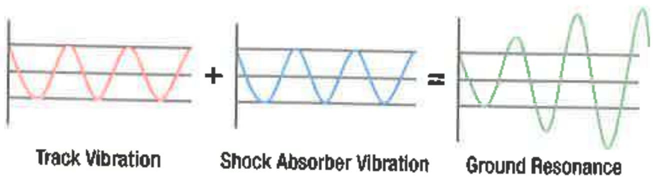


Figure 3-10. Ground resonance representation.

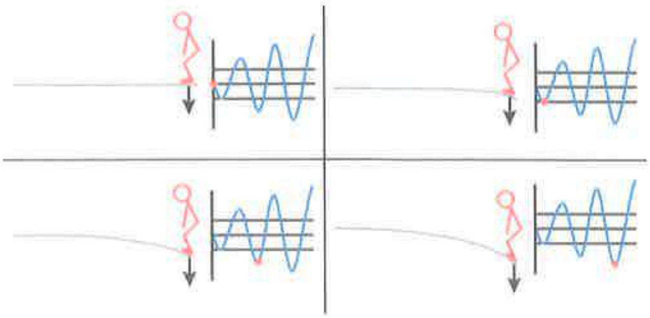


Figure 3-11. Amplification of the movement by synchronization.

If the diver pushes down the board at its lower position, he will amplify the movement and increase the amplitude to store more power hence increasing his jump.

The solution to stop the resonance is to desynchronize the vibrations therefore stopping the amplification effect. The solution in the case of a helicopter ground resonance, is either to take off, hence stopping the effect on the shock absorber, or to reduce the rotor speed to desynchronize the vibrations.

VIBRATION REDUCTION METHODS

AMPLITUDE REDUCTION

The simplest method to reduce a vibration is to reduce its amplitude. Often, if it is not possible to completely remove the vibration, it can be possible to greatly reduce its effects by reducing the amplitude. On a helicopter, most vibrations are imbalance vibrations. To reduce its amplitude, an adjustment of the mass of the aberrant component must be made. The goal is to move the center of gravity of the spinning object to align with the center of rotation or center of tilt.

The first step is to measure and define the center of gravity. When the mechanic understands where the imbalance or excess mass is, there are two possibilities to modify the weight and so balance the system:

- Remove some mass in the same direction of the excess of mass. (***Figure 3-12 and Figure 3-13***)
- Add some mass in the opposite direction of the excess of mass. (***Figure 3-14 and Figure 3-15***)

VIBRATION OPPOSITION

When it is not possible to act directly on the system, an opposite vibration can be generated to delete the effect of the existing vibration. (***Figure 3-16***)

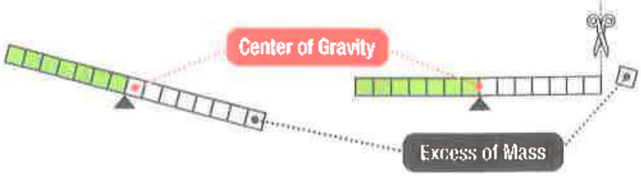


Figure 3-12. Balancing by mass removing.

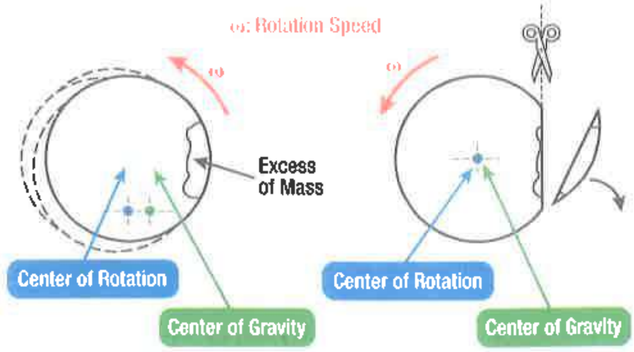


Figure 3-13. Rotating balancing by mass removing.

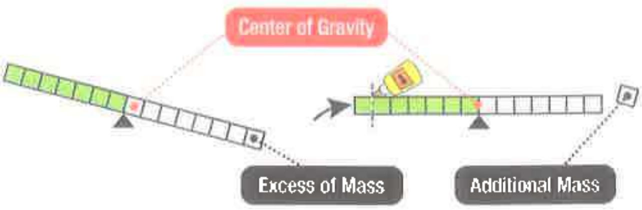


Figure 3-14. Balancing by mass addition.

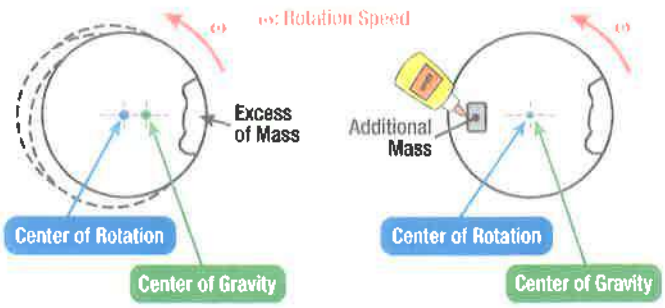


Figure 3-15. Rotating balancing by mass addition.

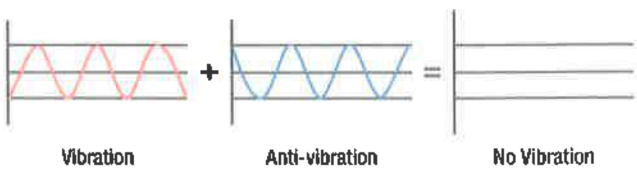


Figure 3-16. Absorption of vibrations by anti-vibration.

If two vibrations have the same harmonic, they tend to start a resonance; but if they have the same frequency, same amplitude, and the opposite phase, they tend to cancel each other.

An active method consists of injecting an effort into the system, which comes to oppose the disturbing effort, and so limiting or canceling its effect on the system. This method requires the installation in the system of additional actions and sensors and of a computer. It is possible to symbolize this system by a mass which moves from top to bottom. This movement is generated by a complex system capable of controlling the frequency and amplitude of this movement to generate a specific vibration shown in green in ***Figure 3-17***. This device is called a resonator.

The goal is to artificially create the vibration exactly opposite to that felt in the helicopter, effectively eliminating it and so having the feeling of no vibration. The resonator should be connected at a point where the two vibrations intersect. (***Figure 3-18***)

This has the main drawback of making the system more complex. This system applies particularly to helicopters which are subject to permanent vibrations. One of the major drawbacks, comes from the injection of an effort, which in case of the loss of control of the system can have a destabilizing effect. To avoid any damage, it is therefore necessary to balance and adjust all the rotating systems.

STATIC AND DYNAMIC BALANCING

Now that you know the basics about vibration, we will study it on the helicopter. Where do these vibrations come from?

Vibrations can come from all rotating parts including engines, gearboxes, shafts, and rotors. It is not possible to adjust the motor and the gears in their different housings to suppress vibrations. That is the job of the manufacturer. If we find some vibration coming from these components, the only solution for the maintenance technician is to check the bearings to confirm whether or not there is a problem with them. Otherwise, the only way is to completely change the gearbox. For shafts it is the same problem as with engines and gearboxes. We are not authorized by the manufacturer to modify them.

The only access we have to reduce vibrations from the power system is through the different rotors. Towards that, we have two stages for balancing:

- Static Balancing
- Dynamic Balancing

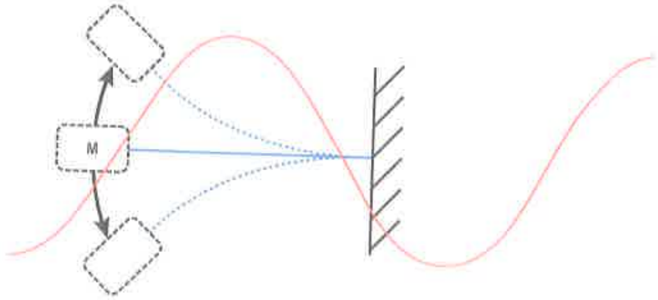


Figure 3-17. Resonator.

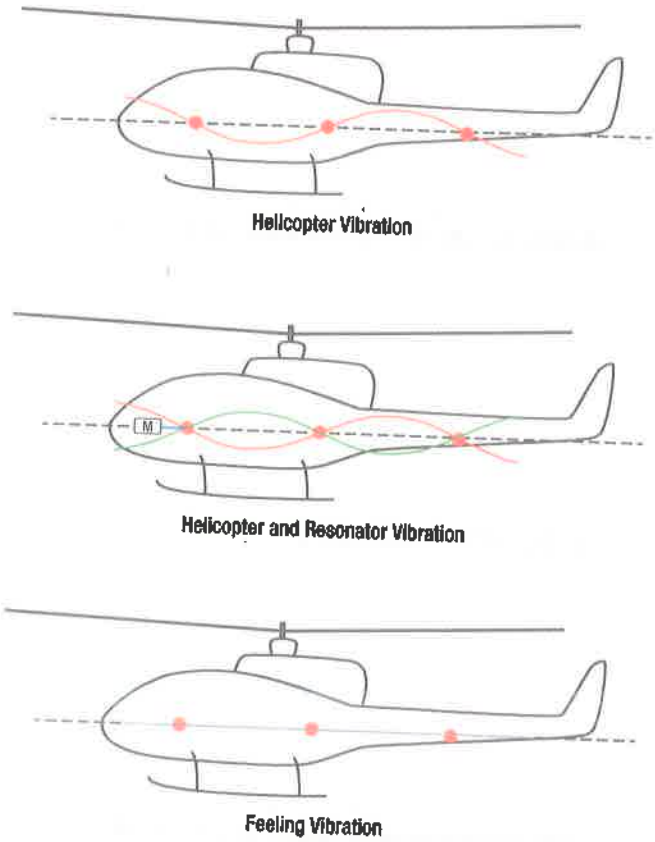


Figure 3-18. Vibration annihilation.

STATIC BALANCING

We have seen that the origin of vibrations is a weight imbalance. For this reason, we begin by explaining how to adjust it on a main or rear rotor blade. The purpose of this balancing is to have the center of gravity in the correct position. For this, it is not necessary to rotate the blade. Certain adjustable masses (weights) are located at the end of the blade which allow for the weights to be added, removed, or moved. (***Figure 3-19 and Figure 3-20***)

All the blades of the same rotor must have the same static moment to have the same reaction when they turn, but what is a moment? (***Figure 3-21***)

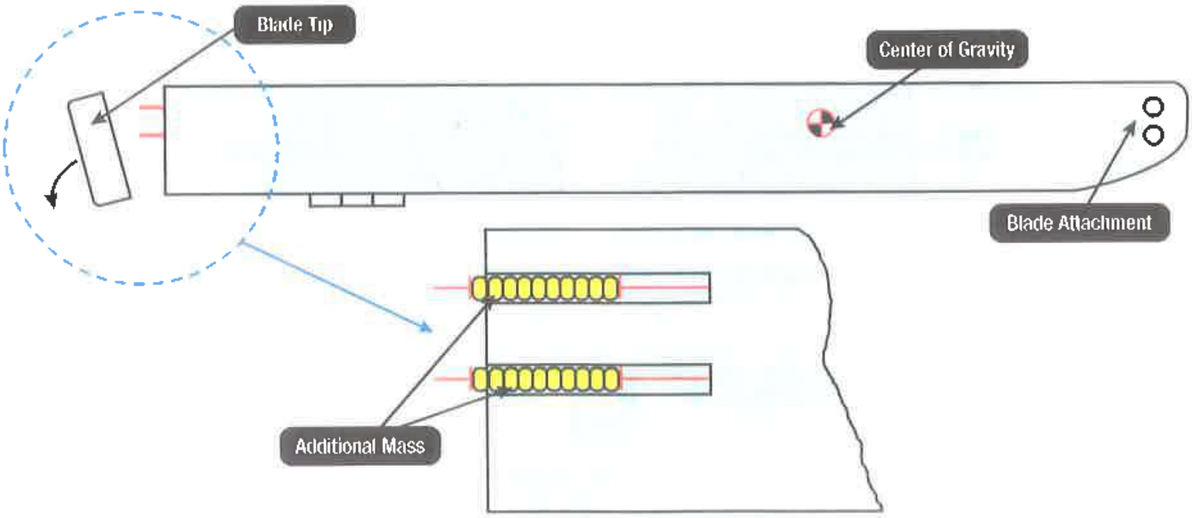


Figure 3-19. Blade Additional mass.

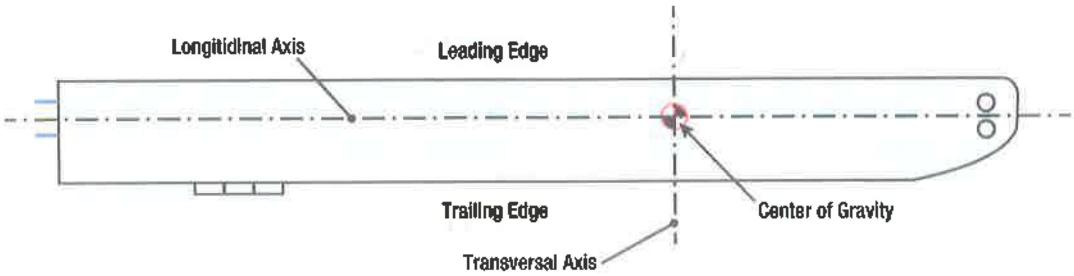


Figure 3-20. Balancing axis.

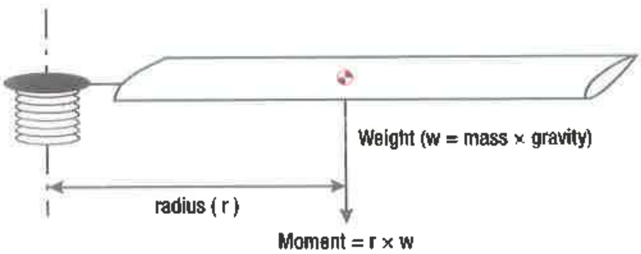


Figure 3-21. Static moment of a blade.

The static moment is the multiplication of the weight by the radius (distance between the axis of rotation and the center of gravity). The problem is that despite the manufacturer's effort to build similar blades, it is not possible to have exactly the same weight for each and it is also difficult to have the center of gravity at the same position on each blade. The direct consequence is that if these values are not the same, the moment cannot be the same on each blade and vibrations will appear.

To move the position of the center of gravity along the longitudinal axis, it is easy to understand that it is sufficient to add or remove mass at the end. In fact, if we add a mass to the two supports, the center of gravity will shift towards the tip of the blade. (***Figure 3-22***)

On the other hand, if we remove a certain mass on the supports, the center of gravity will shift towards the attachment of the blade. (***Figure 3-23***)

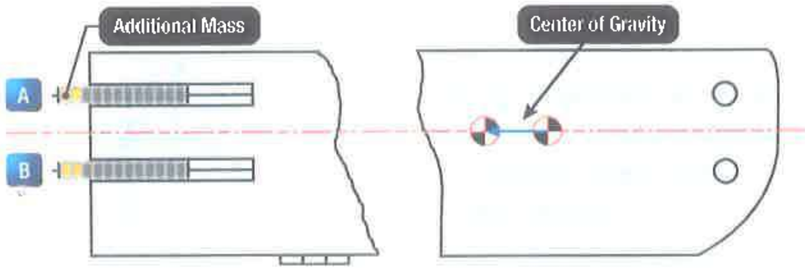


Figure 3-22. Center of gravity to the blade tip.

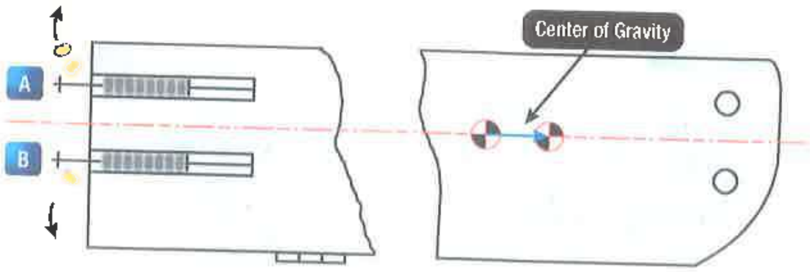


Figure 3-23. Center of gravity to the blade attachment.

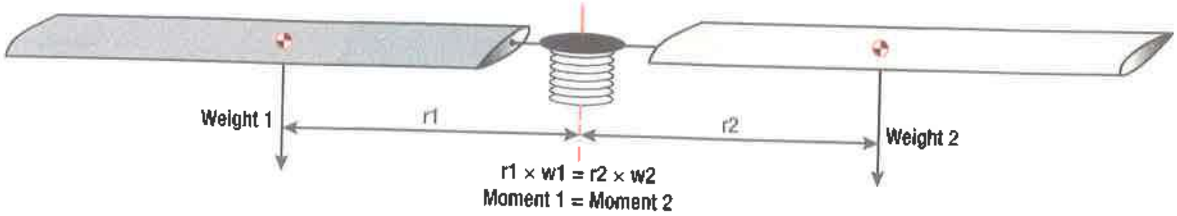


Figure 3-24. Moment equality.

Each blade can have a different weight and radius (not the same mass or position of center of gravity). The important thing is that the moment must be the same for all the blades of the same rotor. In this condition, all blades will theoretically have the same reaction. (*Figure 3-24*)

DYNAMIC BALANCING

Once the moment is known to be the same on each blade of the rotor there is a new problem. We know the center of gravity is in the right position on the longitudinal axis but may not be right on the transverse axis (*Figure 3-25*) and there are consequences if it is not.

Stresses can appear in the event of misalignment of the helicopter's weight and lift. If there is a distance (d) (*Figure 3-25*), the blade will tend to twist on itself. It is important to have the same load on all the blades of the same rotor.

The greater the distance (d), the more the pitch tends to increase and the faster the blade tends to rise. If this reaction is not calibrated to be the same on all the blades at the same time, the lift of each blade will not be the same as the others and vibrations will appear.

While it is not possible to move the position of the lift, it is possible to shift the weight to the leading or trailing edge. To adjust the static balancing, it was important to add or remove the same mass on each support, but to move the weight and adjust "d" we will add or remove mass only on one of the two blade supports.

To move the weight to the leading edge, we will add mass on support A or remove mass on support B. With this movement, the distance "d" will decrease. (*Figure 3-26*)

To move the weight to the trailing edge, we will add mass on support B or remove mass on support A. With this movement, the distance "d" will increase. (*Figure 3-27*)

The purpose of this adjustment is not to eliminate the distance (d) but to have it within an acceptable tolerance and identical on all blades of the same rotor. The reaction of the blade will only be confirmed in rotation and that is why we call it dynamic balancing.

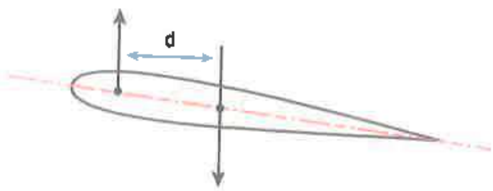


Figure 3-25. Moment of torsion.

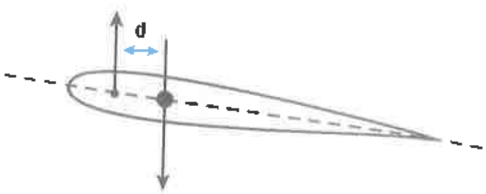


Figure 3-26. Weight to the leading edge.

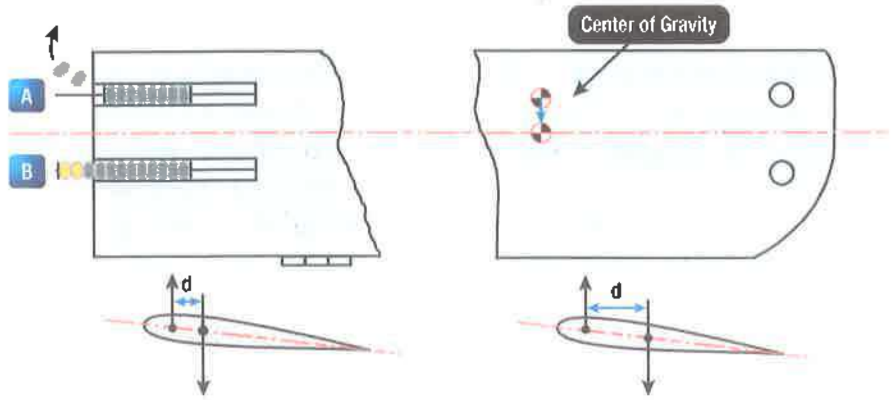


Figure 3-27. Weight to the trailing edge.

When this dynamic balancing is performed, normally the blades fly the same way but, in reality, it is still possible that they have different reactions if variations in aerodynamics appear. One solution is to adjust another dynamic balancing factor known as aerodynamic balancing. Small fixed flaps called Tabs (***Figure 3-28***) are connected on the trailing edge and can be adjusted (bent) to modify the aerodynamic signature to vary the lift, and so making it go up or down.

If the tab is bent upwards (***Figure 3-29***), the blade tends to rise due to the air flow which creates pressure on the top of the tab.

If the tab is bent down (***Figure 3-30***), the blade tends to nose down due to the air flow that creates pressure on the bottom of the tab.

Initially, all these basic balances are carried out by the manufacturer and we receive the blades adjusted with the right weight and the right position of the center of gravity.

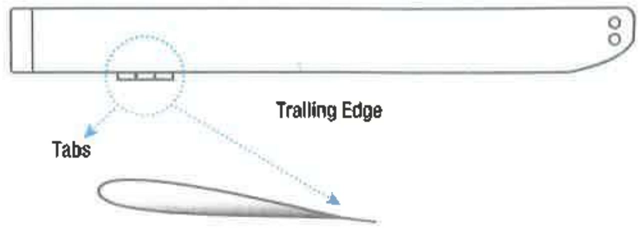


Figure 3-28. Blade tab.

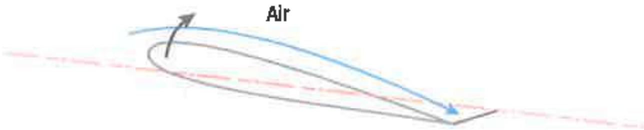


Figure 3-29. Tab up.

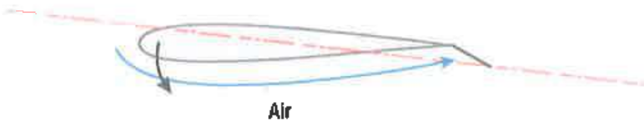


Figure 3-30. Tab down.

ROTOR ALIGNMENT

Despite the efforts and precision work of the manufacturer to provide similar blades, they do not always have the same reaction in flight. Due to minor aerodynamic variations, some fly well and others not as well. The consequence is that for the same pitch and the same rotation speed, all the blades of the same rotor do not fly in the same plane. This generates vibrations called "track". To reduce or eliminate this vibration, it is necessary to check the position of each blade during the rotation to see which one flies upwards and which flies downwards.

MAIN ROTOR AND TAIL ROTOR TRACKING

An old system of rotor tracking is to define a color for each blade and with a flag, observe the path of each. (***Figure 3-31 and Figure 3-32***) At the end of the blade tip is a touch system on which we add colors. When this touch system hit the flag, it drew the different colors on it representing the height of each blade. (***Figure 3-33***)

It became easy to analyze the flight of each blade. For this example, we will adjust to the blade fly in the midplane, the blue. This means that we are going to increase the size of the yellow pitch change rod to make it fly higher. For the red blade, we will reduce the size of the pitch change rod to reduce the angle of attack and reduce its lift. Another test should be performed to verify if the adjustments are sufficient. If so, we will get a mix of colors and in this perfect case, (***Figure 3-34***), the "track" vibration is suppressed.

We continue to use this system for basic adjustments but if vibrations appear in flight it will not be possible to control the position of the blades in the air. This is the reason why a new generation of tools has been created based on a strobe and reflective targets. With these, it is easy to see the track of each blade during rotation. First it is necessary to install reflective targets under the tip of the blade in the direction of the cockpit. Each target has a different symbol on them to recognize the different blades. For example: a triangle, a square, a circle and if we have 4 blades, a diamond. When the rotor is turning, it is not possible to see the targets due to the rotation speed, but if we synchronize the strobe flash with the rotor RPM, the targets appear under the rotor disk. As said previously, the main advantage is the possibility of measuring the gap between each blade during a real flight. The mechanic is in the cockpit during the flight and to check the position of the blades, he/she must only aim the strobe under the rotor disc to see the targets appear. After measuring, he/she can see this example: (***Figure 3-35***)

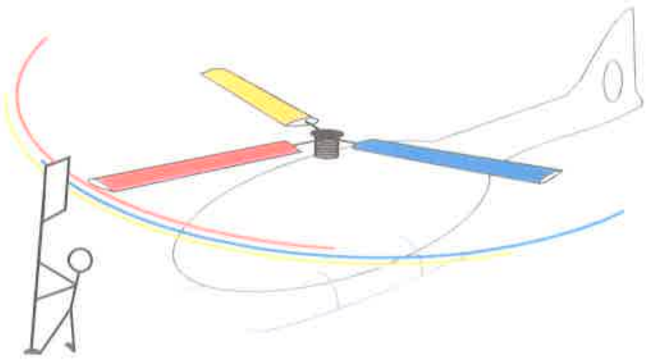


Figure 3-31. Track check 1.

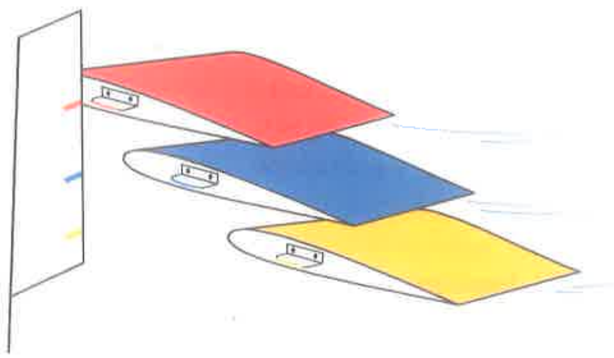


Figure 3-33. Blade touch system.

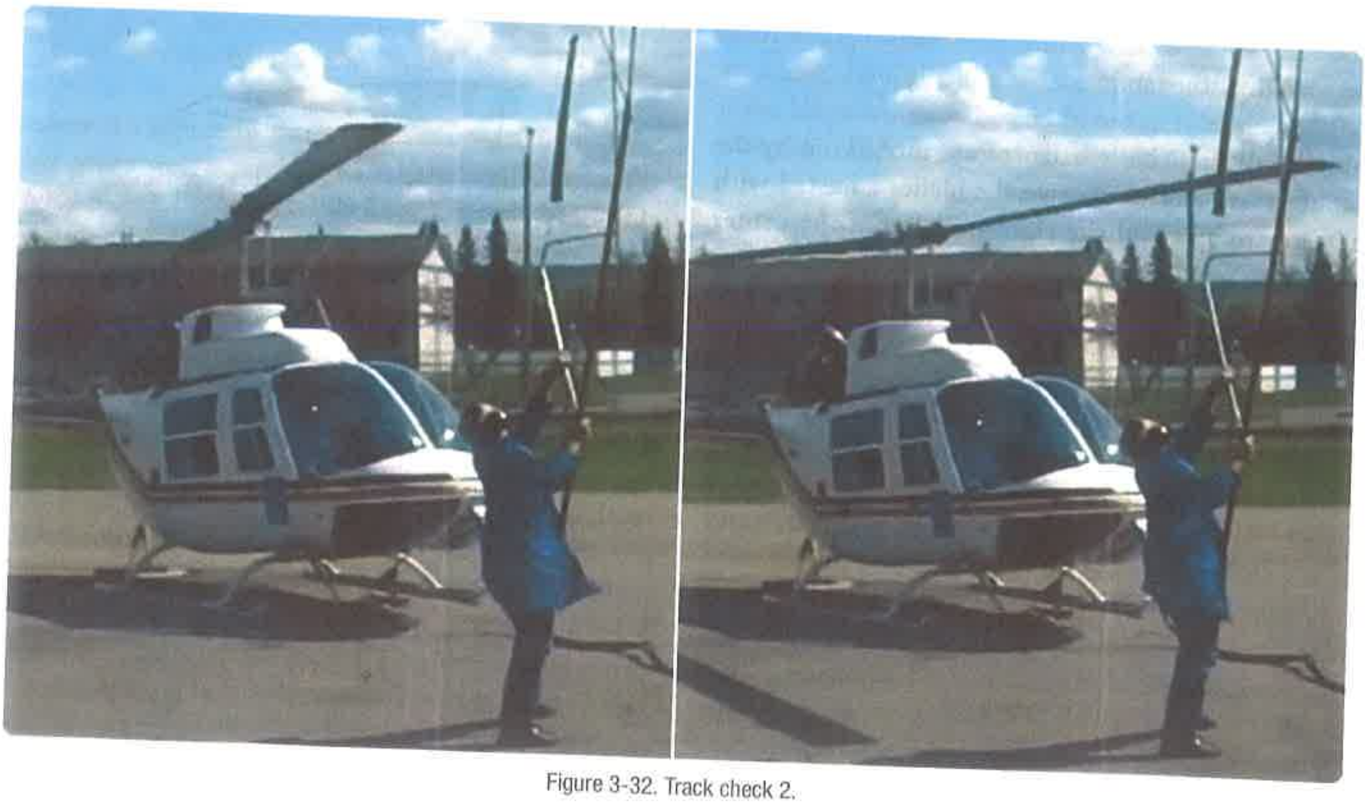


Figure 3-32. Track check 2.

Knowing this, the same adjustment can be made as with the flag. Increase the angle of attack of the yellow blade and decrease the angle of attack of the red one. When all are acceptable, perhaps after several tries, the final result is the appearance of a mix of the symbols under the rotor disc. (***Figure 3-36***)

These techniques are only necessary on the main rotor. Because the tail rotor is smaller, and the flapping hinges are free, these stresses are not great, and it is not necessary to adjust the track on the rear rotor. The choice of the manufacturer is to leave the tail blades free, allowing the blades to fly in an average plane of rotation without transmitting vibrations to the structure. In the case of the Fenestron, air only passes through the tail rotor in a perpendicular direction to the blade. The fact that there is no flapping hinge in a Fenestron creates an impossibility for the blades to fly in different planes of rotation, which is why the tracking vibration on a Fenestron does not exist.

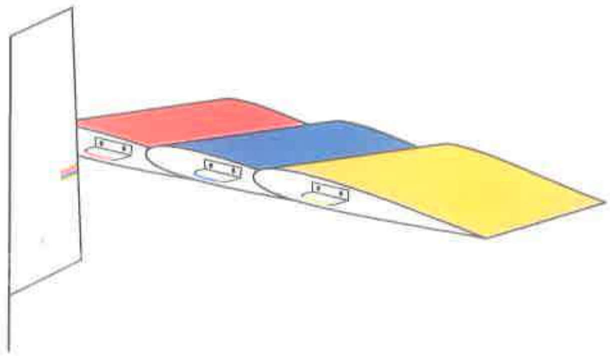


Figure 3-34. Blade alignment.

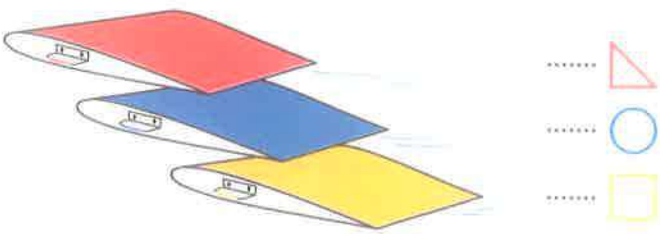


Figure 3-35. Blade target system with strobe.

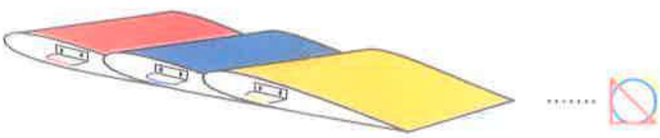


Figure 3-36. Blade alignment with target.

A common vibration between the main rotor and the tail rotor is an "imbalance". We have described this vibration before and it is necessary to explain how to correct it on the rotor of a helicopter.

After installing an accelerometer in the same plane as the rotor we want to check, (vertical for the tail rotor and horizontal for the main rotor), we will measure the vibrations in flight.

The accelerometer measures the displacement on the longitudinal axis of the helicopter to indicate the wave of the vibration. If an imbalance appears during the rotation of the rotor, it will measure the movement and provide this information to a computer which will analyze the vibration and automatically find the solution to solve the problem. There are two parts to the solution:

1. What is the mass to add opposite to the imbalance?
2. Where is the imbalance?

The amplitude of the vibration provides this information. The greater the amplitude of the vibration, the more additional mass is needed. If we have a small amplitude it means the rotor is almost balanced and maybe within tolerances. A second measurement is needed to know where to apply the additional mass. To obtain this information, it is necessary to install a position sensor between a fixed part (the fixed swashplate) and a rotating part (the rotary swashplate). This measurement will give us the start point and the position in this rotation of the maximum amplitude. (***Figure 3-37***)

On this wave we can see that the maximum amplitude is at the front when the red blade is at the front and maximum at the rear when the red blade is at the rear. The conclusion is that excess mass is on this blade. The computer analyzes the information given and provides a value; (100 gr for example). It is not possible to remove

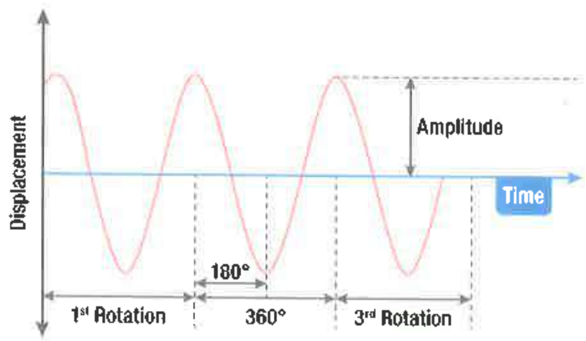


Figure 3-37. Vibration analysis.

weight from the main rotor head, but it is easy to add to it. In the case of a four bladed main rotor head this is not difficult but in the case of a three bladed head there is no support to install the additional mass on it. (***Figure 3-38***)

The solution given is to divide the weight into two equal parts in this case and distribute them over two blades (the blue and the yellow). If the computer orders to add 100 gr at 208° from the start of rotation, the work will be to add 73 gr in the yellow blade and 27 gr in the blue one; (***Figure 3-39***) this being a prescribed proportional distribution.

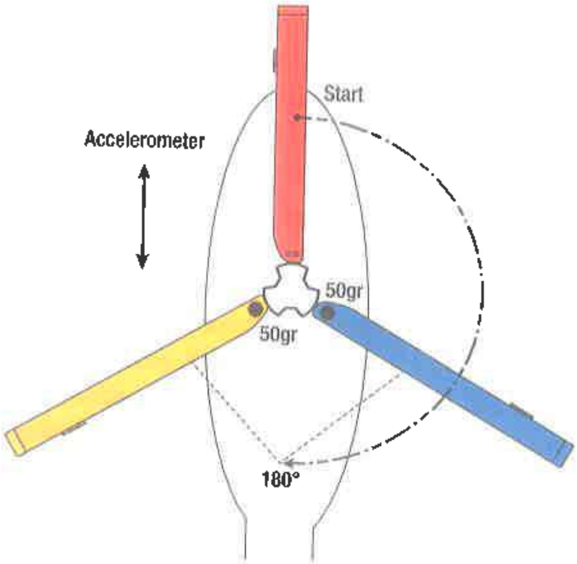


Figure 3-38. Weight repartition 180°.

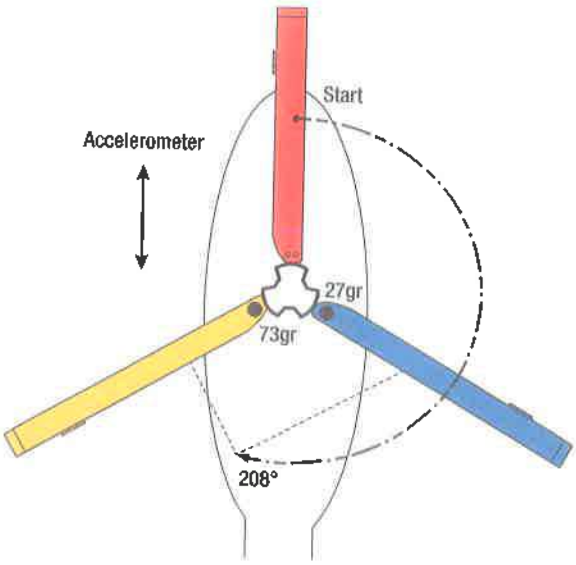


Figure 3-39. Weight repartition 208°.

Question: 3-1
What two components define a vibration?

Question: 3-2
What causes the type of vibration known as ground resonance?

Question: 3-3
What is the primary cause of vibrations of the rotor?

Question: 3-4
What is the most effective way to reduce vibration from a spinning object?

Question: 3-5
Where is the most likely defect within an engine or gears that would cause a vibration?

Question: 3-6
Which are the only two factors which must be considered when conducting static balancing?

Question: 3-7
For which component on a helicopter is it important to determine dynamic balance?

Question: 3-8
What is the primary cause of tracking errors on a helicopter rotor?

ANSWERS

Answer: 3-1
Amplitude; the size of an unwanted movement, and frequency; the number of times per second, the full unwanted movement occurs.

Answer: 3-2
A combination of a mechanical vibration combined with its resonance on the shock absorbers of the landing gear.

Answer: 3-3
One or more blades not turning in the same plane.

Answer: 3-4
To better align its center of gravity with its center of rotation.

Answer: 3-5
Its bearings.

Answer: 3-6
The object's weight and the distance of that imbalance from the center of rotation.

Answer: 3-7
The rotor blades.

Answer: 3-8
One blade creating more lift than the other due to varying angles of attack or variations in each blades airfoil shape.

HELICOPTER AERODYNAMICS, STRUCTURES AND SYSTEMS

TRANSMISSION

SUB-MODULE 04

PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY – B1.3 B1.4

Sub-Module 04

TRANSMISSION

Knowledge Requirements

12.4 - Transmission	
Gear boxes, main and tail rotors; Clutches, free wheel units and rotor brake; Tail rotor drive shafts, flexible couplings, bearings, vibration dampers and bearing hangers.	3

12.4 - TRANSMISSION

INTRODUCTION: POWER TRANSMISSION

It is easy to make an airplane fly: two wings and an engine. It creates power that is directly transformed into thrust. However, with a helicopter, that power is not thrust but torque to turn the various rotors (main and rear). The solution to transmit that torque from the engine to the rotors is to use shafts and gearboxes. Shafts lead to connecting boxes and boxes must then change the direction of that power and reduce or increase its speed of rotation. While it is possible to find accessories on the transmission system, its main components are the main rotor transmission, tail rotor drive system, clutch, and freewheeling unit. Multiple boxes are needed, each having a name and function depending on their position in the power transmission kinematic. Their names and locations are described in ***Figure 4-1***.

GEAR BOXES, MAIN AND TAIL ROTORS

GEAR BOXES

As the engine turns faster than what is possible for the rotor, it is important to reduce the speed of the rotor's rotation. Additionally, a mechanical law states that if you want to increase torque, you must reduce the speed of rotation. This is the same with cars. You have a lot of torque when the transmission is in first gear, but the car cannot move fast. To run fast you need to be in 5th gear. However, in 5th gear it is difficult to start because of the low torque.

On a helicopter, the relationship between the rotational speed of the main rotor and the rotational speed of the engine is important. For example, on a Gazelle helicopter, the turbine engine turns at 44 000 RPM and the main rotor turns at 387 RPM. That result requires a 99% reduction. For the smaller tail rotor it is different as the torque needed is less, but the speed greater; about 6 000 RPM. In addition to adjusting the speed of rotation, a directional change must also occur based on the orientation of the engine as compared to the plane of rotation of each rotor. A helicopter's gear boxes must accomplish both of these functions.

MAIN AND TAIL ROTORS

The main gear box (***Figure 4-2***) is the primary component of the transmission system, it receives the power from the engine at its input coupling and must distribute it in two directions. Many gears and bearings inside the system are needed to change the direction and speed of rotation. We can see in ***Figure 4-3***, that the speed of rotation is maximum at the input, average towards the tail rotor and minimum at the main rotor.

This can be complex, depending on the number of engines on a helicopter. With two engines connected to the main gearbox, separate power sources must be simultaneously transmitted to the rotors. The different stages in the gearbox gradually reduce the speed. In the first stage: if the engine is horizontal, the Main Gear Box (MGB) must redirect the power vertically to turn the main rotor. In this case, within the input coupling, a bevel gear changes the direction of the movement. If the two gears are the same size, it is just an angle change, but if the driven gear is larger than the driving one, the speed is reduced. (***Figure 4-4***)

At the second reduction stage, the manufacturer develops methods to achieve a greater speed reduction, requiring

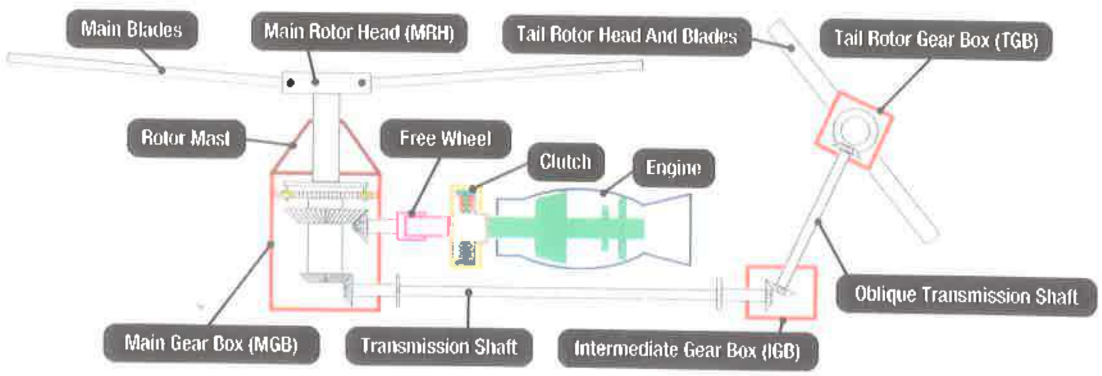


Figure 4-1. Power kinetic.

Figure 4-2. Main gear box.

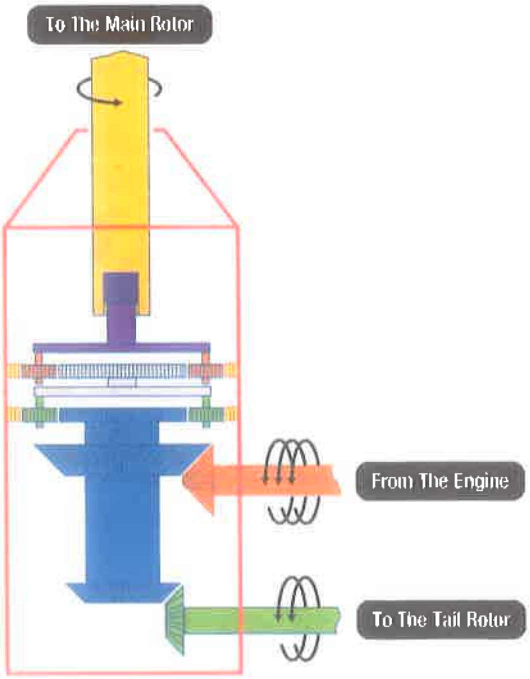


Figure 4-3. Main gear box.

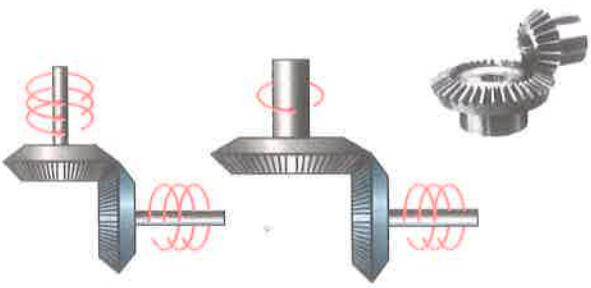


Figure 4-4. Speed reduction.

both small sized components and good reliability. The most widely used system to achieve this reduction and accept great amounts of torque is the epicyclic module. (***Figure 4-5 and Figure 4-6***) This configuration of the planetary stage equalizes the loads between the planets within the gear case, by means of:

- Self-centering floating planetary gear.
- Planetary gears with flexible support and self aligning spherical roller bearings.
- Fixed crowns.

The three planet pinions are driven by the sun gear attached to the bevel gear of the MGB. (***Figure 4-7***) The planet pinions rotate inside the fixed ring gear and drive the carrier which is connected to the second stage reduction of the main gearbox. The result is a reduction of the rotation speed as the sun gear turns faster than the planet carrier.

To continue the reduction, a second stage is installed. The planet becomes the sun (in grey) and a new planet carrier (in purple) spins slower than the new sun. The 3 planet pinions (in orange) are driven by the sun gear attached to the 2nd stage reduction of the MGB. The

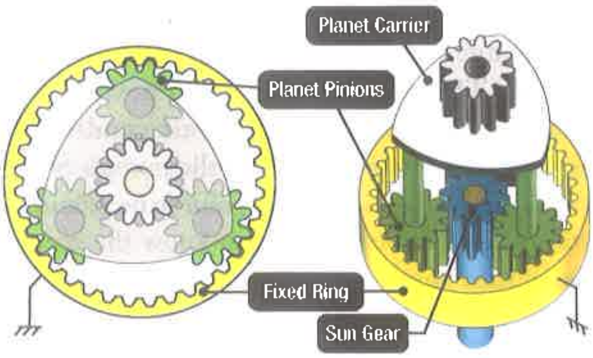


Figure 4-5. Epicyclic module.

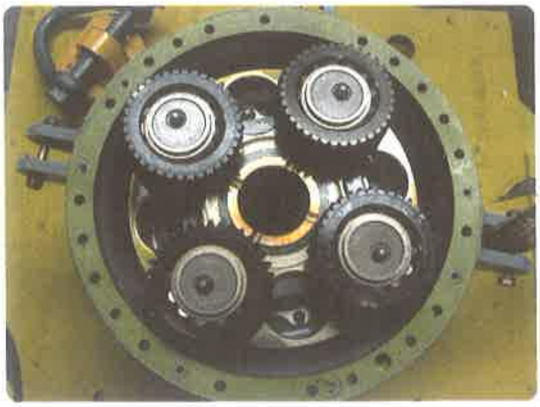


Figure 4-6. Epicyclic module (1 stage).

planet pinions rotate inside the fixed ring gear (in yellow) and drive the planet carrier and the main rotor mast in rotation. It is also possible to add stages as needed until you get the correct reduction. At the end, the planet carrier is connected to the rotor mast and gives the rotation speed of the blades. (***Figure 4-8***)

The rotor mast exits the main gearbox through a tapered cover (***Figure 4-9***) which supports two bevel bearings. The purpose of this cover is to close the main gearbox but more importantly to transmit torque to the main rotor. The function of the two bearings is to accept vertical stress, due to the weight of the main rotor when the helicopter is on the ground, and due to the weight of the structure hanging under the main rotor when the helicopter is in flight. This is where the torque reaction that we saw in *Sub-Module 01* appears.

The main gearbox separates the motion from the engine(s) in two ways. One being upward toward the main rotor head and the other down and rearwards toward the tail rotor. The tail rotor reduction is less than for the main rotor and so an epicyclic system is not required. The size difference between the gears of the lower bevel reduction is sufficient. (***Figure 4-10***)

A third connection in the MGB is for the use of accessories. This provides rotational power for the electrical generators, alternators, air conditioning compressors, oil pumps, and hydraulic pumps. Sensors allow the speed of rotation at this point to be monitored, and with a calculation it is easy to know the speed of rotation of the final systems.

The tail rotor drive system consists of drive shafts. As we saw in ***Figure 4-1***, two other boxes are necessary to turn the tail rotor which is a far distance from the MGB.

The Intermediate Gear Box (IGB) typically provides only a direction change between the horizontal tail and the pylon. It is possible to change the speed of rotation if necessary, by adjusting the size of the gears, but the ratio is not greater than 1.5:1. Two flanges are used to connect the horizontal shaft to the input of the IGB, and the oblique shaft to its output.

In ***Figure 4-11***, we can see two bevel gears of the same size but not in the same axis of rotation. Thus the speed stays the same but there is a change in direction.

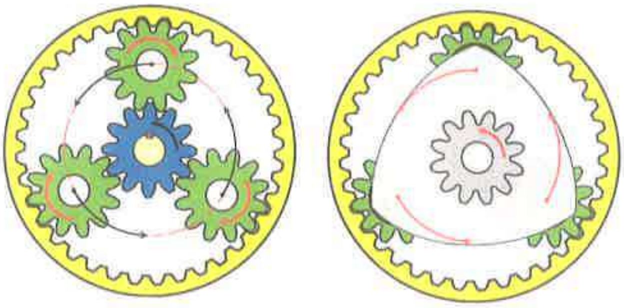


Figure 4-7. Reduction movement.

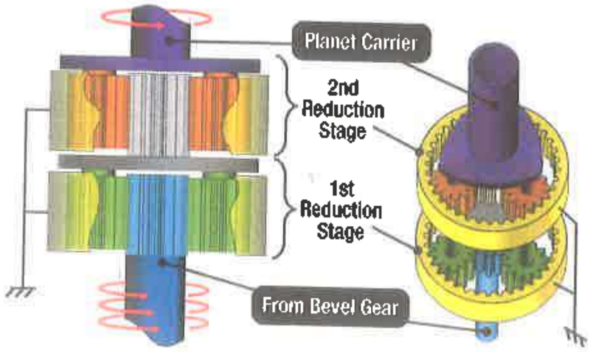


Figure 4-8. Epicyclic module.

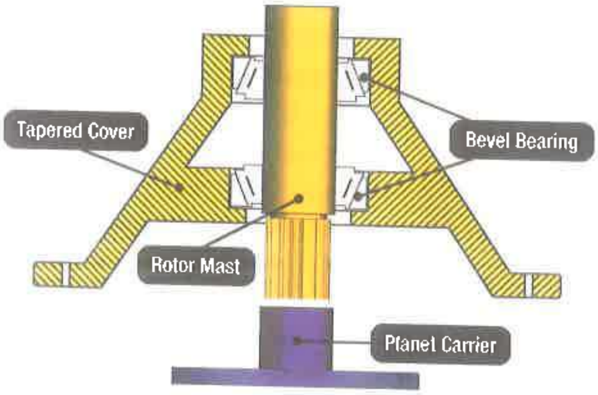


Figure 4-9. Tapered cover.

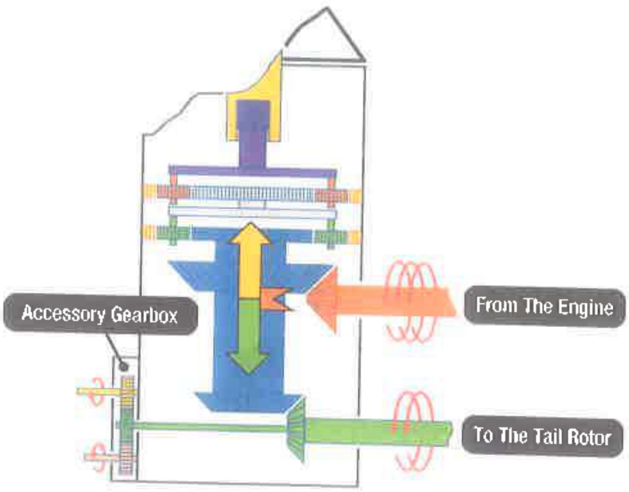


Figure 4-10. Accessory and tail connection.

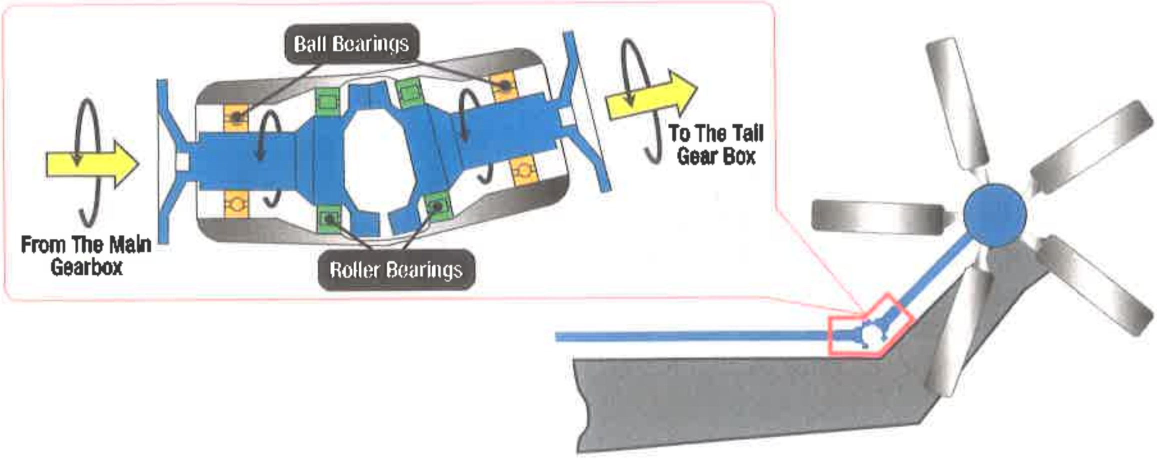


Figure 4-11. Intermediate gear box.

The last box of the powertrain is the Tail Gear Box (TGB). This is the final speed reduction of the tail rotor. Its main function is to change the drive direction by means of two bevel gears at right angles. (*Figure 4-12*)

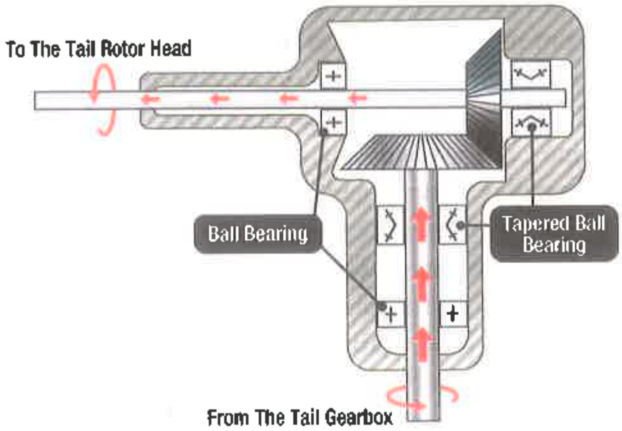


Figure 4-12. Tail gear box.

Bearings allow the various shafts to rotate smoothly. Two double angled ball bearings oppose the axial stresses generated by the tail rotor during yaw movements.

CLUTCHES, FREE WHEEL UNITS AND ROTOR BRAKES

CLUTCHES

Unlike free turbine engines on which the compressor is mechanically independent, a clutch is required with a linked turbine engine. The clutch makes it possible to start the engine without driving the rest of the powertrain by limiting the torque (and limiting overheating). When the engine starts, the shaft connected to the engine rotates. The centrifugal force compresses the spring and moves the pads outward. Some friction is encountered and the shaft connected to the main gearbox begins to rotate. (*Figure 4-13*)

As the engine speed continues to increase, the centrifugal force permanently connects the pads to the outer shaft

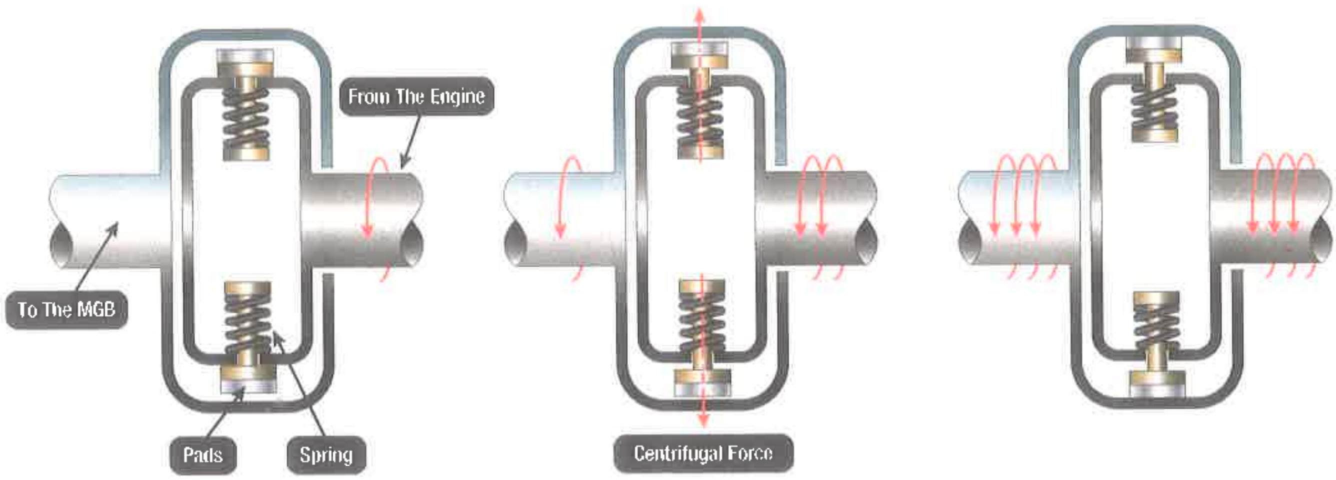


Figure 4-13. Clutch.

with a device called a ferrodo which is a carbon based part like a car brake. At this moment, the inner and outer shafts are turning at the same rotational speed.

When you reduce the engine speed, the clutch springs overcome the centrifugal force and the pads return to their original position. The motor is then disconnected from the main gear box. You can then stop the rotation of the rotors with the rotor brake if you have one. Otherwise, you must wait for the rotor to stop before getting out of the helicopter.

FREE WHEEL UNITS

For the safety of flight, the power connection between the engine(s) and MGB, must include a freewheel device. In the event of an engine failure, it is necessary to separate the engine from the rotors to let the rotors continue to spin in autorotation for landing. The freewheel on a helicopter has two working positions. When all is well, the freewheel is engaged and transmits power from the generator to the receiver like a simple drive shaft. The shaft connected to the motor (***Figure 4-14***) turns and pushes with the help of springs. The rollers on the outside lock them against the shaft connected to the MGB (in brown). The direct consequence is the rotation of the rotor.

In the event of an engine failure, the rotor must continue to rotate. When the receiver spins faster than the generator, the freewheel disengages and separates the two movements. The outer shaft turns and pushes the rollers in the inner direction by compressing the spring. At this time, the internal shaft is disconnected and does not disturb the movement of the rotor. (***Figure 4-15***)

On some twin engine helicopters, it is possible to manually disconnect one of the two freewheels. This option makes it possible to run the corresponding engine for hydraulic or electric generation without rotor movement. If the freewheel is not an option, it is possible to insert a clutch and rotor brake into the powertrain.

ROTOR BRAKES

The purpose of the rotor brake is to reduce the rotor rotation time when the pilot shuts down the engine. Another function is in the case of wind. The risk when you start the rotor in windy conditions is that the rotor may hit the helicopter structure. Typically at the start of rotation, the rotor tends to drop due to its weight and

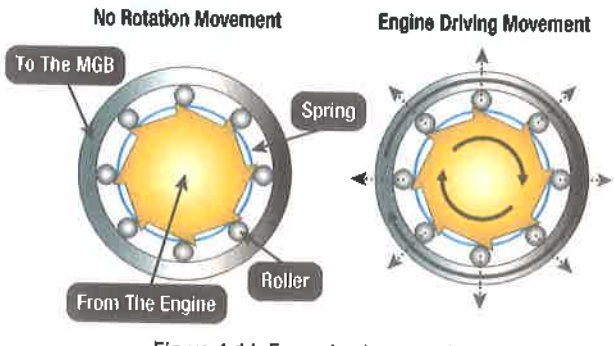


Figure 4-14. Free wheel engaged.

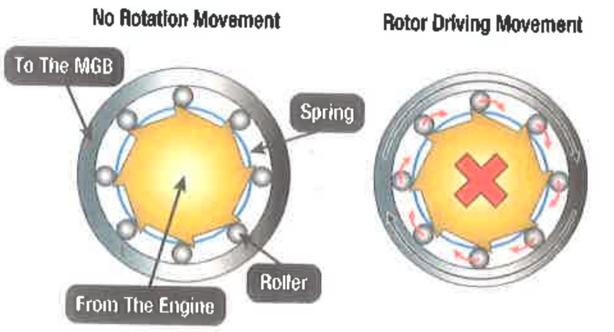


Figure 4-15. Free wheel disengaged.

insufficient lift. The advantage of the brake is to stop the rotational movement and release it after the clutch is engaged. In this situation, the rotor will quickly achieve a high enough speed to produce enough lift to spin away from the structure. The brake is the same as for a car. A disc connected to the power transmission shaft at the input of the MGB will be controlled in rotational speed by a friction pad that the pilot moves with a handle in the cockpit. When the pad touches the disc, it reduces the speed of rotation until the rotor stops. When the pilot releases the cockpit handle, the return spring separates the two pads from the disc and leaves it free. (***Figure 4-16***)

TAIL ROTOR DRIVE SHAFTS, FLEXIBLE COUPLINGS, BEARINGS, VIBRATION DAMPENERS AND BEARING HANGERS

TAIL ROTOR DRIVE SHAFTS

To connect the main gearbox to the intermediate gearbox and finally to the rear gearbox, we use drive shafts which are simple aluminum tubes. (***Figure 4-17***) On each end, an attached flange allows the connection to a gearbox or to another shaft.

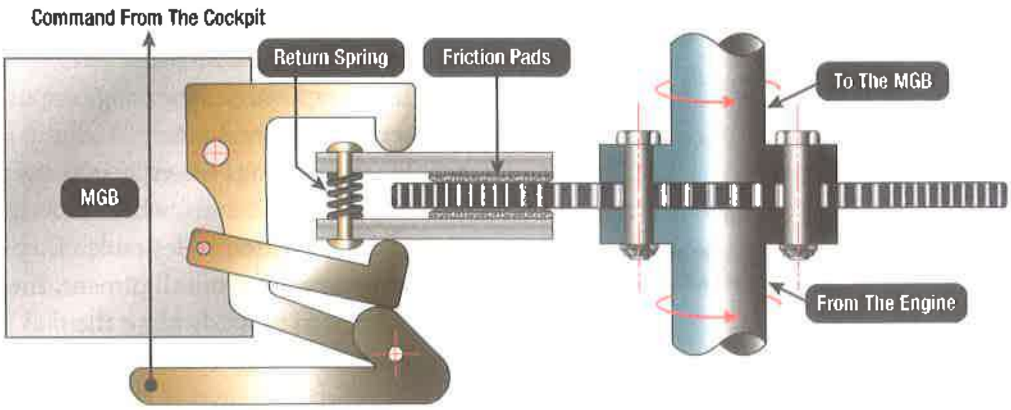


Figure 4-16. Rotor brake.

On top of the main gearbox, a single mast supports the main rotor head and its blades. A greater problem is in its connection to the tail rotor due to the flexibility of the tail boom. On landing, the tail boom is bent down with the weight of the tail rotor. In flight, it is bent right to left according to the yaw direction control. To withstand misalignments and to avoid damaging the inlet seal of each box, the design has two options. The first is to install a long, thin and flexible shaft with three or four bearings to support it on the rear beam. This solution is used on light helicopters due to the low torque sent to the tail rotor. (***Figure 4-18***) The second is by use of flexible couplings.

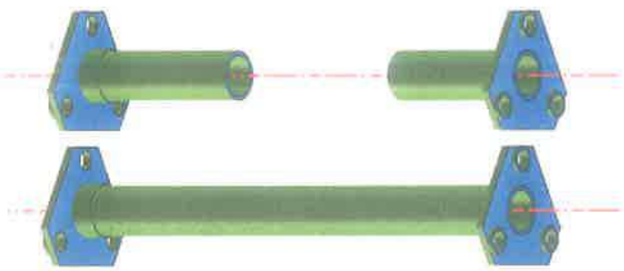


Figure 4-17. Shaft.

FLEXIBLE COUPLINGS

On a large helicopter. The distance between the MGB and the IGB is exceptionally long and the torque more of a factor. In this condition, it is necessary to reduce the length and increase the diameter of each portion of the shaft. To connect each portion flexible couplings are used to allow for misalignment due to the deformation of the rear beam. (***Figure 4-19***) These couplings are connected to the structure with bearing brackets to guide the rotating shaft.

The deformation of the tail creates an axial movement of the different shafts, due to varying distances between the two connected points such as the MGB and the IGB. The function of flexible discs is to accept the deformation due to these variations. When a stress appears on the shaft, an axial movement warps the flexible disc. This movement is allowed by the curved part of the washer under the head of the Jo-bolt rivet. (***Figure 4-20***) This protection of the shafts has the advantage of reducing maintenance costs, as it is less expensive to change flexible discs rather than structural shafts.

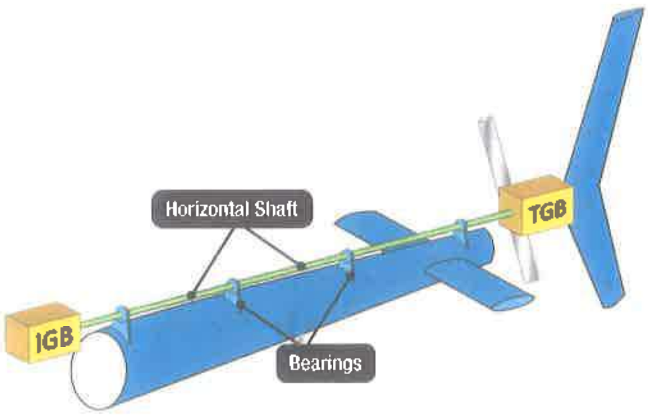


Figure 4-18. Light helicopter horizontal shaft.

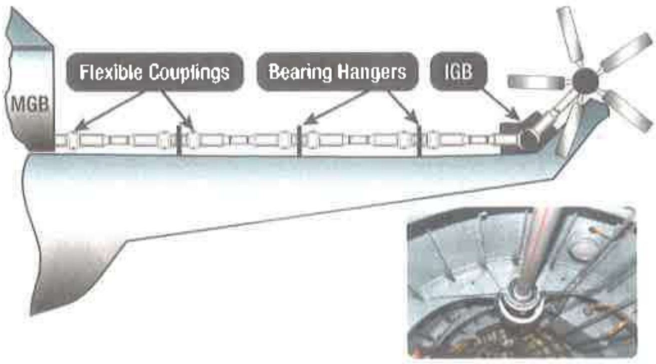


Figure 4-19. Heavy helicopter horizontal shaft.

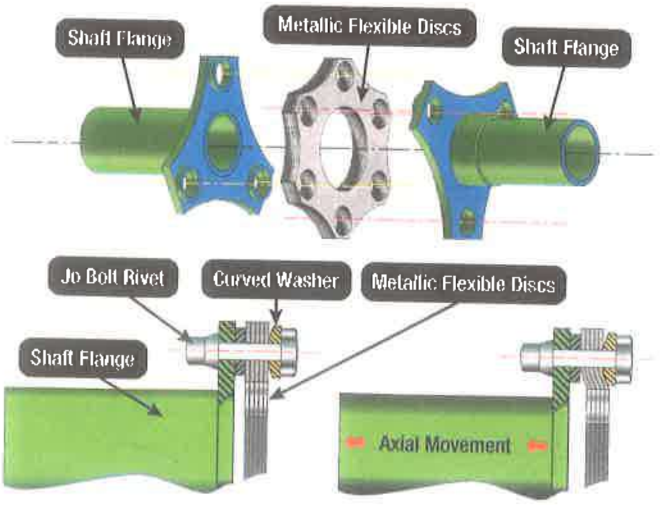


Figure 4-20. Flexible coupling.

A second function of flexible coupling is to connect two shafts which do not have the same axis of rotation. A short angle will create constraints on a rigid connection and will gradually destroy the bearings or cause leaks to appear at the gearboxes. The flexible connection described above is the main type used in aeronautics, but some variations exist. Another arrangement for flexible metal discs is the Kaplex coupling, which is a single rubber disc as a replacement for the metal discs. (*Figure 4-21*)

While flexible couplings allow small movements, the last connection of the horizontal shaft with the IGB or the TGB (depending on the configuration) requires more freedom. To enable this connection, a sliding coupling is inserted inside the power transmission train.

BEARINGS

Bearings for Transmissions

Bearings for transmissions are used to support the gears and shafts which transmit power from the engine to various components. These bearings are highly reliable as they support critical parts of the helicopter. Each serves a special purpose carrying rotational loads, thrust loads, or oscillating movements which are associated with the flight characteristics of the helicopter. Their types are:

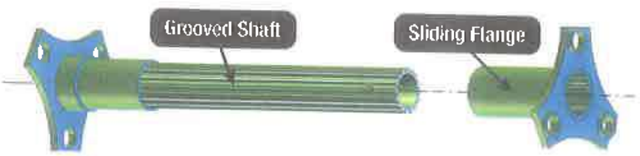


Figure 4-21. Sliding coupling.

- Ball and roller bearings are used in an IGB, as seen in ***Figure 4-11***.
- Ball and tapered-ball bearings are used in a TGB, as seen in ***Figure 4-12***.
- Spherical bearings are especially used on control linkages and rod ends, where movement in more than one direction is desirable. They are used to compensate for misalignment, therefore it is important to properly align them at installation. Spherical bearings are often used with forked ends and so the movement of the part may be limited in case of a misalignment.

Elastomeric Bearings

Elastomeric bearings are used for oscillating loads where complete rotation is unnecessary. Their types are:

- Cylindrical elastomeric bearings absorb high radial loads and provide movement in a radial oscillation, such as the teeter-totter motion of a two bladed main rotor assembly.
- Spherical elastomeric bearings provide movement about the three axes and absorb heavy torsional loads. This bearing could be used for the tail rotor pitch change mechanism.
- Conical bearings are capable of absorbing high radial and axial loads with some movement in both directions. These types of bearings used in combinations, can provide the movements necessary and carry the loads required for a complete rotor system.

Maintenance of Bearings

Bearings in hanger support fittings and outer couplings have to be inspected for grease leakage, wear, roughness and binding. Improper care of these bearings may lead to catastrophic failure. They require lubrication in order to ensure long life and prevent failure. Many may be lubricated by grease. The acceptable type of grease will be specified by the manufacturer and may be a high temperature, low temperature, or multi-purpose type. A schedule for changing the grease will be determined by the manufacturer and should be considered as a minimum requirement. Other bearings may be lubricated with oil, by splash, spray, or pressure feed. Sealed bearings require no additional lubricant. No lubrication is necessary for elastomeric bearings because the elastomer itself is the lubricant. Note: Further details on bearing construction are given in *Module 06 - Materials and Hardware.*

VIBRATION DAMPERS

As the name suggests, dampers are used to dampen or reduce the frequency of oscillation of the vibrating components of a machine by absorbing part of the energy produced by a vibration. Main and tail rotor dampers are used to fulfill a critical vibration management function by absorbing and dissipating this energy. (***Figure 4-22***)

Classification of Vibration Dampers

Various damping methods are classified as passive or active. A variety of systems are in use for damping.

- Passive methods include rubber mountings, etc.
- Active methods include computer controlled anti-vibration generators.

Resonant Mass

One dynamic response to vibration is when the frequency of the vibration is modified by the addition of a spring supported mass or masses. If the characteristics of the mass and spring are adjusted to resonate at the vibrating frequency of the component, then the vibration will be damped and not transmitted to the airframe.

Nodal Beam

The helicopter fuselage is flexed due to the oscillating vertical forces of the rotor. As the center of the fuselage and the rotor and transmission move up, each end of the fuselage moves down and so becomes 'out of phase'. The points at which this change of direction occurs are called nodal points where no movement occurs. A nodal beam system consists of flexible members tuned to vibrate in resonance with the rotor and will add an equal but opposite force to whatever is causing it to vibrate. The main rotor gearbox is mounted on the nodal beam which in turn is attached to the fuselage by elastomeric bearings thus helping to dampen the main rotor vibrations.

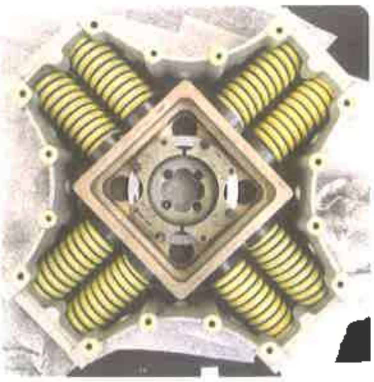


Figure 4-22. Bell 407 Vibration Damper.

Pendulum Damper or Bifilar

Counterweights are attached to small arms at right angles to the rotor shaft center line and fitted above the blade attachments to rotate with the main rotor. Since the weights are subject to centrifugal force which varies with rotor RPM, and because vibration is related to RPM, the damping is effective throughout the operating speed range of the rotor.

Hydraulic Damper

This kind of damper is designed to dampen out the vibrations coming from the gearbox by means of a piston and cylinder arrangement and a spring, with a working medium such as a water/glycol fluid mixture. As the gearbox vibrates vertically, the combined action of the spring and the cyclic's pressurizing of the fluid sets up an anti-resonance frequency. This effect cancels the vibrations that would normally go to the fuselage.

Computer Controlled Dampers

Computer controlled dampers are an active system. Transducers are placed in the fuselage at strategic locations where they send signals to a computer measuring the vibration characteristics at their locations. The computer in turn sends control signals to resonator units fitted to the fuselage sides to apply vibration forces in anti-phase to the vibrations coming from the rotor. (***Figure 4-23***)

These resonators are not unlike loudspeakers that have a mass moved by a coil that moves to produce the required control load. Advantages of this system are lighter weight, reduced noise, greater effectiveness than passive systems, and minimal use of power (under one watt).

BEARING HANGERS

In both large and small helicopters, to guide the long shaft it is necessary to use bearing supports. This prevents bearing wear and at the same time allows the alignment of the shaft to be adjusted. If the two or three bearings of the horizontal shaft are not perfectly aligned, a vibration will appear and destroy the bearings and the seals in the gearbox. Two adjustment directions are possible. A vertical support with the addition of shims between the hanger and the beam, and a horizontal adjustment with a slotted hole. (***Figure 4-24***)

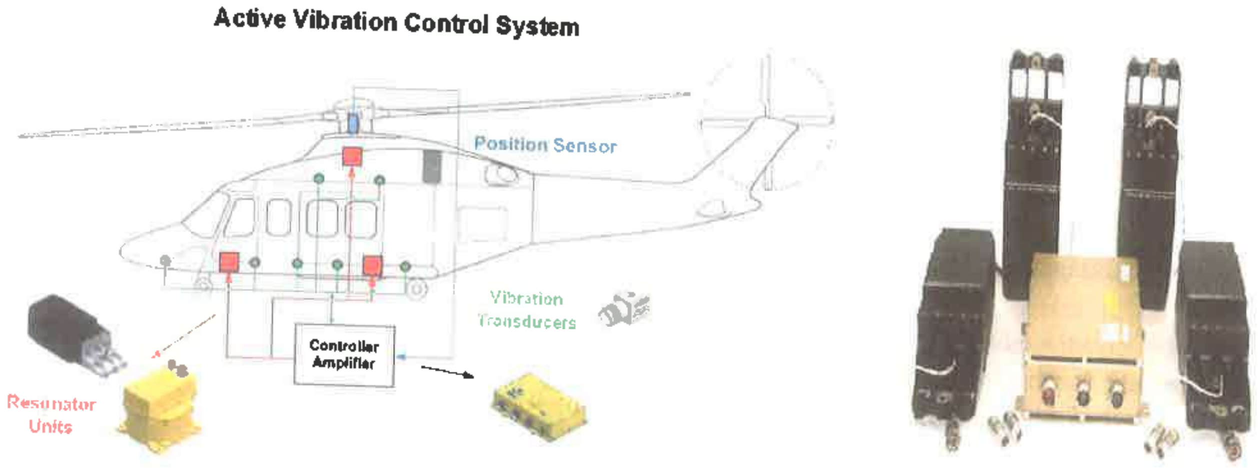


Figure 4-23. Active Vibration Control System (AVCS).

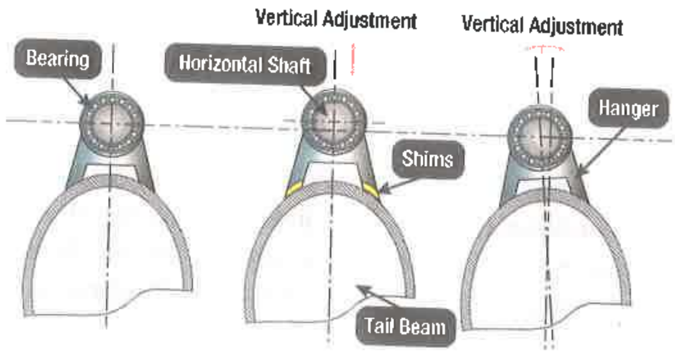


Figure 4-24. Bearing hanger adjustment.

Question: 4-1
What are the two functions of gear boxes on helicopters?

Question: 4-2
Within an epicyclic gear box, which gears turns which?

Question: 4-3
Of the three gear boxes within a helicopter drive train (the MGB, IGB, and TGB), which typically serves only to provide a change of direction?

Question: 4-4
What event causes a freewheel device to separate the engine from the rotor?

Question: 4-5
Within a clutch system, what causes the clutch pads to make contact with the ferrodo?

Question: 4-6
What are the two purposes of a rotor brake?

Question: 4-7
What factor primarily causes the need for flexible tail rotor shafts and couplings?

Question: 4-8
What are the primary maintenance needs of bearings within the drive train?

ANSWERS

Answer: 4-1
To set the speed of rotation of each rotor. To accomplish directional changes within the powertrain.

Answer: 4-2
The sun gear drives the planet gears which drives the planet carrier.

Answer: 4-3
The Intermediate Gear Box (IGB).

Answer: 4-4
When the rotational speed of the rotor exceeds the speed of the engine.

Answer: 4-5
Centrifugal force generated from the spinning engine shaft.

Answer: 4-6
To stop the rotor from spinning after engine shut down. To hold the rotor in place during start up until enough torque is generated to quickly bring the rotor to a lift producing speed.

Answer: 4-7
Movement of the tail boom due to its weight and inflight stresses due to yaw.

Answer: 4-8
Grease leaks and grease change schedules.

HELICOPTER AERODYNAMICS, STRUCTURES AND SYSTEMS

AIRFRAME STRUCTURES

SUB-MODULE 05

PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY → B1.3 B1.4

Sub-Module 05

AIRFRAME STRUCTURES

Knowledge Requirements

12.5 - Airframe Structures

(a)	Airworthiness requirements for structural strength; Structural classification, primary, secondary and tertiary; Fail safe, safe life, damage tolerance concepts; Zonal and station identification systems; Stress, strain, bending, compression, shear, torsion, tension, hoop stress, fatigue; Drains and ventilation provisions; System installation provisions; Lightning strike protection provision;	2
(b)	Construction methods of: stressed skin fuselage, formers, stringers, longerons, bulkheads, frames, doublers, struts, ties, beams, floor structures, reinforcement, methods of skinning and anti corrosive protection. Pylon, stabilizer and undercarriage attachments; Seat installation; Doors: construction, mechanisms, operation and safety devices; Windows and windscreen construction; Fuel storage; Firewalls; Engine mounts; Structure assembly techniques: riveting, bolting, bonding; Methods of surface protection, such as chromating, anodizing, painting, cladding; Surface cleaning. Airframe symmetry: methods of alignment and symmetry checks.	2

12.5 - AIRFRAME STRUCTURE

PART A

AIRWORTHINESS REQUIREMENTS FOR STRUCTURAL STRENGTH

The structure of an aircraft must be strong enough to carry all the loads to which it might be subjected, including the repeated small to medium loads experienced in normal flight, and the large loads experienced in extreme conditions. To avoid losing performance in flight, the helicopter airframe must have an aerodynamic shape. Structural elements called members, having a high strength to weight ratio, must be fitted into this shape and bear the forces necessary to balance the helicopter in flight. The aircraft must be capable of withstanding much more force than that imposed by its own weight. When the purpose of a particular design is established, the designers develop the helicopter structure according to strict standards established by aviation authorities to ensure safety.

The requirements for airworthiness are set by EASA. For non-EASA states, the requirements for airworthiness are set by its authority, often with the guidance of EASA. The objective, in order to safeguard aircrews and the public, is to only allow flight certification for aircraft meeting established minimum standards.

Two types of helicopters are defined by EASA Certification Specifications (CS); small helicopters by CS 27 and the large helicopters by CS 29. The definition of a small helicopter is a weight limitation of 3 175 kg and a passenger seat limitation of 9. Above these limitations a helicopter is defined as large.

CERTIFICATION SPECIFICATION (CS) 27

If the helicopter is less than 3 175 kg and has a maximum of 9 passenger seats, the CS27 is applied. Multi-engine rotorcraft may be type certificated as Category A.

CERTIFICATION SPECIFICATION (CS) 29

Large helicopters may be certificated in accordance with either the Category A or Category B requirements. A multi-engine rotorcraft may be type certified as both Category A and Category B with different operating limitations for each category. Rotorcraft greater than 9 072 kg and 10 or more passenger seats must be type certified as Category A. Rotorcraft with a maximum weight greater than 9 072 kg and 9 or less seats may be type certified as Category B, subject to certain conditions. Rotorcraft with a maximum weight of 9 072 kg or less but with 10 passenger seats may be type certified as Category B subject to certain conditions. Rotorcraft with a maximum weight of 9 072 kg or less and 9 or less passenger seats may be type certified as Category B rotorcraft. This summary of related Certification Specifications is shown in the table. (*Table 5-1*)

For information purpose only, the subparts of CS 27 and CS 29 are defined as:

- Subpart A – General
- Subpart B – Flight
- Subpart C – Strength Requirements
- Subpart D – Design And Construction
- Subpart E – Powerplant
- Subpart F – Equipment
- Subpart G – Operating Limitations And Information

	LESS THAN 3 175 kg	BETWEEN 3 175 KG - 9 072 kg	MORE THAN 9 072 kg
0-9 Passengers	CS 27 - Category A Rotorcraft SMALL HELICOPTERS	CS 29 - Category B Rotorcraft	CS 29 - Category B Rotorcraft or Category A requirements if Subparts C, D, E and F are met.
10+ Passengers	CS 29 - Category B Rotorcraft or Category A requirements if CS 29.67(a)(2), 29.87, 29.1517, and of Subparts C, D, E, and F are met.	CS 29 - Category B Rotorcraft or Category A requirements if CS 29.67(a)(2), 29.87, 29.1517, and of Subparts C, D, E, and F are met.	CS 29 - Category A Rotorcraft

Table 5-1. CS-27 and CS-29 Summary.

STRUCTURAL CLASSIFICATION, PRIMARY, SECONDARY, TERTIARY

Aircraft structure is divided into 3 categories for assessing damage and for the application of repair protocols suitable for the structure under consideration. Manufacturer manuals designate which category a structure falls under and the technician is required to repair and maintain that structure in accordance with rules specified for the category under which it falls. The three categories for structure are: Primary, Secondary and Tertiary.

PRIMARY STRUCTURE

Primary structure is any portion of the aircraft structure that, if it fails on the ground or in flight, would likely cause any of the following:

- A loss of control of the aircraft.
- Catastrophic structural collapse.
- Injury to occupants.
- Power unit failure.
- Unintentional operation.
- Inability to operate a service.

The primary structure is composed of highly stressed components, and if damaged, may cause failure of the helicopter and loss of life of the aircrew.

Some examples of primary structure are engine mounts, fuselage frames, fuselage spars, and main floor structural members. Within the primary structure are elements called Principle Structural Elements. These elements are those which carry flight and ground loads. Primary structure may also be represented as a Structurally Significant Item. These elements are specified in a supplemental structural inspection document. Due to their structural importance, they may require special inspections and they have specific repair limitations.

SECONDARY STRUCTURE

Secondary structure is all non-primary structural portions of the aircraft which have integral structural importance and strength exceeding design requirements. These structures may weaken but without risk of failure such as those described as primary structure. Secondary components of the airframe are highly stressed but, if damaged, will not cause failure of the helicopter or loss of life. Prominent examples of secondary structure are fuselage ribs, stringers, and specified sections of the aircraft skin. The secondary structure often provides the aerodynamic shape to the helicopter construction.

TERTIARY STRUCTURE

Tertiary structure is all remaining structure. Tertiary structures are lightly stressed elements that are fitted to the aircraft for various reasons. Examples of tertiary structure are fairings, fillets, various support brackets, etc.

FAIL SAFE, SAFE LIFE, DAMAGE TOLERANCE CONCEPTS

The safety factor of an aircraft is usually expressed as a ratio of the maximum load carrying capability of an airframe to its expected loading. Loading may be static, or due to impact, fatigue, wear, etc. The purpose of using a safety factor is to assure that the design does not fail in the event of unexpectedly high loads or the presence of material/design defects. Multiple factors of safety are applied to decrease the probability of failure. Rigorous controls over airplane structures and systems, beginning with its fabrication and assembly through inspection and maintenance are done on all aircraft. Extensive fatigue and static testing is conducted on its components and systems.

To determine appropriate factors of safety, the potential harm that a failure can produce is considered. If failure would result in a mere inconvenience, then a small factor of safety may be acceptable. If failure would be expensive or even life threatening, then a larger factor is justified.

FAIL-SAFE CONCEPT

Fail Safe designs are those that incorporate various techniques to mitigate losses due to system or component failures. The design assumption is that failure will eventually occur but when it does the device, system or process will fail in a safe manner. Fail Safe means the structure has been evaluated to assure that a catastrophic failure is not likely in the event of a failure of a single, structural element. A Fail Safe designed aircraft is designed so that the aircraft may continue to operate safely until the defect is detected in a scheduled maintenance check. Manufacturer testing and fatigue analysis is used when developing Fail Safe structural elements. The full structure is then considered to be damage tolerant.

This type of design is common on modern helicopters and is an extension of the concepts described below. The principle of fail safety consists in providing redundant load paths as back-ups in the event of localized failure. If one load path cracks completely through, or sustains accidental damage; the remaining load paths carry the additional load. For example, this could be the use of multiple stringers and frames in fuselage construction including redundant panels, multiple stringers and ribs in wings, bonded and bolted fittings, and bonded and bolted landing gear beams. This concept requires that any damage is detected during an inspection and then repaired.

EXAMPLES OF FAIL SAFE DESIGN TECHNIQUES

- Redundancies: If failure of a critical subsystem will cause severe losses, backup systems are often employed.
- Multiple load paths: If a structural element fails, the load it was carrying will be transferred to other members. It is essential that the fracture be detected before multiple members fail.
- Intentional weak link: A weaker, inexpensive and easy to replace component may be intentionally used to prevent damage to an expensive or difficult to repair component.
- Early detection: Fail Safe design considers that cracks will easily be detected before they reach critical length. But as this kind of detection is critical, it is important that proper materials be used that can withstand large cracks before fracturing.
- Crack arresters: Crack arresters are used to prevent cracks that exceed a critical length from fracturing the entire part. Crack arresters may be added to the structure. These are often in the form of riveted straps added to the skin. The straps will contain the crack to a small area of the structure. As a crack approaches, the arresters start to carry more and more load, thus decreasing the load near the crack's tip.

SAFE LIFE CONCEPT

Safe Life is the concept that a component or system is designed to not fail within a defined period if it is used within normal operating limits. It is assumed that testing and analysis can provide an adequate estimate for the component's or system's expected lifetime. At the end of this expected life, the part is removed from service. For example: pumps, filters, valves, landing gear, etc. These parts cannot be repaired or refurbished to extend the component's life. The basis for Safe Life design is fatigue analysis to estimate how long the component can be in service before it will likely fail. The product should be designed so that it can be easily inspected in service.

Safe life structural elements are those which have an extremely low risk of unacceptable degradation or failure for a stated time. The fatigue capability of the structure is learned through testing. The stresses applied while in service are designed to be significantly lower. Also, the calculated time in service before failure is greatly reduced so that a failure of the structure before its safe life is highly unlikely. The effects of corrosion, wear and fatigue are considered when operating under the safe life design principle.

DAMAGE TOLERANCE CONCEPT

Designing aircraft with fail safe principles can be somewhat unreliable. Accidents have occurred proving this. Engineering improvements to a fail-safe structure typically come with the extra penalty of adding weight. Thus, the damage tolerance concept of engineering is favored as it considers multiple sites cracking, and the residual strength of partially failed elements. Damage tolerance also considers the effects of environmental damage (corrosion), and discrete damage (accidental). The damage tolerance principle requires that any helicopter damage is detected and repaired before the strength is below a minimum level. By distributing loads over a larger area and designing multiple load paths for carrying loads, a structure can be damage tolerant. The structure retains its integrity, and the damage does not worsen in service between inspections when it can be detected and repaired. Thus, damage tolerance means that the structure has been evaluated to ensure that even if serious fatigue, corrosion, or accidental damage occurs, the remaining structure can withstand reasonable loads without failure or excessive deformation until the damage is detected.

ZONAL AND STATION IDENTIFICATION SYSTEMS

ZONAL SYSTEM

Zones can be used to describe areas in a helicopter. These zones typically coincide with the major sections of the airframe. A zonal identification permits you to reference the different sections within the aircraft's documentation. Each chapter has an ATA 100 number as follows:

100 Fuselage
200 Cockpit
300 Tail beam
400 Engine, transmission, main and tail rotors
500 Left multipurpose carrier
600 Right multipurpose carrier
700 Landing gear
800 Optional equipment
900 Customization

STATION IDENTIFICATION SYSTEM

To identify a particular location on a helicopter, the airframe is divided into three planes that are at an angle of 90° to each other. It is then necessary to have references from which a distance from each plane will be given. This distance is in millimeters or inches and a sign (+ or -) defines the direction. A vertical plane is in the front of the helicopter and moves along the longitudinal axis. The zero reference is usually not on the nose extremity but on the first vertical component. This gives the X coordinate of a structure. (*Figure 5-2*)

A vertical plane cuts the helicopter into two equal parts along the longitudinal axis; the left side and the right. Positive values are to the right and negative values are to the left. This is the Y station. (*Figure 5-3*)

A horizontal plane cuts the helicopter into two parts. All the values above the reference plane are positive and all the values under the reference plane are negative. This is the Z station. (*Figure 5-4*)

Using these three coordinates, it is easier to identify the exact position of any structural component. Each manufacturer's maintenance manual explains their

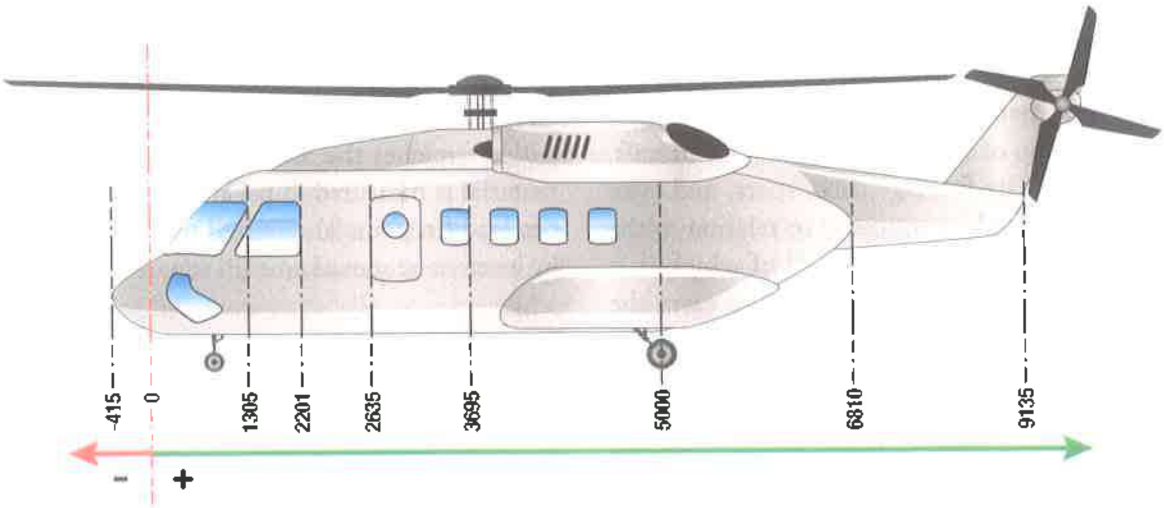


Figure 5-2. X Station.

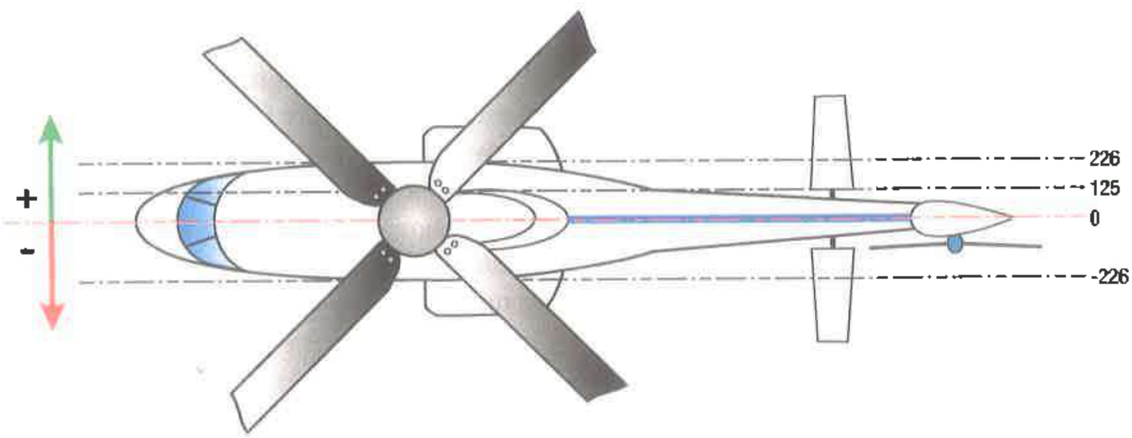


Figure 5-3. Y Station.

Figure 5-4. Z Station.

numbering system and often contains numerous diagrams and tables showing the location of various components, and under which panel they may be found.

STRESS, STRAIN, BENDING, COMPRESSION, SHEAR, TORSION, TENSION, HOOP STRESS, FATIGUE

Most helicopter structures are of a semi-monocoque configuration. A shell structure is reinforced by frames and longerons to absorb loads. Spars, ribs and stingers covered by a simple metal skin complete the structure. Aircraft structural members are designed to carry a load or to resist a direction of stress. In designing an aircraft, every square inch of its fuselage, ribs, spars, and even each metal fitting must be considered in relation to the physical characteristics of the material of which it is made. Every component must be designed to carry the load to be imposed upon it. The determination of such loads is called stress analysis. Although planning this design is not the function of the aircraft technician, it is, nevertheless, important that the technician understands and appreciates the various stresses involved, in order to avoid compromises to the original design through improper repairs.

STRESS AND STRAIN

The term "stress" is often used interchangeably with the word "strain." While related, they are not the same. External loads or forces cause stress, which is defined as "a load applied to an area of material". Stress produces a deflection or deformation in the material, and the amount of deformation is called strain. Stress is always accompanied by strain. When a material is subjected to a load, that material is deformed, regardless of how strong the material is or how light the load is.

There are five major stresses to which all aircraft are subjected: (***Figure 5-5***)

- Tension
- Compression
- Torsion
- Shear
- Bending

TENSION

Tension is a force tending to pull things apart. (***Figure 5-5A***) The engine pulls the aircraft forward, but air resistance tries to hold it back. The result is tension which stretches the helicopter. The tensile strength of a material is measured in psi and is calculated by dividing the load (in pounds) required to pull the material apart by its cross-sectional area (in square inches).

COMPRESSION

Compression is the stress that resists crushing. (***Figure 5-5B***) The compressive strength of a material is also measured in Pounds Per Square Inch (psi). Compression is the stress that tends to shorten or squeeze helicopter parts.

TORSION

Torsion is the stress that produces twisting. (***Figure 5-5C***) It is a combination of tension and compression. While moving the aircraft forward, the engine also tends to twist the aircraft to one side, but other airframe components hold it in place. Thus, torsion is created. The torsion strength of a material is its resistance to twisting or torque. Helicopter rotor shafts are subjected to torsional stresses.

SHEAR

Shear is the force which tends to cause one layer of a material to slide over an adjacent layer. (***Figure 5-5D***)

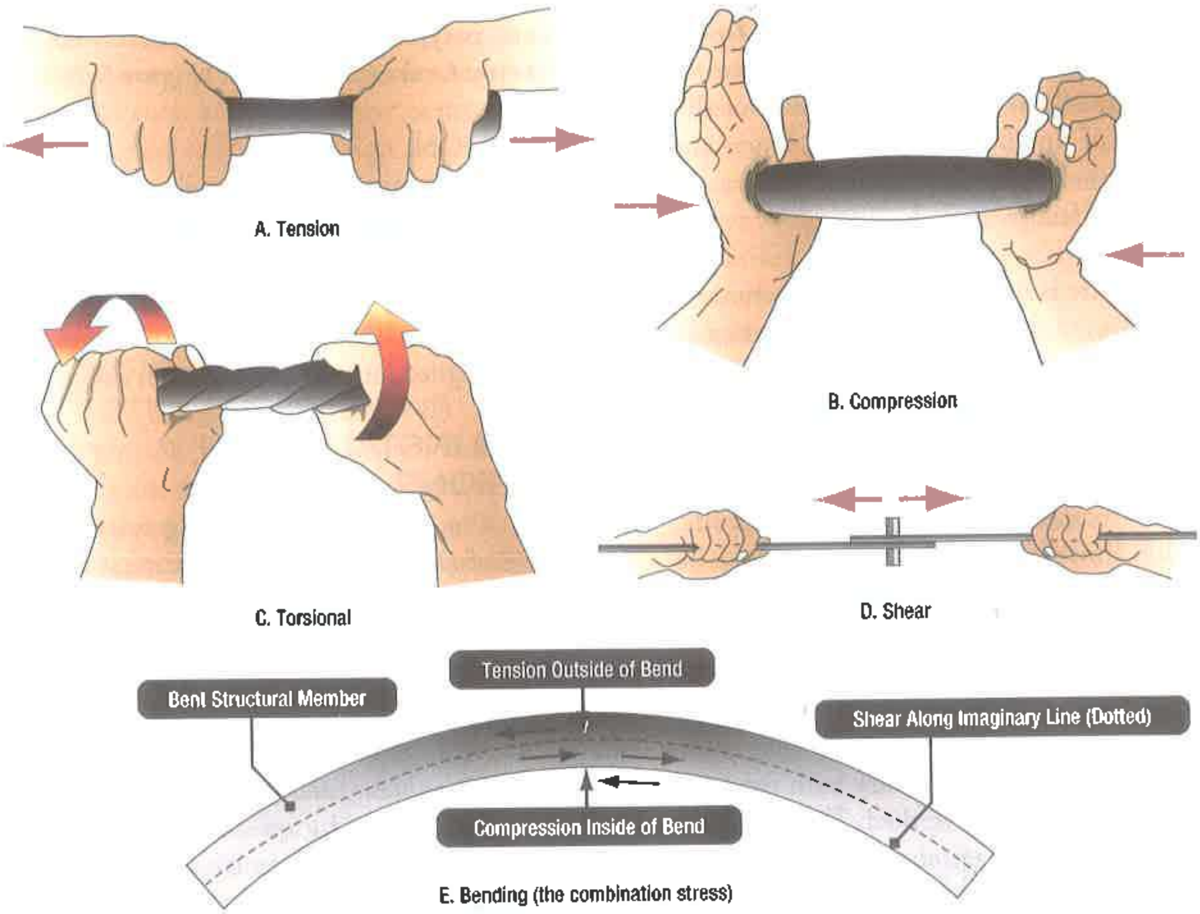


Figure 5-5. Various structural stresses.

Two riveted plates under tension subject the rivets to a shearing force. Usually, the shearing strength of a material is either equal to, or less than its tensile or compressive strength. Helicopter parts, especially screws, bolts, and rivets, are often subject to a shearing force.

Engineers must consider numerous other characteristics in addition to these stresses when designing helicopters and their components. For example, cowling, fairings, and similar parts may not be subject to loads requiring a high degree of strength. Instead they must have streamlined shapes to meet aerodynamic requirements, such as reducing drag or directing airflow.

BENDING
Bending stress is a combination of compression and tension. The rod in (***Figure 5-5E***) has been shortened (compressed) on the inside of the bend and stretched on the outside of the bend. A single member of the structure may be subjected to a combination of stresses. In most cases, structural members of a helicopter are designed to carry end loads rather than side loads. Thus they are subjected to tension or compression rather than to bending. The rotor blades of the helicopter are an exception and under a bending stress.

HOOP STRESS
Hoop stress is the stress on the structural components of the airframe caused by cabin pressurization. However, due to the low altitude flight characteristics of helicopters, pressurization is unnecessary and so this stress is not considered.

METAL FATIGUE
Metal fatigue is experienced by a component when a load is repeatedly applied and released or if applied and reversed. This cycle weakens the material over time even though the load applied with each cycle may be far below one which could cause damage in a single application. All materials have an elastic limit. If applied loads do not exceed this limit, the material should be unaffected by the load and will return to its original state when the load is removed. However, an aircraft in flight constantly experiences varying loads. Over time, these small load changes cause fatigue in the form of minute cracks in the metal structure. Each seemingly

inconsequential crack exposes new material to the weather elements which may then further weaken the material due to the effects of corrosion. Additionally, when a multitude of tiny fissures combine, larger and more significant cracks may develop and weaken the metal to the point of failure.

An aircraft's structure is tested by the manufacturer to determine what limits may not be exceeded when in service. Often, fatigue testing is accomplished on full scale rigs which subject the structural elements to cycles of loading and unloading well beyond that which will be experienced in service. A fatigue index is applied, and the aircraft is monitored throughout its service life. If its fatigue life limit is reached, an aircraft may be reevaluated to know its actual condition. If the loading and environmental exposure of the structure was not as harsh as predicted, it is possible to extend the service life of the aircraft. In this case an increase in inspection frequency and/or strengthening modification(s) may be required. Fatigue characteristics vary with the type of metal and how this metal is worked. The thickness of the material and type and number of fastener holes can also alter the fatigue life. Aging helicopters are monitored and treated by technicians to protect against corrosion which accelerates fatigue. On a helicopter the part that is most carefully monitored is the tail boom, due to its flexibility and the stresses applied to it during landing or autorotation.

DRAINS AND VENTILATION PROVISIONS

DRAINS

The collection of water and other fluids in the many cavities found on an aircraft can lead to corrosion and/or a fire hazard. Additionally, for helicopters used near sea level, ocean salt can damage various structures. In the event of a fuel or hydraulic leak, its pooling in a part of the structure can be dangerous due to its flammability. Drainage and ventilation are used to address this issue. Drains are located throughout the outer surface of the fuselage and open to the exterior. (***Figure 5-6***) Leakage and staining from drains can help the mechanic locate an internal leak and fix it quickly.

Leveling compound is sometimes used to build up a low area near a drain valve to ensure that no fluid is trapped, but instead is quickly drained from the orifice. The compound is typically a waterproof rubber like sealant without structural characteristics. (***Figure 5-7***)

VENTILATION

Any cavity in the aircraft structure that may experience the presence of water or a flammable vapor must be ventilated to permit the vapor to escape or evaporate. If necessary, vent pipes are used providing an escape route for the vapor. The technician should ensure that all openings designed for ventilation are not clogged.

SYSTEM INSTALLATION PROVISIONS

In addition to designing supporting systems for the operation of the aircraft, designers must also make those components fit into the aircraft. Depending on the system and components, provisions for access and servicing must also be addressed. Items that receive regular maintenance such as filters, fluid checks, bearing lubrication, etc., must be located so that technicians can easily access them. Line Replaceable Units (LRU's) must also be able to be quickly uninstalled and installed. Aircraft maintenance is a significant expense for the operator. Anything that can be done to locate components for easy access saves time and lowers the cost of operating the aircraft.

Figure 5-6. Drain plug.

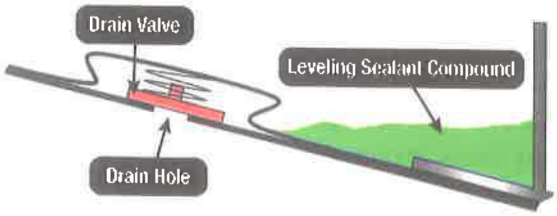


Figure 5-7. Leveling sealant compound.

Avionics and electronic components are frequently mounted in an easy to access avionics bay. Structural components such as landing gear connections must be easily accessible for inspection following unusual stresses such as a hard landing. Specific openings must be included to allow the control cables, fluid lines, electric cables and pneumatic ducting to pass through and be serviced.

LIGHTNING STRIKE PROTECTION PROVISIONS

During flight, the parts of a helicopter most often struck by lightning are the main or a tail rotor blades. Even if the direct consequence does not affect flight, the damage to the blade is expensive as it often must be replaced. Precautions are taken to ensure safe and continuous operation of an aircraft in the event of a lightning strike.

A single lightning strike may contain more than 100 000 amperes of current. This current must not be allowed to build up or arc from one point on the structure to another. Aircraft use an aluminum structure as a ground path for electrical devices. Most components are therefore mounted directly to the structure or attached to it with bonding straps. This ensures that all components are at the same electrical potential and that equal and low resistance paths for current flow exist. (*Figure 5-8*) Not only are the electrical components bonded to the aircraft structure, but different parts of the structure are bonded together as well. For example, hinged flight controls have a bond strap between the movable surface and the main airframe structure.

As an aircraft flies throughout the air, its surface can also become highly charged with static electricity. Static dischargers, or wicks, are installed on aircraft to reduce radio receiver interference. This interference is caused by corona discharge emitted from the aircraft as a result of precipitation static. Corona then occurs in short pulses which produce noise at the radio frequency spectrum.

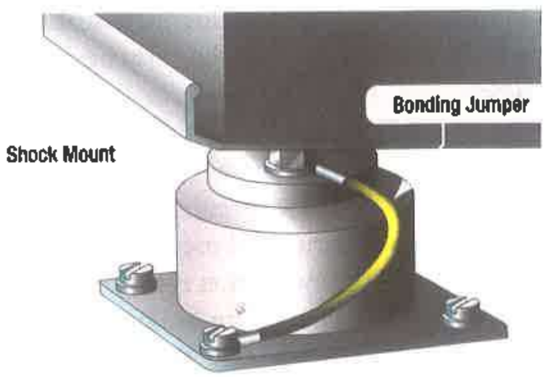


Figure 5-8. Bonding jumper.

Static dischargers are normally mounted on the trailing edges of the control surfaces, blade tips and the vertical stabilizer. They discharge precipitation static at points located at a critical distance away from avionics antennas so that there is little or no coupling of the static to cause interference or noise. Flexible and semi flexible dischargers are attached to the aircraft structure by metal screws, rivets, or epoxy. These connections should be periodically checked. A resistance measurement from the mount to the airframe should not exceed 0.1 ohm.

The condition of all static dischargers and bonding straps are inspected in accordance with the manufacturer's instructions. When lightning strikes an aircraft with all bonding devices intact and working, there is no difference in potential from one part of the aircraft to another. The electrical energy dissipates over the entire surface of the aircraft and returns to the atmosphere through the static wicks. Composite materials used in modern aircraft are not naturally conductive. To achieve the same static and lightning protection as aluminum aircraft, conductive wires or layering of conductive material are built into the composite structure to ensure even distribution of electrical charges when all bonding procedures are followed.

BONDING PROCEDURES AND PRECAUTIONS

When making bonding or grounding connections in aircraft, these general procedures should be observed:

- Bond or ground parts to the primary aircraft structure where practical.
- Make bonding or grounding connections in such a way as to not weaken any part of the structure.
- Bond parts individually wherever possible.
- Make bonding or grounding connections against smooth clean surfaces.
- Install bonding and grounding connections so that vibration, expansion or contraction, or relative movement will not break or loosen the connection.
- Locate bonding and grounding connections in protected areas when possible. Also when possible, locate connections near hand holes, inspection doors, and other accessible areas for easy inspection and replacement.

- Do not fasten bonding or grounding connections through any non-metallic material.
- Inspect all grounding and bonding straps to ensure they are free from corrosion which will adversely affect performance.
- No more than 4 ground wires should be connected to a common ground stud. Each ground for electric power sources should be connected to separate ground points. Grounds for utilized equipment may be connected to a common point only when supplied from the same power source.

PART B

CONSTRUCTION METHODS OF: STRESSED SKIN FUSELAGE, FORMERS, STRINGERS, LONGERONS, BULKHEADS, FRAMES, DOUBLERS, STRUTS, TIES, BEAMS, FLOOR STRUCTURES, REINFORCEMENT, METHODS OF SKINNING AND ANTI-CORROSIVE PROTECTION

The fuselage is the main structure or body of the aircraft. It provides space, first for engines, components and flight controls, crew, and eventually cargo or passengers. There are two general types of fuselage construction: truss and monocoque.

TRUSS TYPE

The earliest helicopter airframes were made of open trusses of either wood strips or bamboo. A truss is a rigid frame made up of beams, struts, and bars to resist deformation due to the applied loads. A later lattice-type fuselage frame was usually made of steel tubes welded together in such a way that all elements of the truss could withstand both tensile and compressive loads. (***Figure 5-9 and Figure 5-10***)

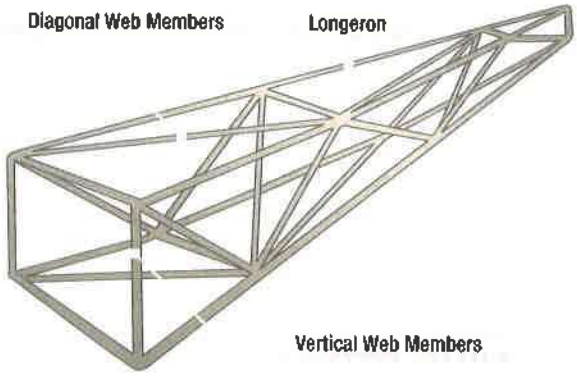


Figure 5-9. Truss structure.


Figure 5-10. Truss tail beam.

In light aircraft, mesh fuselage frames may be constructed of aluminum alloy and may be riveted or bolted in two pieces; the main fuselage and tail boom, with bracing achieved using solid rods or tubes. Within this design, it is necessary to have the possibility of disconnecting the tail from the fuselage to change it in the event of damage during a hard landing. This configuration is the best solution to balance lightness and stiffness. However a problem with this type of structure is the vibration of the helicopter and the flexibility of the tail boom combining to create cracks at the bar junctions. Secondly, when the manufacturer added a streamlined shape to improve the helicopter's aerodynamics, a new problem was an increase in weight.

STRESSED SKIN FUSELAGE

Many modern helicopters are now constructed using a monocoque construction. With this method, the skin itself carries much of the load and is supported on the inside by numerous structural elements such as bulkheads, frames, longerons and stringers. Generally, a Fail Safe design approach is used so that loads are spread through a variety of paths which allow partial failure without affecting the overall integrity of the aircraft structure. This stressed skin design may be divided into two classes:

- Monocoque (***Figure 5-11***)
- Semi-monocoque

The heaviest of these structural members are bulkheads which are partition type walls that typically span the full fuselage, often with an opening for access to its other side. They are located at intervals to carry concentrated loads, and at points where fittings attach other units such as, the tail boom, powerplants, and stabilizers. The biggest problem in monocoque construction is maintaining enough strength while keeping the weight within allowable limits.

To overcome the strength/weight problem of monocoque construction, a modification called semi-monocoque was developed. It consists again of frame assemblies, bulkheads, and formers as with the monocoque design but, additionally, the skin is reinforced by longitudinal members called longerons. Longerons usually extend across several frame members and help the skin support primary bending loads. They are typically made of aluminum alloy either of a single piece or of a built-up construction.

Stringers are also used in the semi-monocoque fuselage. These longitudinal members are typically more numerous and lighter in weight than the longerons. They come in a variety of shapes and are usually made from single piece aluminum alloy extrusions or of formed aluminum. Stringers have some rigidity but are mainly used for giving shape and for skin attachment. Stringers and longerons together prevent tension and compression from bending the fuselage. (***Figure 5-12***)

Other bracing between the longerons and stringers can also be used. Often referred to as web members, these additional support pieces may be installed vertically or diagonally. Note that various manufacturers use different names to describe structural members. For example, there is often little difference between some rings, frames, and formers. Manufacturer instructions and specifications for a specific aircraft are always the best guides.

The semi-monocoque fuselage is primarily constructed of aluminum and magnesium alloys, although steel and titanium are sometimes found in areas of high temperatures. Individually, none of the components mentioned above is strong enough to carry all the loads imposed during flight and landing. But, when formed with gussets, rivets, nuts and bolts, screws, and even friction stir welding they combine to form a strong and rigid framework. (***Figure 5-13***)

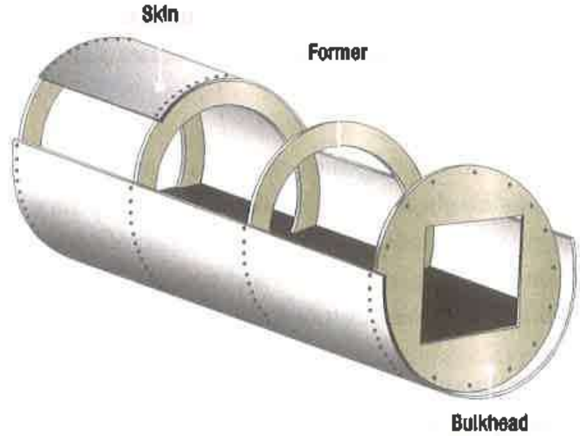


Figure 5-11. Monocoque structure.

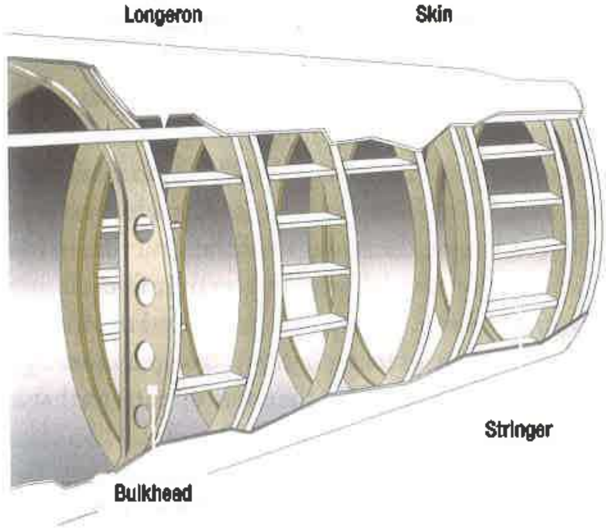


Figure 5-12. Semi-monocoque structure.

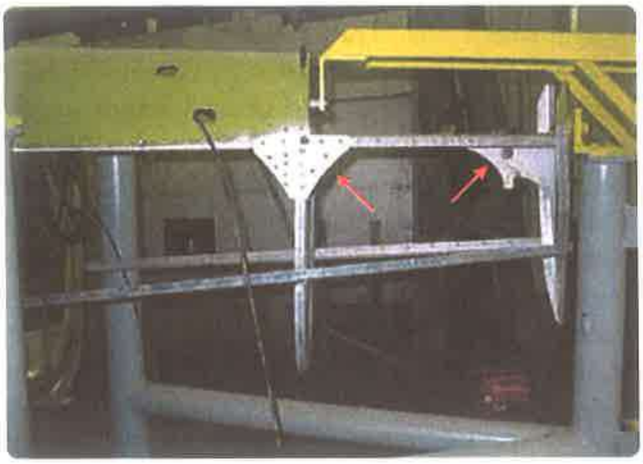


Figure 5-13. Gussets.

To summarize, in semi-monocoque fuselages, the strong longerons hold the bulkheads and formers, and these, in turn, hold the stringers, braces, web members, etc. All attach together and to the skin to achieve the full strength benefits of semi-monocoque design. However it is important to recognize that the skin still carries part of the load. The fuselage skin thickness can vary with the amount of load carried and the stresses sustained at that particular location.

Spreading loads among these structures and the stressed skin means no single piece is failure critical. This means that a semi-monocoque fuselage, because of its stressed skin construction, may withstand considerable damage and still be strong enough to hold together

BONDED AND COMPOSITE CONSTRUCTION

The problem with monocoque and semi-monocoque designs is the weight needed to obtain the required strength. To improve the strength to weight ratio, many modern helicopters are built with fiberglass, honeycomb, and bonded structures which have excellent resistance with low weight. Another advantage is that composite structures are easy to build, which reduces the price. While once using glass fibers bonded in an epoxy resin matrix, most manufacturers now use fibers of Kevlar and graphite to increase the strength of the structure in most applications, along with boron and ceramic fibers in some special applications. Composite structural components have the advantage of being lighter, stronger, more rigid, and better able to withstand the sonic vibrations that are commonly encountered in helicopters.

FORMERS, STRINGERS, LONGERONS, BULKHEADS, FRAMES

The bulkheads and circumferential frames define the shape of the fuselage while longerons aid in dispersing longitudinal loads. Transverse beams are used to strengthen the structure and stringers provide a means to securely attach the skin. Reinforced cavities are built into the fuselage to connect the tail boom, landing gear and other heavy components such as the APU. Cutouts for the doors and windows are reinforced locally to maintain proper load distribution around the openings. Attached to the fuselage are the tail boom, the pylon with the main gear box, the engines, the stabilizers and landing gear assemblies. Attachment points vary widely in location and method. Without exception, the structure in the area of major component attachments must be reinforced to transmit loads from these attached assemblies to the fuselage. (*Figure 5-14*)

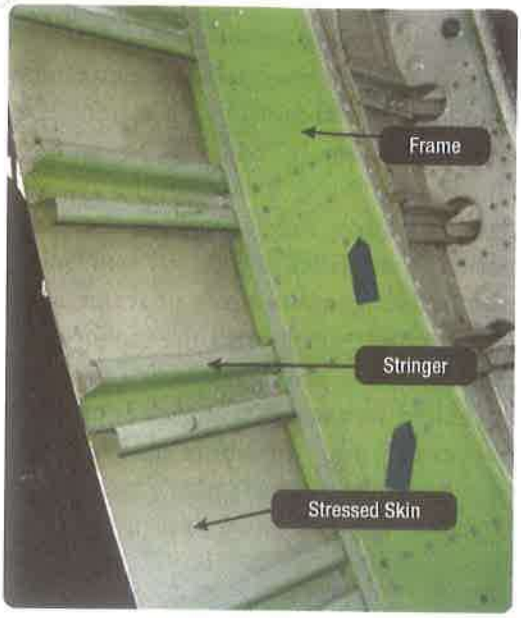


Figure 5-14. Frame, stinger, stressed skin.

DOUBLERS

An easy way to reinforce an area of skin which receives greater loads is with a doubler. A doubler is simply a second reinforcing layer of skin material used to strengthen its load carrying capacity. The advantage is that a doubler is inexpensive and can be shaped for a specific area needing reinforcement. Doublers are also used in sheet metal repair work.

STRUTS AND TIES

A strut is a bar or rod shaped reinforcement designed to resist compression loads. A tie is a rod or beam designed to take a tensile load. Both are used as needed to reinforce the aircraft structure throughout the fuselage to carry the expected and required loads.

BEAMS, FLOOR STRUCTURES, AND REINFORCEMENTS

In addition to the structural members mentioned, beams, floor structure and various other reinforcement members are also used to construct an aircraft. A beam may be installed laterally or longitudinally. Beams typically support the floor of the flight deck and the passenger compartment. They are located to provide secure attachment of the floor panels. The seats and attachment points in the cargo bay are strongly attached and secured. The floor itself is often made of numerous honeycomb panels that are screwed to the floor support structure. Flight deck floor panels may also be constructed from sheet metal. (*Figure 5-15*)

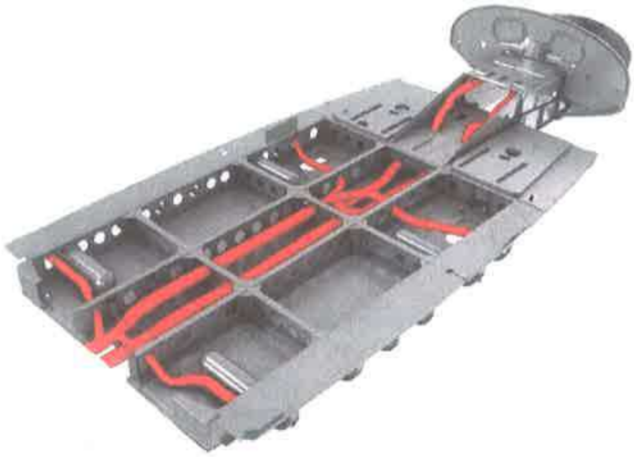


Figure 5-15. The floor structure of an AW-189 helicopter.

SKINNING

Attached to the outside of the aircraft structure is the skin, be it stressed or not. Simple, light aircraft generally have skin made from sheet aluminum which is formed to fit, and then wrapped and riveted to the structural members. Larger more complex aircraft use heavier material to form the aircraft skin to help transfer and carry the greater loads experienced during high performance flight. Some simple sheet metal skin may be found. However, various skin thicknesses are used to meet the design loads which vary by location around the aircraft.

Since in many areas the required skin thickness varies, machining the skin from a solid billet of material, including integrally formed stringers and risers has become a standard practice. By milling the skin from a single piece of material the thickness may be varied precisely to meet design requirements. Maximum strength is thus achieved with minimum weight.

Another process used in skinning a large aircraft is chemical etching. Etching of thick material to form thinner skin material with supporting raised patterns are produced without any stress. Skin with a "waffle plate" pattern is produced this way.

ANTI-CORROSIVE PROTECTION

Corrosion prevention is a consideration when materials are selected for construction. Additional anti-corrosion measures are then taken before and during construction. These range from heat treatments, to a variety of surface treatments, to design and assembly techniques.

Heat treatment of a metal can refine its grain structure so that it has the properties required for a specific function while reducing its susceptibility to corrosion.

Surface treatments can protect metals from contaminants and moisture which cause corrosion. Plating and cladding of materials are common methods of corrosion protection. Others, known as sacrificial coatings are designed to degrade rather than having the material they cover degrade. Common surface treatments such as paints and primers are used, as well as metal specific treatments such as anodizing and chromating. Numerous specialized surface treatments have been developed for specific metals in specific applications, all of which endeavor to keep the causes of corrosion at bay.

The design of an assembly can also be used to prevent corrosion. Something as simple as a well designed drain path or a drain hole placed in a strategic location can prevent corrosion of those materials in vulnerable areas. Wet assembly techniques and the use of sealants also provide a barrier to corrosion causing agents.

Manufacturers use all techniques at their disposal to produce corrosion resistant aircraft. However, varied operating environments and maintenance practices combined with service loads during operation make corrosion inevitable. Processing of data obtained from operations is used with a wide variety of inspection and testing techniques to find and eliminate corrosion before it reaches a critical phase. Anti-corrosion treatments and repairs specific to each aircraft are detailed throughout the manufacturer's maintenance manual, in ATA Chapter 51, Structures.

Corrosion rarely occurs on a clean dry aircraft when properly treated by the manufacturer during construction. While in service, it is impossible to avoid exposure of the aircraft to weather. Agents of corrosion such as dirt and moisture are encountered. A program of keeping the aircraft clean and its surface in good condition are main actions for operators to prevent corrosion. Technicians must assist by wiping up spills and removing deposits that contribute to the corrosive environment. Scratches, dents, and scoring should be avoided while performing maintenance. Drain holes must be clear so they can function as designed.

PYLON, STABILIZER, AND UNDERCARRIAGE ATTACHMENTS

FUSELAGE AND TAIL STRUCTURE

The fuselage is the entire forward structure from the cockpit to the middle section where cargo and/or passengers are located. (*Figure 5-16*)

The tail boom is considered separately from the fuselage. Attached at the end of the tail boom is the tail rotor. As we know, this is the anti-torque control for the main rotor. Although some torque loads are relieved in forward flight with a vertical fin, the side load is still present on the tail boom during all modes of flight. In addition to side load, many helicopters also have a horizontal stabilizer, which is pushing downward in cruise flight conditions. These loads are usually carried by the cantilevered tail boom that attaches to the main fuselage. These attachment areas must be subject to inspections because of the loads induced on them. They must withstand considerable vertical bending loads from the aerodynamic forces created by the horizontal stabilizer, plus lateral loads created by the vertical stabilizer and the tail rotor. If the tail rotor is positioned higher or lower than the tail boom centerline (on top of the vertical stabilizer for example) then it needs to withstand torsional loads as well. (*Figure 5-17*)

PYLON

The function of the pylon is to support the load of the engine(s) and the main gear box mounted on the top of the fuselage. (*Figure 5-18*)

STABILIZER

Fixed surfaces that help stabilize the aircraft in flight are known as stabilizers. On most aircraft they consist of a horizontal and a vertical stabilizer located at the aft end of the fuselage and are known as the empennage. (*Figure 5-19*)

The structure of the stabilizers is similar to that which is used in wing construction. Stabilizers are built using spars, ribs, stringers, and skin like those found in an airplane's wing. They perform the same functions, shaping and supporting the stabilizer and transferring stresses. Bending, torsion, and shear created by air loads in flight pass from one structural member to another. Each member absorbs some of the stress and passes the remainder on towards the others. Ultimately, the spar

Figure 5-16. Fuselage.

Figure 5-17. Tail beam.

Figure 5-18. Main pylon.

Figure 5-19. Horizontal Stabilizer on Light helicopter.

transmits any overloads to the fuselage. The function of the horizontal stabilizer is to prevent a forward tilt as it moves forward. In this situation it receives a lot of stress due to downward aerodynamic force.

UNDERCARRIAGE ATTACHMENTS

The landing gear on a helicopter is mounted under the fuselage. (***Figure 5-20***) It must be strong enough to withstand landing forces when the aircraft is fully loaded. The main landing gear attachment points must be designed to transmit forces throughout the airframe. Longerons and stringers that extend from the structure have this function. In cases of retractable landing gear, wheel wells are needed so that the gear can be retracted to reduce drag. The wheel wells are solid structural boxes, open at the bottom of the fuselage. For the main wheels, these are often additional side boxes due to the presence of fuel tanks in the central structure.

SEAT INSTALLATION

Two different seating systems may be found in the cockpit. Pilot's seats must be secured with two or five-point restraint harnesses. Seats must be adjustable in position for the comfort of the pilot. Passenger seats are generally of a basic type without the possibility of adjustment. They may have two or five attachment points. Despite the possibilities of displacement, the seats must ensure real safety in the event of a hard landing or a crash. In a large helicopter it is also important to have the ability to easily remove all the seats to instead load the fuselage with cargo. (***Figure 5-21***)

DOORS: CONSTRUCTION, MECHANISMS, OPERATION, AND SAFETY DEVICES

There are different types of doors on an aircraft. Operating mechanisms vary depending on the type of door, being cabin entry doors, cargo bay doors, and emergency doors. Aircraft doors are constructed similarly to the fuselage. Vertical and horizontal structural members are skinned on the outside and inside of the door for strength. Latching and locking mechanisms are installed within the structure.

Figure 5-20. Undercarriage.

Figure 5-21. Cargo seat installation.

Cockpit doors are of the plug type. The size of the door is slightly larger than the door opening where the door "plugs" into the fuselage. The door contacts the door frame structure and seals around the entire perimeter of the door.

Cargo doors open outward to clear the door opening for loading and unloading. For strength, large door structural members align with the fuselage frame member. When the door mechanism is moved to CLOSED, locking pins inside the door extend outward from the structural members and engage the fuselage frame structure via holes around the door frame. A mechanism for latching is included. A seal around the door perimeter seals the opening. Proximity sensors and wiring for door position status information are installed. In case of an emergency, cockpit and cargo doors are jettisonable to permit their fast opening and evacuation of the aircraft. (***Figure 5-22***)

Figure 5-22. Cargo door.

On larger helicopters, the cargo doors are often powered for ease of operation. Electric and hydraulic powered door systems exist. A warning system uses proximity switches or sensors to provide an illuminated warning on the flight deck when a door is not closed.

WINDOWS AND WINDSCREEN CONSTRUCTION

The number of windows depends on the size of the helicopter. While it is sometimes possible to increase the number of windows, it is more difficult to increase the window's size due to the weakening of the structure created by replacing the metal skin with glass or plexiglass.

A window is not able to withstand stresses and must be installed between two frames, thus also limiting its size in the fuselage. Fortunately, these constraints are less in the cockpit allowing for a larger window for the pilot.

Window design is based on a solid bottom structure on which the windows are installed between two frames to connect the bottom of the window to the top of the structure. Flight deck windows on transport aircraft are constructed of laminations of tempered glass and plastic. The order and thickness of the laminations vary from aircraft to aircraft, however typically the outer laminations are of glass. Especially for the forward facing windshield, a conductive lamination or embedded conductors are included to electrically heat the window assembly. The heated window is more resistant to impact breakage and also allows for anti-icing. The window laminations are then set in a sealed frame which is bolted onto the fuselage structure.

Fixed and sliding windows are used. Sliding windows are located on the side of the cockpit and fixed windows for passengers and cargo. Some fixed windows have a removable seal to knock out the windows in an emergency.

FUEL STORAGE

The difference between an airplane and a helicopter is the place within the structure where fuel may be stored. As space is not as available in a helicopter, fuel tanks should be centered under the pylon to avoid the center of gravity from becoming unbalanced.

On a small helicopter, the entire crew is in the cockpit and the fuel tank is installed in the central part of the structure. In the case of a truss type structure, a rigid tank is inserted between the bars and suspended inside with a cable tensioner. In a monocoque or semi-monocoque structure, a flexible tank is inserted between the frames. Wet structure fuel tanks, in which the skin of the helicopter itself doubles as a fuel tank, are not used in helicopters due to the many vibrations that will degrade the sealant and create leaks.

On large helicopters, the space under the cargo area will be used to insert multiple flexible tanks to avoid losing all the fuel if there is a leak in one tank. Transfer pumps are used to balance the weight when consuming fuel. Usually, 4 or 6 fuel tanks are divided into two fuel groups. One for the left engine and one for the right. In all cases the engines are above the fuel tanks and the fuel system cannot operate by gravity but instead must be under pressure to send fuel from the tank to the engine.

FIREWALLS

The powerplant firewalls form the support structure for the engine cowlings. A firewall is basically a stainless steel or titanium bulkhead. It protects the crew and equipment from engine heat by limiting the temperature near the hot engine rather than letting it spread throughout the airframe. (***Figure 5-23***)

ENGINE MOUNTS

Engine mounts are also found in the nacelle. These are the structural assemblies to which the engine is fastened. Engine mounts are rigid to accept vibration, temperature, and other stresses.

STRUCTURE ASSEMBLY TECHNIQUES: RIVETING, BOLTING, BONDING

RIVETING, BOLTING

The structures of most operating aircraft are primarily made of aluminum. However, technology in the field of composite materials such as glass and carbon fiber is steadily increasing. A myriad of fasteners are used to join aluminum components. Most common are rivets, nuts and bolts, and a wide variety of special application fasteners. A full explanation of aircraft materials and hardware is found in *Module 06 - Materials and Hardware* from this series.

As early "rag and tube" aircraft were replaced by aluminum construction, rivets dominated assembly techniques. Both light and heavy aircraft today still use the rivet as a primary fastener on both structural and non-structural elements. But as larger and heavier aircraft are designed, structural members increase in size and complexity. Rivets become less suitable to assemble these larger structures. Stronger fasteners, specifically designed for use in aircraft, were introduced. Bolts are used in many locations on aluminum aircraft

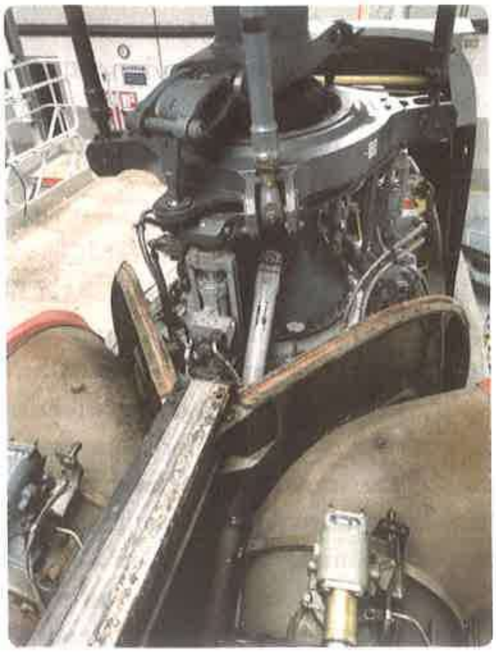


Figure 5-23. Firewalls.

when fastening large structural members, and when attaching both fixed and movable components. Special bolts such as Hi-locks, Jo-bolts and lock-bolts are common, as are clevis bolts where hi-shear loads are present. Close tolerance bolts are used where a tight fit is required. Blind fasteners are used in areas where access to only one side of an assembly is possible. A variety of blind fasteners are used including several which are classified as rivets.

Structural sections and components of the aircraft that are made from composite material may be assembled and attached in a variety of ways. Sleeves and fittings incorporated during construction of a panel often facilitate the use of bolts. Other fasteners may be specified depending on the design and location of the structure. It is of the utmost importance to follow the manufacturer's instructions when assembling composite structures.

BONDING

Many components are bonded or require special fasteners with specific torque considerations. Epoxy sheet bonding using autoclave curing is sometimes used to bond with metal components, resulting in extremely high strength attachments.

Large aircraft maintenance manuals contain specific instructions for the bonding of all materials and sections of the aircraft. The manufacturer's Structural Repair

Manual (SRM) gives a descriptive overview of the aircraft structure and specific rules and procedures to be followed in construction and repair of airframe components and sections.

METHODS OF SURFACE PROTECTION: CHROMATING, ANODIZING, PAINTING, CLADDING

The manufacturer's maintenance manual details the surface protection compounds that must be applied by the technician for each area of the aircraft. Again, ATA Chapter 51 in the maintenance manual and SRM should be consulted. Different areas on the aircraft may be prone to different contaminants and the recommended treatments are designed accordingly. Do not assume that a product is suitable for treatment of any part of an aircraft structure without consulting the manufacturer's data.

The recommended processes of surface treatment are tailored to each material to:

- Control friction and wear.
- Improve corrosion resistance.
- Change a physical property (such as conductivity, resistivity).
- Alter dimensions.
- Vary appearance.
- Improve the functions and/or service lives of the materials.

Common surface treatments can be divided into two categories; those which cover the surface and those which treat or alter the surface.

CHROMATING

An alternative to anodizing for surface protection of magnesium and zinc alloy parts is chromating. Chromating generally involves immersion in a potassium bichromate solution. The chromate coating protects the surface from corrosive elements and has a yellowish appearance on magnesium alloys. Alocrom 1200 is one such chromate product.

ANODIZING

One of the most common surface treatment methods for aluminum based alloys is anodizing. Anodizing is an electrolytic treatment that coats the metal with a hard, waterproof and airtight, oxide film. Anodizing usually contains a dye of various colors which permit an easy identification of a part which has been anodized. As the oxide film acts as an isolator, when attaching a bonding lead, the film must carefully be removed to ensure electrical conductivity. (***Figure 5-24***)

PAINTING

Many aircraft structural elements and parts are painted to protect them from corrosion. The paint acts as a barrier so that the agents of corrosion cannot reach the material being protected. To be effective, paint must be applied to a clean dry surface. It must be compatible with the material composition so that a good bond is formed and properly adheres when applied. Material surface treatments such as paint primers and Alodine are used before painting because they bond strongly to the base material as well as to the paint.

CLADDING

Cladding a corrosion prone material with another non-corrosive material is a popular means of surface protection. This is done as the raw material is formed into the product by the manufacturer. Sheet aluminum may be clad to protect the corrosive copper or zinc alloy from which many aluminum products are made. Aluminum alloy sheets are commonly protected by the application of pure aluminum to both sides of the sheet. In the process, known as Alclad, the pure aluminum is pressure rolled on. The heat generated in that process welds the pure aluminum layer to the sheet. The pure aluminum then forms a stable aluminum oxide surface when exposed to air, which protects the pure aluminum itself and the material which has been clad.

SURFACE CLEANING

Nearly all surface treatments to aircraft metals begin with a thorough cleaning of the material. This may

Figure 5-24. Anodizing protection.

include stripping off old paint before new paint or primer is applied. Strippers are specifically recommended by the manufacturer that do not react with the base metal of the structure. Only use strippers that are recommended by the manufacturer. A cleaned surface is then often treated with Alodine before a primer and new paint coating is applied. Clad aluminum parts use a different formula of Alodine than non-clad alloys. Again, be sure to use the correct formula.

Personal safety procedures should be followed when cleaning, stripping and applying any surface treatment. Solvents, strippers, cleaners, and conversion coatings can all be hazardous to the health of the technician. Avoid breathing vapors from products of this type and avoid prolonged skin contact. Use protective gloves, goggles, respirators and other protective gear. Always ensure adequate ventilation in the work area when using any chemicals, and if questionable wear a ventilator. Know the location of the nearest eyewash fountain when working with these substances. If accidentally splashed in your eyes, flush with water and get medical attention immediately. Generally, specified paint strippers are used on metal surfaces only. Protect all surrounding areas from accidental contact with the stripper. Polyethylene film and suitable adhesive tape is used for masking.

Teflon lines, self-lubricated bearings, electrical terminal plugs, nylon coated wires and nylon bushings should particularly be protected from contact with chemicals used in strippers. Plastics, laminates, composites, fiberglass, and bonded structures usually have paint removed by abrasive cleaning. Do not use strippers on composite structures. Use only the methods described by the manufacturer.

EXTERIOR AIRCRAFT CLEANING

The manufacturer's maintenance manual gives detailed instructions on cleaning procedures. Areas to be protected and the proper cleaning agents to use must be noted.

Aircraft are always cleaned before major inspections. Typically, high-pressure water or steam is sprayed in conjunction with cleaning agents to the exterior of the aircraft. While a clean aircraft aids in corrosion prevention, the cleaning process may put water and agents where they are not desirable and thus themselves may cause corrosion. Areas of the aircraft into which the cleaning spray should not enter such as pitot tubes, static ports as well as tires and brake assemblies must be covered or sealed. Aircraft are generally washed outside in an area with adequate and environmentally responsible drainage. Washing with cleaning agents should not be used in high temperatures where the agent may dry before being rinsed off. In certain locations, this may require washing to occur inside of a hangar.

Use the ratio of agent to water that is recommended. Use of the wrong agent may cause the agent to attack materials. Hydrogen embrittlement occurs when certain agents soak into an aircraft's metal. Minute cracks form and stress corrosion develops. Engine and wheel well areas may require a special washing technique or cleaning agents due to dirt, oil, grease, and exhaust buildup. Be aware that some cleaning procedures are followed by greasing various locations that may have had grease washed out during the cleaning process. Again, follow the manufacturer's instructions.

Snow And Ice

Chemical salts and other melting agents are often used during the winter months. This slush will inevitably become splashed or sprayed onto the aircraft and could be detrimental. The contaminated areas should be washed down with clean water as soon as possible after exposure. If needed, a wetting agent may be added.

Salt Air Operating Environment

Operations in salt air environments will be more susceptible to salt deposits and salt contaminant corrosion. Helicopter cleaning and protection programs should be tailored accordingly to these operating environments and may include increased frequency of washing and lubrication procedures, engine compressor washing.

Acrylic Windows

Acrylic windows should be washed with soap or a mild detergent in warm water. Polishing minor scratched surfaces may be accomplished with an approved plastics polish and finished with an anti-static polish or cloth.

AIRFRAME SYMMETRY: METHODS OF ALIGNMENT AND SYMMETRY CHECKS

A symmetry check is carried out on a helicopter structure primarily to check whether any distortion has taken place, especially regarding the tail boom and the horizontal and/or vertical stabilizer. This may be carried out after an overstress event such as a heavy landing. A symmetry check may also be carried out after replacement or repair of a major structure.

After leveling the helicopter to place it in a reference position, the mechanic will use a theodolite (otherwise known as a transit) to check that the tail is properly aligned with the fuselage. The fuselage is a strong structure and the risk of distortion is low. The tail which is fragile and flexible can be misaligned. The only solution is to check if the alignment remains within tolerance or not. If not, the tail must be changed. Two controls must be done; its lateral position and its horizontal position.

LATERAL CONTROL OF THE TAIL POSITION

The tail rotor tends to pull the tail to oppose the torque reaction of the fuselage. This reverse movement between the fuselage and tail creates a lot of stress on the junction and so it is important to frequently check the tolerances. While it is often difficult to align the tail center exactly with the fuselage centerline, they must be roughly on the same axis. After jacking and leveling the helicopter, the method is to draw the fuselage centerline on the ground with two reference points referred to on the center structure. (***Figure 5-25A and B***)

The next step is to project the reference point of the rear beam C onto the ground and check if this point is within the tolerances. If the point is out of tolerance, the only solution is to change the rear beam. (***Figure 5-26 and Figure 5-27***) At this point, it is not possible to adjust this assembly.

HORIZONTAL CONTROL OF THE TAIL POSITION

The aim is to check whether the tail beam has moved since its original connection between the fuselage and the beam. Three movements are possible in this control:

- In the event of a hard landing, due to the inertia movement, the tail moved downward.
- In the event of a hard landing the tail touches the ground and it is raised.
- Due to a high position of the tail rotor, the rear beam tends to twist.

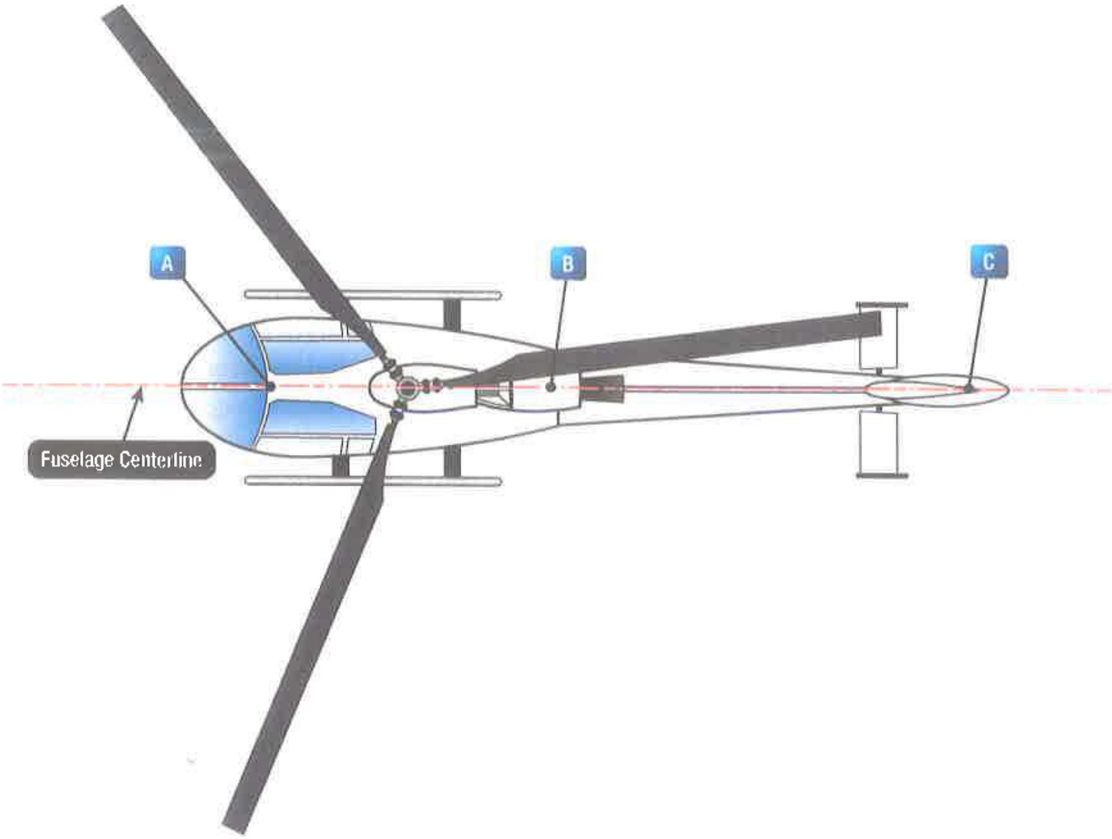


Figure 5-25. Center line definition.

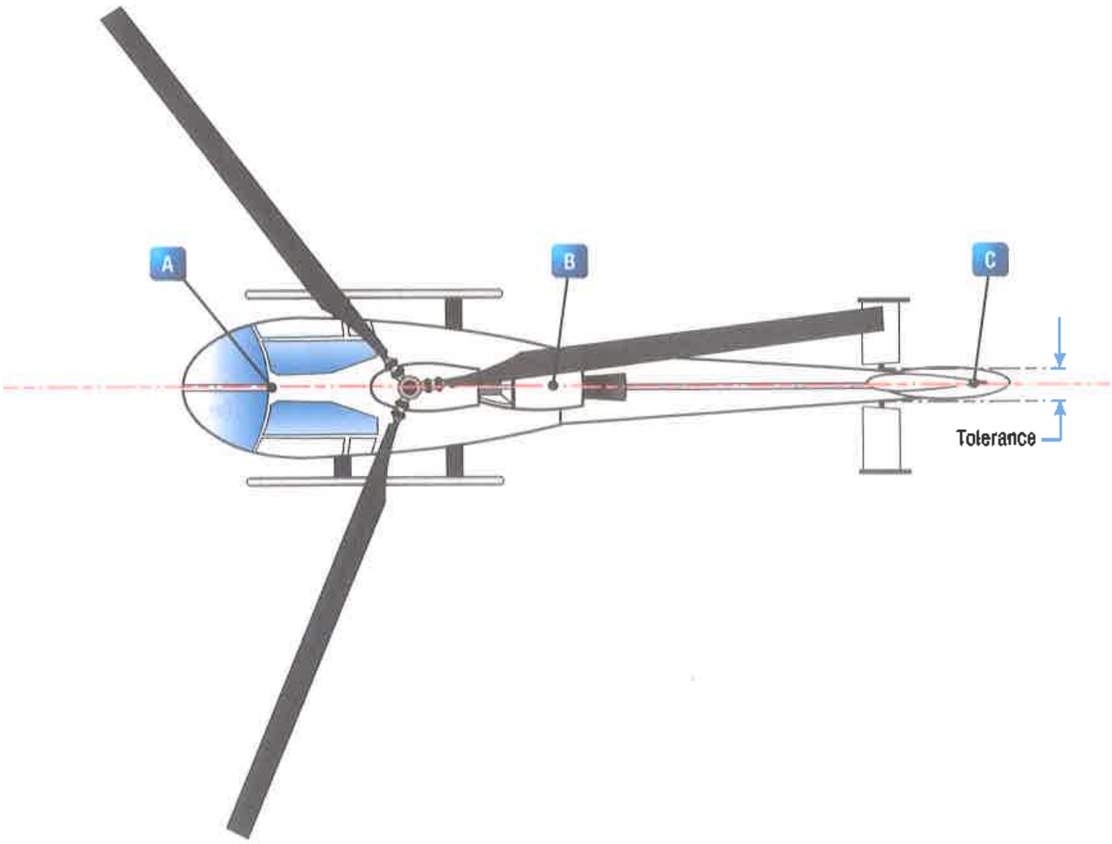


Figure 5-26. Lateral measurement in tolerance.

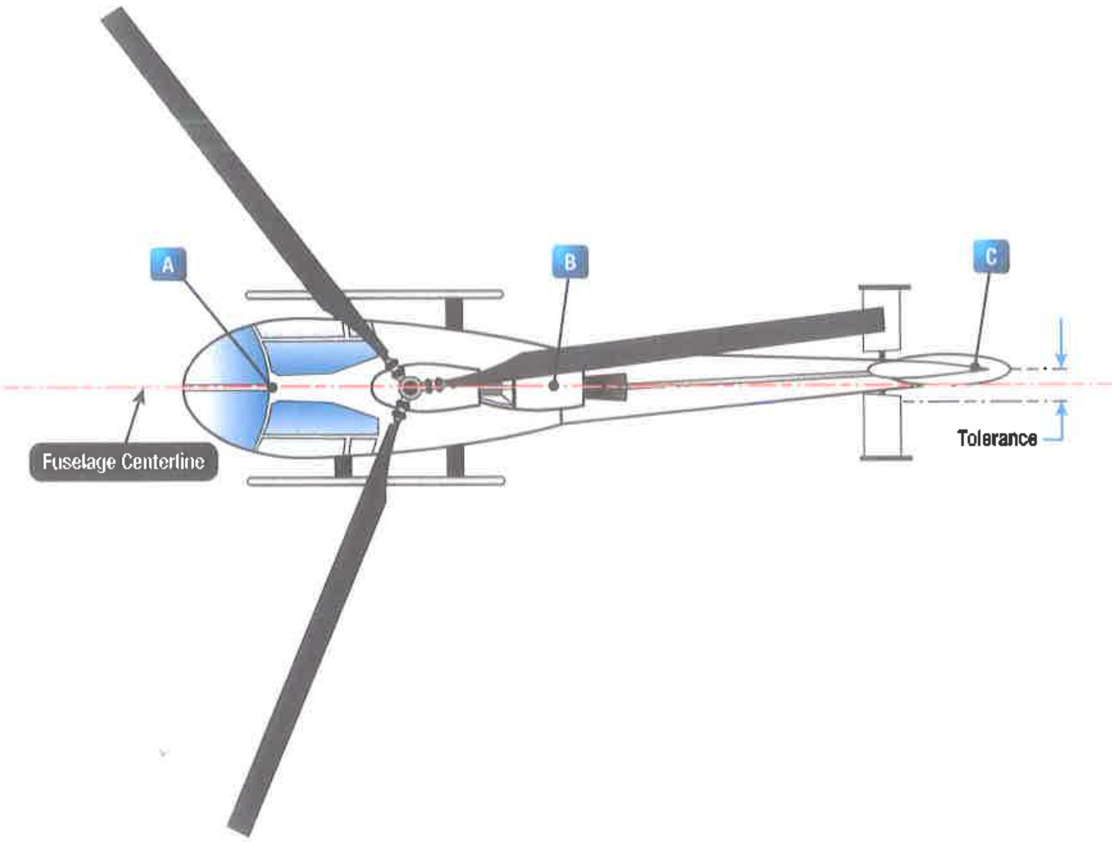


Figure 5-27. Lateral measurement out of tolerance.

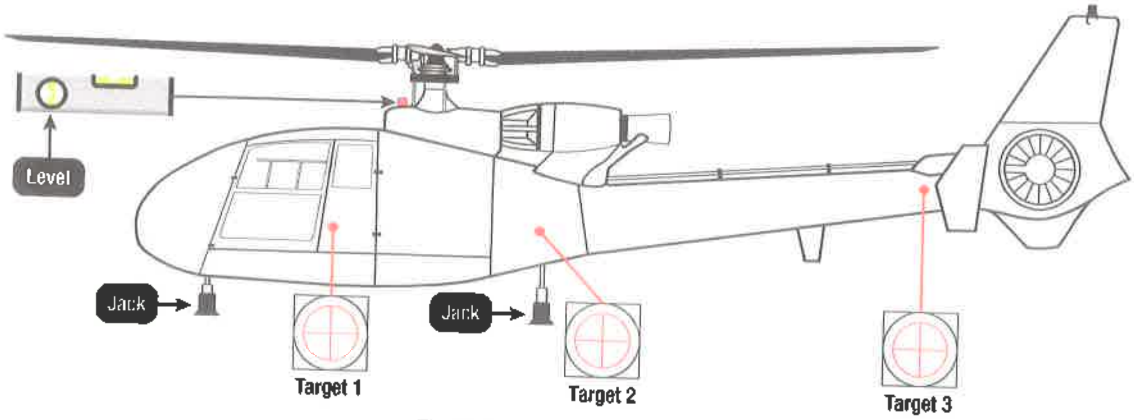


Figure 5-28. Target description.

To control tail sagging, a horizontal inspection should be performed. First, the helicopter is installed on jacks and raised to a position as specified by the manufacturer's reference lines. While doing so, the helicopter must be continuously balanced in its roll axis with a level. (*Figure 5-28 and Figure 5-29*)

Two target points are located on the fuselage to define the reference plane (T1 and T2), and one on the rear beam (T3) to measure the distance from the plane.

The next step is to adjust the length of the front jacks to position the helicopter's red cross on the blue one that represents the referenced line. The two crosses must be perfectly superimposed. (*Figure 5-30*) The "roll" should again be checked with the level to confirm the correct position.

After this adjustment, the second target needs to be adjusted. In (*Figure 5-31*) we can see that the target is below the reference plane, and we need to move the helicopter upwards with the rear jack. After this adjustment the two targets (blue and red) are perfectly superimposed. (*Figure 5-32*) The fuselage horizontal line is now defined and this line becomes the reference.


Figure 5-29. Target picture.

The last step is to measure the distance between the rear target and the reference point defined by the theodolite through the two fuselage targets with the "roll" axis in its horizontal position. (*Figure 5-33*) The measurement must again be carried out on the right side under the same conditions as those previously explained.

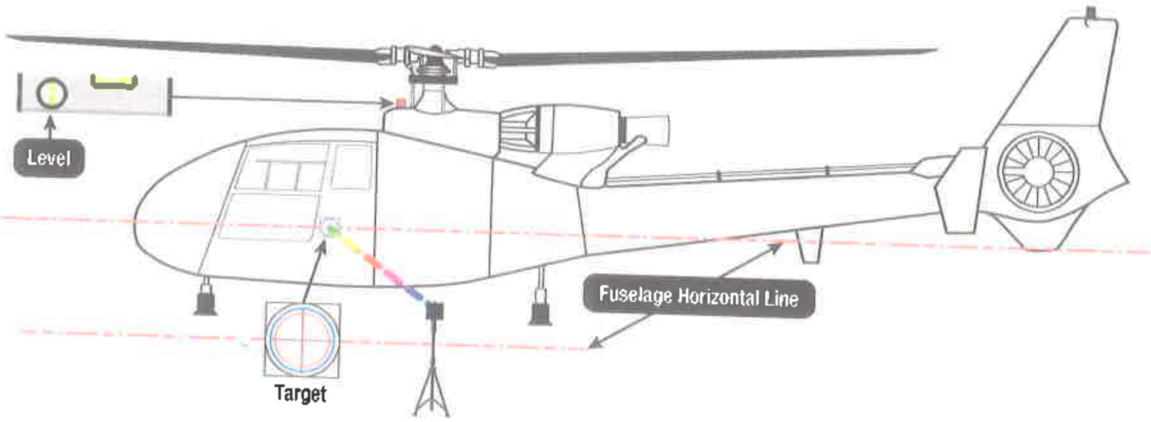


Figure 5-30. Front adjustment.

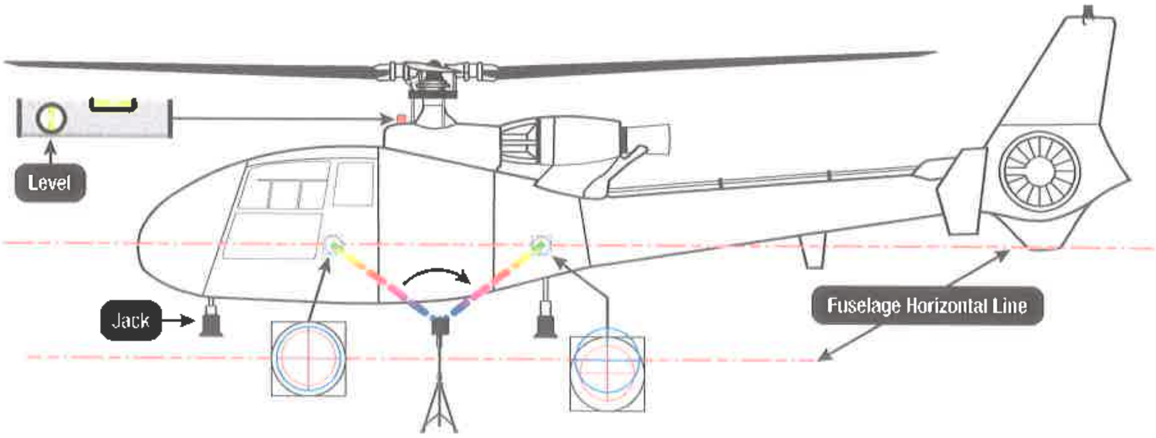


Figure 5-31. Rear adjustment 1-2.

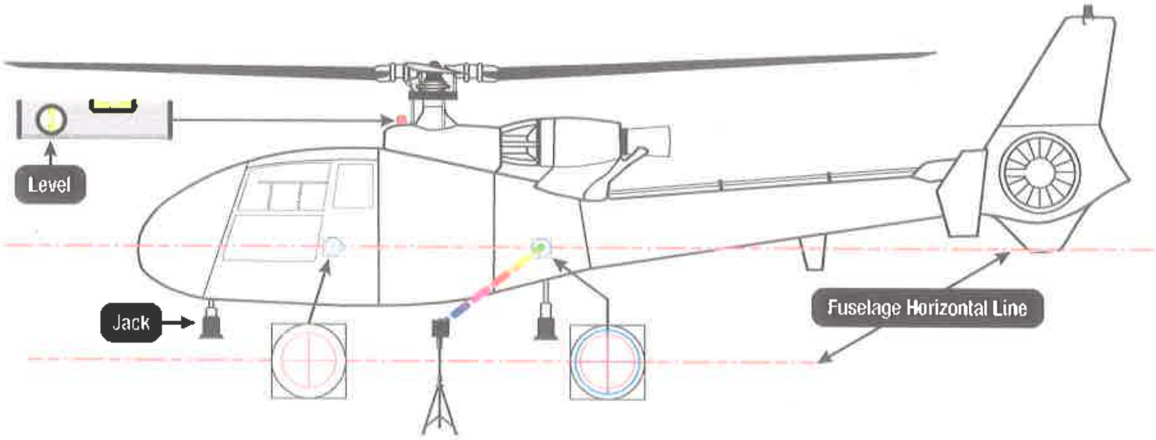


Figure 5-32. Rear adjustment 2-2.

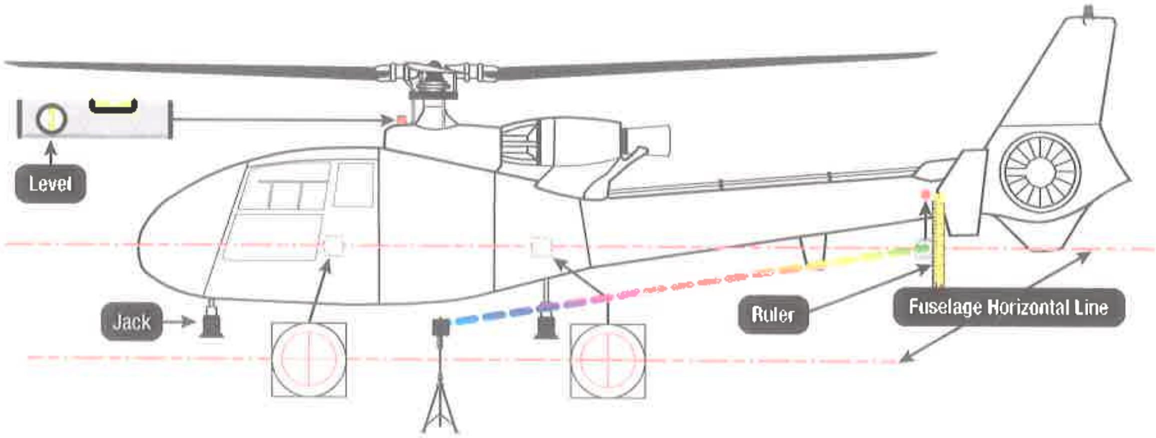


Figure 5-33. Measurement reading.

With this competed, we obtain two values:

- The left distance between the target 3 and the reference horizontal line.
- The right distance between the target 3 and the reference horizontal line.

When the manufacturer installs the rear beam on the fuselage, they measure the same distance between target 3 and the horizontal plane to give the reference value from which all future measurements can be analyzed.

Three causes are possible for inaccurate reference values:

- The right and left values are smaller than the manufacturer reference value: The tail beam is sagged.
- The right and left values are bigger than the manufacturer's reference value: The tail touched the ground and moved upward.
- One value is bigger and the other smaller than the manufacturer's reference value: The tail is twisted.

In each case, certain tolerances are accepted based upon the manufacturer's value and are written in the documentation.

Question: 5-1
Describe what type of helicopter is covered by Certification Specification 29 (CS 29) category A.

Question: 5-2
If a helicopter suffers a hard landing, what is the primary structural concern which must be checked?

Question: 5-3
Which safety design concept depends on the predicted integrity of a single component?

Question: 5-4
Which components of a helicopter are identified under the category ATA 700?

Question: 5-5
In what way is an electrical bond tested for its integrity?

Question: 5-6
Which type of fuselage construction relies both on stressed skin and an internal structure?

Question: 5-7
What is the greatest problem and its correction regarding composite structures?

Question: 5-8
What is the primary limitation regarding the number of windows which may be installed in a helicopter?

ANSWERS

Answer: 5-1
Large helicopters, with a maximum weight over 9 072 Kg and 10 or more passenger seats.

Answer: 5-2
The proper alignment of the tail boom.

Answer: 5-3
The Safe-Life concept assumes that a specific structural element will not fail within its predicted lifetime.

Answer: 5-4
Landing gear components.

Answer: 5-5
With an ohm meter. The electrical resistance from the mount to the airframe should not exceed 0.1 ohms.

Answer: 5-6
Semi-monocoque. A monocoque structure relies only on the stressed skin.

Answer: 5-7
Composites are not electrically conductive. Thus for bonding and lightning protection, conductive wires or material must be embedded into the composite matrix.

Answer: 5-8
The cutout area in the fuselage for each window weakens the entirety of the structure.

HELICOPTER AERODYNAMICS, STRUCTURES AND SYSTEMS

AIR CONDITIONING (ATA 21)

SUB-MODULE 06

PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY → B1.3 B1.4

Sub-Module 06

AIR CONDITIONING (ATA 21)

Knowledge Requirements

12.6 - Air Conditioning (ATA 21)

12.6.1 - Air Supply	
Sources of air supply including engine bleed and ground cart.	2
12.6.2 - Air Conditioning	
Air conditioning systems;	3
Distribution systems;	
Flow and temperature control systems;	
Protection and warning devices.	

12.6 - AIR CONDITIONING (ATA 21)

12.6.1 - AIR SUPPLY

SOURCES OF AIR SUPPLY INCLUDING ENGINE BLEED AND GROUND CART

The function of a helicopter is not to fly high, and so at these lower altitudes it is not necessary to pressurize the cockpit. However, air conditioning is used in many higher performance helicopters, not just for the comfort of the crew and passengers, but also to cool down the electronic elements which increase in complexity and demands with each new generation.

CERTIFICATION SPECIFICATIONS

The EASA certification specification CS 29.831 Ventilation explains these conditions requiring solutions for the aircraft.

Extract from EASA CS 29.831 Ventilation

a. Each passenger and crew compartment must be ventilated, and each crew compartment must have enough fresh air (but not less than 0.3 m^3 (10 cu ft) per minute per crew member) to let crew members perform their duties without undue discomfort or fatigue.
b. Crew and passenger compartment air must be free from harmful or hazardous concentrations of gases or vapors.
c. The concentration of carbon monoxide may not exceed one part in 20 000 parts of air during forward flight. If the concentration exceeds this value under other conditions, there must be suitable operating restrictions.
d. There must be means to ensure compliance with sub-paragraphs (b) and (c) under any reasonably probable failure of any ventilating, heating, or other system or equipment.

RAM AIR

The first source of air is outside air captured by dynamic frontal intakes. When the helicopter is in motion, fresh air enters through these ducts and renews the air in the cockpit. (***Figure 6-1***) To increase the flow, an electric fan can be added. The temperature in the cockpit is then the same as the outside temperature. This method is used in some small unpressurized helicopters to supply air to either a combustion heater or to an exhaust heat exchanger. It is located at the nose of the helicopter, or at the side of the fuselage.

BLEED AIR

A pneumatic system uses engine and APU bleed air as a high or low pressure source for the air-conditioning. The bleed air is divided into two categories, engine bleed air and APU bleed air.

Bleed Air From An Engine

One significant hot air source comes from the turbine engine. The main operating principle of a turbine engine involves the compression of large amounts of air, mixed with fuel and burned. Air bled from the compressor section of the engine is relatively free of contaminants. As such, compressor air is a great source of air conditioning. However, the volume of air for engine power production is reduced by bleeding some from the compressor. Even though the amount of bleed air compared to the overall amount of air compressed for combustion is relatively small, it should still be minimized.

Bleed Air From An APU

The source of air for air conditioning does not always have to be bleed air from the main engines. When the helicopter is on the ground, bleed air from the Auxiliary Power Unit (APU) can be used to operate the air-conditioning system. The aircraft's APU is designed to deliver pressurized bleed air for engine starting and for operation of the aircraft air conditioning packages. Sufficient pneumatic air from the APU is routed through ducting to the air conditioning packs, so that the cabin can be cooled while the aircraft is on the ground and the engine not running. This is one common method for keeping the cabin at a comfortable temperature.

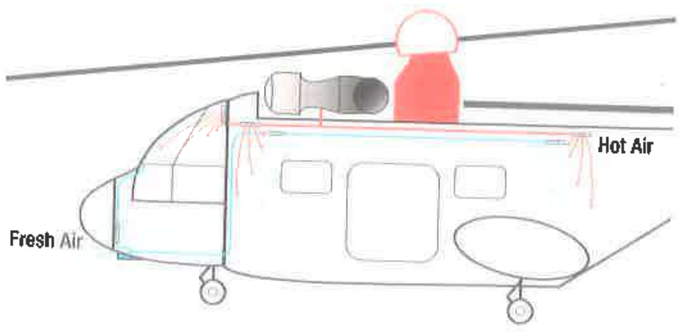


Figure 6-1. Air supply.

GROUND CART

When an aircraft is on the ground, operating the engines or the APU for providing air for air conditioning is expensive. It increases the time in service of these expensive components and expedites costly overhauls that must be performed at specified time intervals. Often, a ground cart is available to provide the pressurized air for the air conditioning packs. This is typically a portable powerplant that drives a high volume compressor. The cart is towed to the aircraft's location and is connected into the aircraft pneumatic system with a hose. The connecting point is upstream of the air conditioning packs. Cart air is regulated to the normal pneumatic system pressure and can also be used for pneumatic system troubleshooting without the expense of running the APU or the main engines.

HEATED AIR SOURCES

If the incoming air temperature from outside ducts needs to be increased it can be done so by mixing with hot air coming from different sections. The extent of heating will have different requirements depending on which of these functions are in need:

- To heat the cockpit.
- To ventilate the equipment.
- To ventilate the cockpit.
- To demist the windshield.

Heaters

It is possible to use a heater operating with an electric resistor, or to use a small combustion chamber dedicated to air heating. In these systems, a valve controls the flow of hot air, to mix it with cold air. The lowest temperature comes from the outside. To increase it the valve is opened and as much hot air as needed is added to get the right adjusted temperature.

Heat Exchangers

Another solution is to use a heat exchanger along with a heat source to heat the ambient air before it is channeled into the cockpit. This system makes it possible to use the hot exhaust gases to increase the temperature, without sending toxic exhaust gasses directly into the cockpit. This system is used on piston-engine helicopters and on most single-engine helicopters. It is controlled by means of a simple cable operated flapper valve. (***Figure 6-2***)

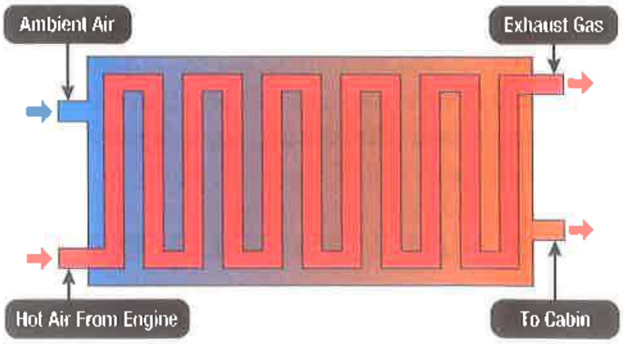


Figure 6-2. Heat exchanger.

Combustion Heaters

A combustion heating system is an airtight burner using fuel from the helicopter. Fuel is ignited and burned to provide heat. Ventilation air is forced over the chamber picking up heat, which is then dispersed into the cabin area.

This method of heat is very safe. An overheat switch is provided on all combustion heaters, which is wired into the heater electrical system to shut off the fuel in the case of malfunction. In the unlikely event that the heater fuel solenoid remains open or the control switches fail, the remote fuel solenoid and/or fuel pump is shut off by the mechanical overheat switch, thus stopping all fuel flow to the system.

Unlike heat exchangers, combustion heaters do not regard carbon monoxide poisoning as a major concern. Combustion heaters have low pressure in the combustion tube, which is vented through its exhaust into the air stream. The ventilation air on the outside of the combustion chamber is of higher pressure than inside, and ram air or pressurization increases the pressure on the outside of the combustion tube. In the event of a leak in the combustion chamber, the higher pressure air outside the chamber would travel into the chamber and out the exhaust.

HUMIDITY CONTROL

The dangers for electronic systems do not just involve high or low temperatures. Another parameter which needs to be simultaneously controlled is humidity. Humidity, also called "relative humidity," refers to the amount of water vapor contained in the atmosphere and is expressed as a percentage of the maximum amount of vapor the air can hold. This amount varies with temperature. Warm air can hold more water vapor, while colder air can hold less. The air distribution system participates in the humidity control, or at least by dehumidification by ventilation (such as demisting windshields). This is also one of the purposes of an air conditioning system where water vapor removal is done by various means.

12.6.2 - AIR CONDITIONING

AIR CONDITIONING SYSTEMS

The purpose of the air conditioning system is to control the temperature, airflow and humidity within the helicopter cabin. On some helicopters, there are separate air conditioning systems for the cockpit and the cabin. Once thought to be a luxury item, air conditioning systems are becoming mission essential regarding cabin comfort and air crew performance, particularly for Helicopter Emergency Medical Service operators. Two types of air conditioning systems are commonly used on aircraft:

- Vapor cycle systems use ram air and are found on all reciprocating engine powered helicopters and some small to medium sized turbine powered helicopters. This type of system is like those found in houses and cars.
- Air cycle systems use engine bleed air or APU pneumatic air during the conditioning process and are found on most turbine powered larger helicopters.

VAPOR CYCLE AIR CONDITIONING

Energy can be neither created nor destroyed; however, it can be transformed and moved. This is what occurs during vapor cycle air conditioning. Heat energy is moved from the cabin air into a liquid refrigerant. Due to the additional energy of the heat, the liquid changes into vapor. The vapor is compressed and becomes extremely hot. It is then removed from the cabin where the hot vapor refrigerant transfers its heat energy to the outside air. In doing so, the refrigerant cools, condenses back into a liquid and then returns to the cabin to repeat the cycle of energy transfer. (***Figure 6-3***)

Heat is an expression of energy which is typically measured by temperature. The higher the temperature of a substance, the more energy it contains. Heat always flows from hot to cold. This term expresses the relative amount of energy present in a substance. Without a difference in energy levels, there is no transfer of energy (heat).

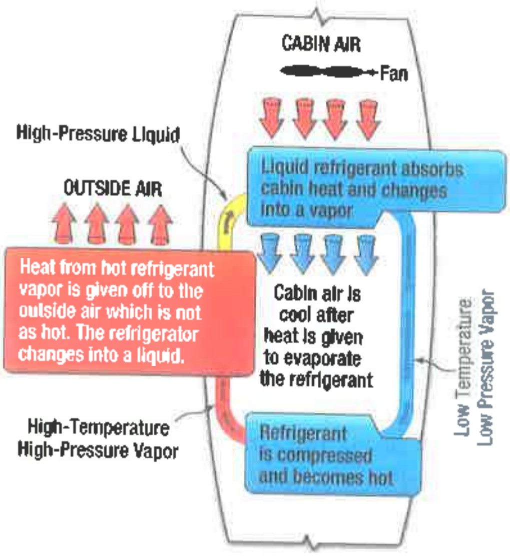


Figure 6-3. Refrigerant cycle.

Adding heat to a substance does not always raise its temperature. When a substance changes its state, such as when a liquid changes into vapor, heat energy is absorbed. This is called latent heat. When vapor condenses into a liquid, this heat energy is given off. The temperature of a substance remains constant during its change of state. All the energy which is absorbed or given off (the latent heat) is used for the change of state process. Once the change of state is complete, heat added to a substance raises the temperature of the substance. After a substance changes state into vapor, the rise in temperature of the vapor caused by the addition of still more heat, is called superheat.

The temperature at which a substance changes from a liquid into vapor when heat is added, is known as its boiling point. This is the same temperature at which vapor condenses into a liquid when heat is removed. The boiling point of any substance varies directly with pressure. When pressure on a liquid is increased, its boiling point increases, and when pressure on a liquid is decreased, its boiling point also decreases.

For example, water boils at 100°C at a normal atmospheric temperature (14.7 psi). When pressure on liquid water is increased to 20 psi, it does not boil at 100°C. More energy is required to overcome the increase in pressure and so the boiling point becomes approximately 103°C. The opposite is also true. Water can also boil at a much lower temperature simply by reducing the pressure upon it. With only 10 psi of pressure upon liquid water, it boils at 90°C. (***Figure 6-4***)

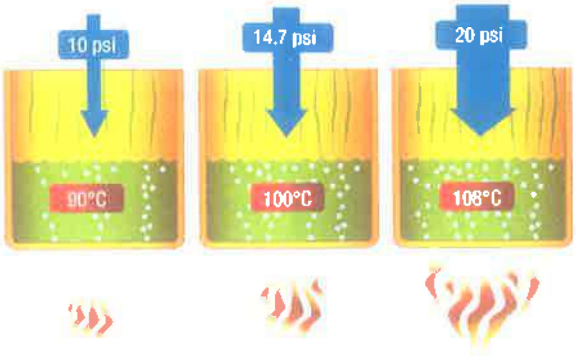


Figure 6-4. Boiling Point.

Vapor pressure is the pressure of the vapor that exists above a liquid that is in an enclosed container at any given temperature. The vapor pressure developed by various substances is unique to each substance. A substance that is said to be volatile, develops high vapor pressure at the standard day temperature of 15°C. This is because the boiling point of the substance is much lower. The boiling point of tetrafluoroethane (R134a), a refrigerant used in most aircraft vapor cycle air conditioning systems, is approximately -26°C. Its vapor pressure at 15°C is about 71 psi. The vapor pressure of any substance varies directly with temperature.

BASIC VAPOR CYCLE

Vapor cycle air conditioning is a closed system in which a refrigerant is circulated through tubing and a variety of components. The purpose is to remove heat from the aircraft cabin. While circulating, the refrigerant changes state. By manipulating the latent heat required to do so, hot air is replaced with cool air in the aircraft cabin. To begin, R134a is filtered and stored under pressure in a reservoir known as a receiver dryer. The refrigerant is in liquid form. It flows through tubing from the receiver dryer to an expansion valve. Inside the valve, a restriction in the form of a small orifice blocks most of the refrigerant. Since it is under pressure, some of the refrigerant is forced through the orifice. It emerges as a spray of tiny droplets in the tubing downstream of the valve. The tubing is coiled into a radiator type assembly known as an evaporator. A fan is positioned to blow cabin air over the surface of the evaporator. As it does, the heat in the cabin air is absorbed by the refrigerant, which uses it to change state from a liquid to a vapor. So much heat is absorbed that the cabin air blown by the fan across the evaporator cools significantly. This is the vapor cycle conditioned air that lowers the temperature in the cabin. The gaseous refrigerant exiting the evaporator is drawn into a compressor. There, the pressure and the temperature of the refrigerant are increased. The high pressure high temperature gaseous refrigerant flows through tubing to a condenser. The condenser is like a radiator comprised of a great length of tubing with fins attached to promote heat transfer. Outside air is directed over the condenser. The temperature of the refrigerant inside is higher than the ambient air temperature so heat is transferred from the refrigerant to the outside air. The amount of heat given off is enough to cool the refrigerant, and to condense it back to a high pressure liquid. It flows through tubing and back into the receiver dryer, completing the vapor cycle. There are two sides to the vapor cycle air conditioning system. One accepts heat and is known as the low side. The other gives up heat and is known as the high side. Low and high refer to the temperature and pressure of the refrigerant. As such, the compressor and the expansion valve are the two components that separate the low side from the high side of the cycle. (***Figure 6-5***)

Refrigerant on the low side is characterized as having low pressure and temperature. Refrigerant on the high side has having high pressure and temperature.

VAPOR CYCLE AIR CONDITIONING SYSTEM COMPONENTS

By examining each component in the vapor cycle air conditioning system, greater insight into its function can be gained.

Refrigerant

For many years, dichlorodifluoromethane (R12) was the standard refrigerant used in aircraft vapor cycle air conditioning systems. Some of these systems remain in use today. R12 was found to have a negative effect on the environment; in particular by degrading the protective ozone layer. In most cases, R12 has been replaced by tetrafluoroethane (R134a), which is safer for the environment. R12 and R134a should not be mixed, nor should one be used in a system designed for the other. Possible damage to soft components such as hoses and seals could result causing leaks and malfunction. Use only the specified refrigerant when servicing vapor cycle air conditioning systems. (***Figure 6-6***) R12 and R134a behave so similarly that the descriptions of the R134a vapor cycle air conditioning system and components in the following paragraphs also apply to an R12 system and its components.

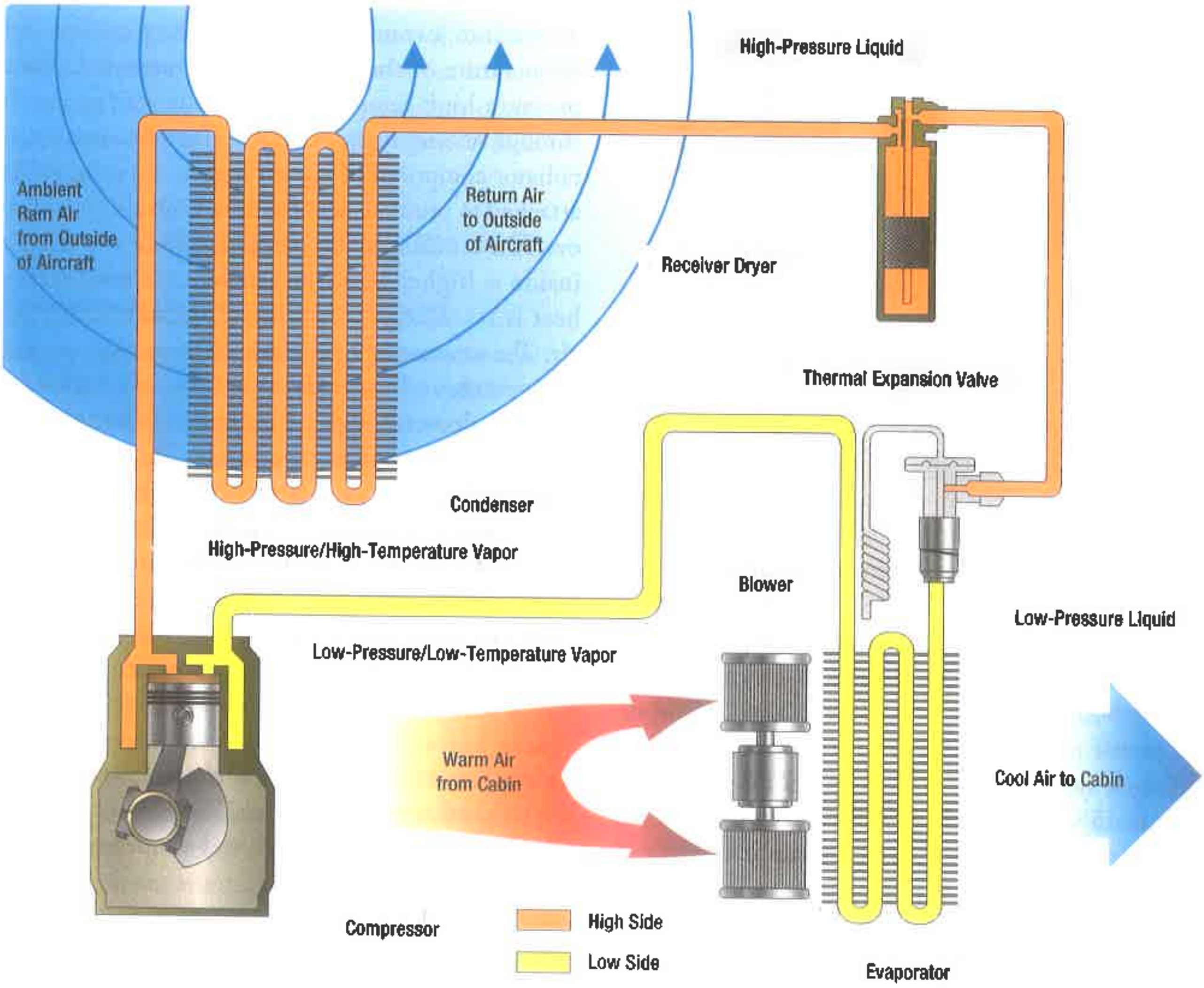


Figure 6-5. Basic vapor cycle system.

Figure 6-6. Cans of R134a.

R134a is a halogen compound (CF3CFH2). It has a boiling point of approximately -26°C. It is not poisonous to inhale in small quantities, but it does displace oxygen. Suffocation is possible if breathed in mass quantities. Regardless of the manufacturer, refrigerants are sometimes called Freon which is a trade name of the DuPont Company.

Caution should be used when handling any refrigerant. Because of the low boiling points, liquid refrigerants boil violently at typical atmospheric temperatures and pressures. They rapidly absorb heat energy from all surrounding matter. If a drop lands on skin, it freezes resulting in a burn. Similar tissue damage can result if a drop gets in one's eye. Gloves and other skin protection, as well as safety goggles, are required when working with refrigerants.

Receiver Dryer

The receiver dryer acts as the liquid refrigerant reservoir of the vapor cycle system. It contains a desiccant that absorbs any moisture that may be in the system and is located downstream of the condenser and upstream of the expansion valve. When it is extremely hot, more refrigerant is used by the system than when temperatures are moderate. Extra refrigerant is stored in the receiver dryer for this purpose. Liquid refrigerant from the condenser flows into the receiver dryer. Inside, it passes through filters and a desiccant material. The filters remove any foreign particles that might be in the system. The desiccant captures any water in the refrigerant. Water in the refrigerant causes two major problems. First, the refrigerant and water combine to form an acid. If left in contact with the inside of the components and tubing, the acid deteriorates the materials from which these are made. The second problem with water is that it could form ice and block the flow of refrigerant around the system, rendering it inoperative. Ice is particularly a problem if it forms at the orifice in the expansion valve which is the coldest point in the cycle. Occasionally, vapor may find its way into the receiver dryer, such as when the gaseous refrigerant does not completely change state to a liquid in the condenser. A stand tube is used to remove refrigerant from the receiver dryer. It runs to the bottom of the unit to ensure liquid is withdrawn and forwarded to the expansion valve. At the top of the stand tube, a sight glass allows the technician to see the refrigerant. When enough refrigerant is present in the system, liquid flows in the sight glass. If low on refrigerant, any vapor present in the receiver dryer may be sucked up the stand tube causing bubbles to be visible in the sight glass. Therefore, bubbles in the sight glass indicate that the system needs to have more refrigerant added. (***Figure 6-7***)

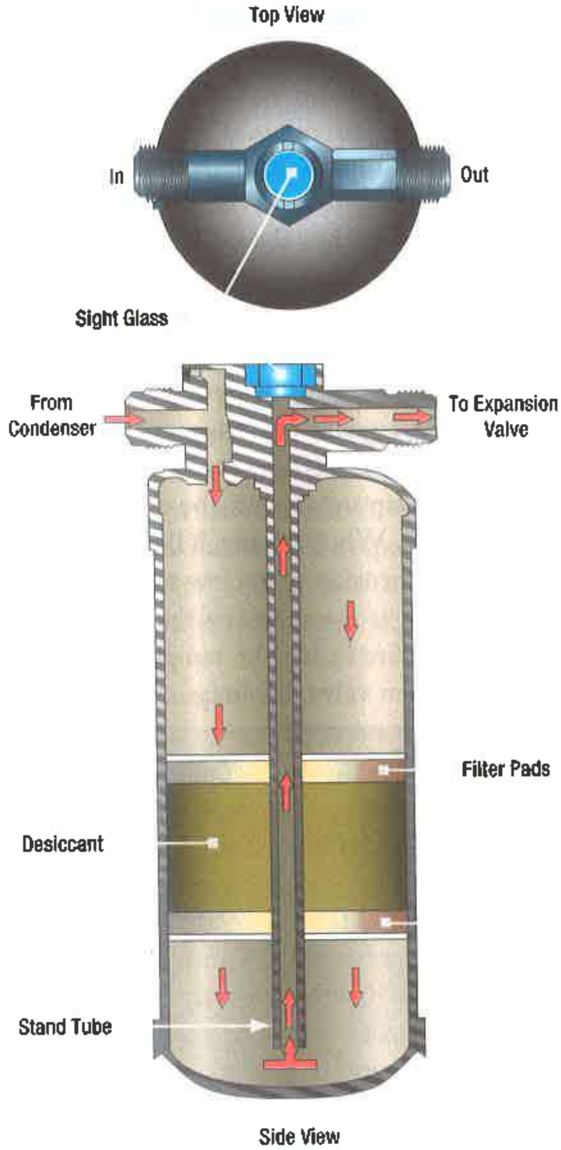


Figure 6-7. Receiver dryer.

Thermal Expansion Valve

The thermal expansion valve regulates the amount of liquid refrigerant flowing into the evaporator. Refrigerant exits the receiver dryer and flows to the expansion valve. The thermostatic expansion valve has an adjustable orifice through which the correct amount of refrigerant is metered to obtain optimal cooling. This is accomplished by monitoring the temperature of the gaseous refrigerant at the outlet of the next component in the cycle, the evaporator. Ideally, the expansion valve should only let the amount of refrigerant spray into the evaporator that can be completely converted to a vapor.

The temperature of the cabin air to be cooled determines the amount of refrigerant the expansion valve should spray into the evaporator. Only so much is needed to completely change the state of the refrigerant from a liquid to a vapor. Too little causes the gaseous refrigerant to be superheated by the time it exits the evaporator. This is inefficient. Changing the state of the refrigerant from liquid to vapor absorbs much more heat than adding heat to already converted vapor (superheated). The cabin air blowing over the evaporator will not be cooled sufficiently if superheated vapor is flowing through the evaporator. If too much refrigerant is released by the expansion valve into the evaporator, some of it remains liquid when it exits the evaporator. Since it flows next to the compressor, this could be dangerous.

The compressor is designed to compress vapor only. If liquid is drawn in, and attempts are made to compress it, the compressor could break, since liquids are essentially incompressible. The temperature of superheated vapor is higher than the temperature of liquid refrigerant that has not totally vaporized. A coiled capillary tube with a volatile substance inside is located at the evaporator outlet to sense this difference. Its internal pressure increases and decreases as temperature changes. The coiled end of the tube is closed and attached to the evaporator outlet.

The other end terminates in the area above a pressure diaphragm in the expansion valve. When superheated refrigerant vapor reaches the coiled end of the tube, its elevated temperature increases the pressure inside the tube and in the space above the diaphragm. This increase in pressure causes the diaphragm to overcome spring tension in the valve. It positions a needle valve that increases the amount of refrigerant released by the valve. The quantity of refrigerant is increased so that the refrigerant evaporates, and the refrigerant vapor does not superheat. When too much liquid refrigerant is released by the expansion valve, low temperature liquid refrigerant arrives at the outlet of the evaporator. The result is low pressure inside the temperature bulb and above the expansion valve diaphragm. The superheated spring in the valve moves the needle valve toward the closed position, reducing the flow of refrigerant into the evaporator as the spring overcomes the lower pressure above the diaphragm. (***Figure 6-8***)

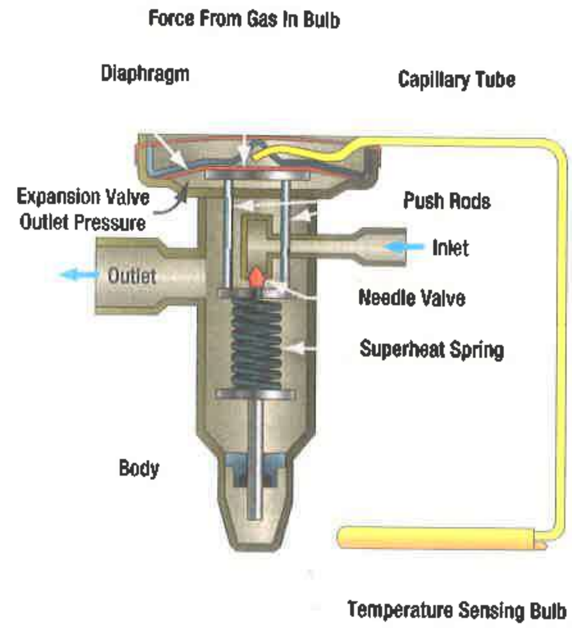


Figure 6-8. Expansion valve - Outlet pressure.

Vapor cycle air conditioning systems have large evaporators and experience significant pressure drops while refrigerant is flowing through them. Externally, equalized expansion valves use a pressure tap from the outlet of the evaporator, to help the superheated spring balance the diaphragm. This type of expansion valve is easily recognizable by the additional small diameter line that comes from the evaporator into the valve (2 total). Better control of the proper amount of refrigerant allowed through the valve is attained by considering both the temperature and pressure of the evaporator refrigerant. (***Figure 6-9***)

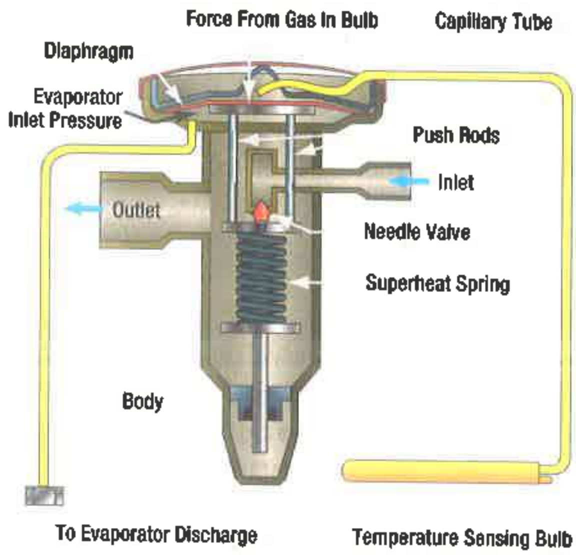


Figure 6-9. Expansion valve - Inlet pressure.

Evaporator

The evaporator absorbs the heat of the helicopter cabin into the refrigerant. As the heat is absorbed the refrigerant evaporates. Most evaporators are constructed of copper or aluminum tubing coiled into a compact unit. Fins are attached to increase surface area, facilitating rapid heat transfer between the cabin air blown over the outside of the evaporator with a fan and the refrigerant inside. The expansion valve located at the evaporator inlet releases high pressure high temperature liquid refrigerant into the evaporator. As the refrigerant absorbs heat from the cabin air, it changes into a low pressure vapor. This is discharged from the evaporator outlet to the next component in the vapor cycle system, the compressor. The temperature and pressure pickups that regulate the expansion valve are located at the evaporator outlet. The evaporator is situated in such a way that cabin air is pulled to it by a fan. The fan blows the air over the evaporator and discharges the cooled air back into the cabin. (***Figure 6-10***)

This discharge can be direct when the evaporator is in a cabin wall. A remotely located evaporator may require ducting from the cabin to the evaporator and from the

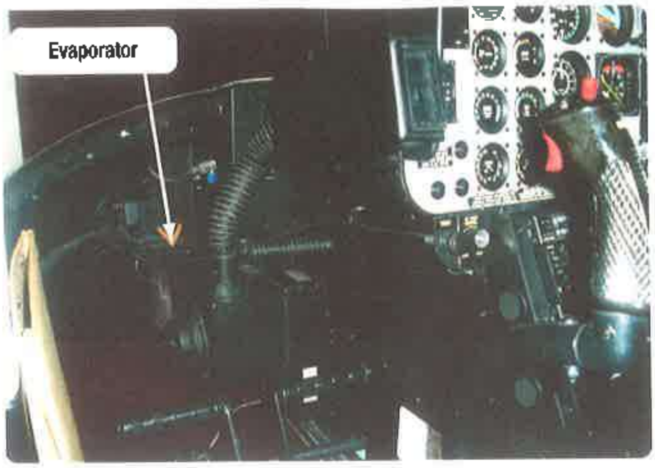


Figure 6-10. Evaporator.

evaporator back into the cabin. Sometimes the cool air produced may be inserted into an air distribution system where it can blow directly on the occupants through individual delivery vents. In this manner, the entire vapor cycle air conditioning system may be located fore or aft of the cabin. A multi-position fan switch controlled by the pilot is generally available.

On ***Figure 6-11*** diagrams, the vapor cycle air conditioning system example has two evaporators that share in the cooling with outlets integrated into a distribution system and cockpit mounted switches for the fans, as well as engaging and disengaging the system. When cabin air is cooled by flowing over the evaporator, it can no longer retain the water that it could at higher temperature. As a result it condenses on the outside of the evaporator and needs to be collected and drained overboard. Fins on the evaporator must be kept from being damaged which could inhibit airflow. The continuous movement of warm cabin air around the

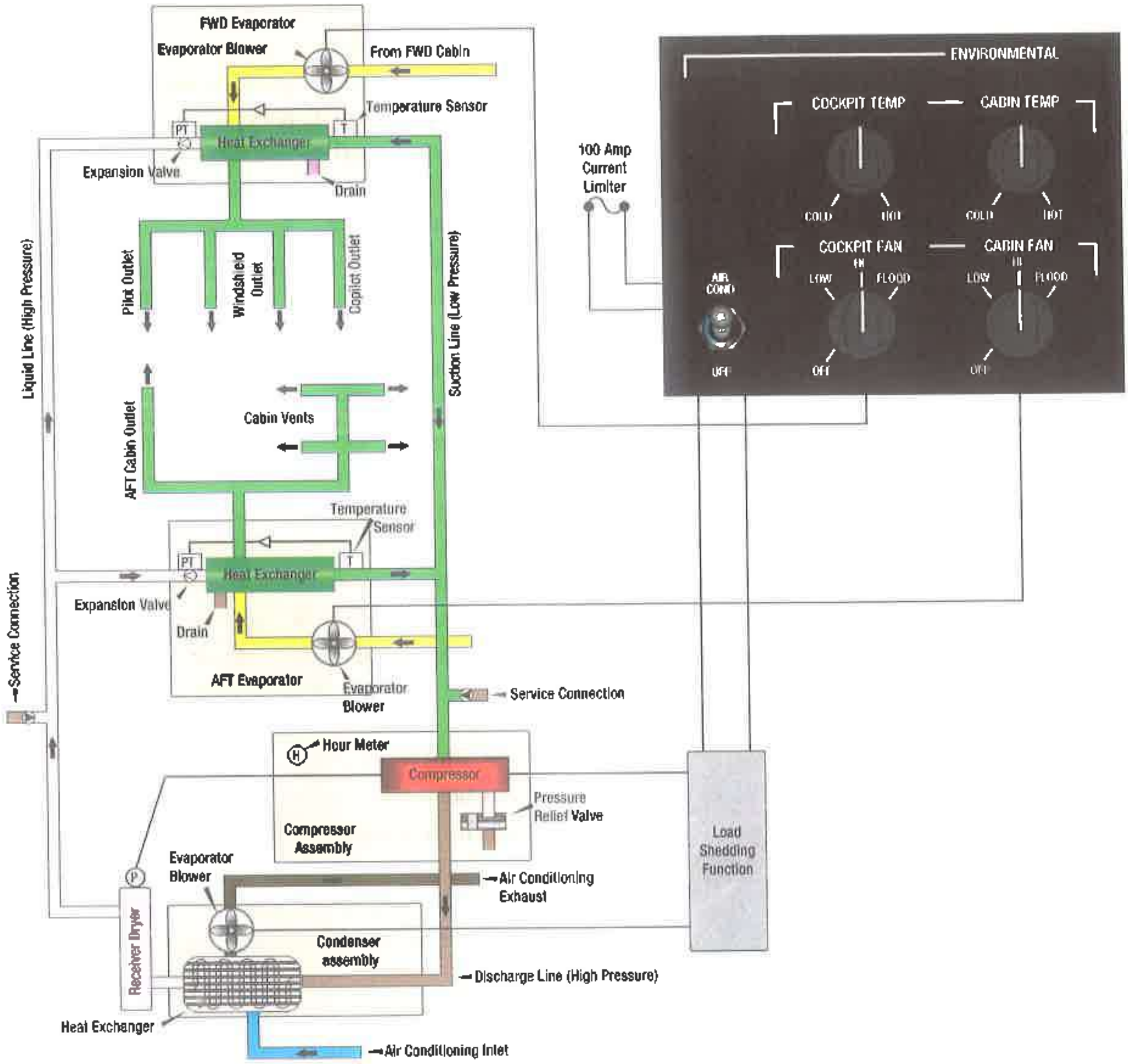


Figure 6-11. Vapor cycle air conditioning example.

fins keeps condensed water from freezing. Ice on the evaporator reduces the efficiency of the heat exchange to the refrigerant.

Compressor

The compressor is the heart of the vapor cycle air conditioning system. It circulates the refrigerant around the vapor cycle system. It receives low pressure low temperature refrigerant vapor from the outlet of the evaporator and compresses it. As the pressure is increased, the temperature also increases. The refrigerant temperature is raised above that of the outside air temperature. The refrigerant then flows out of the compressor to the condenser where it gives off the heat to the outside air. The compressor is the dividing point between the low side and the high side of the vapor cycle system. Often it is incorporated with fittings or has fittings in the connecting lines to it that are designed to service the system with refrigerant. Access to the low and high sides of the system are required for servicing, which can be accomplished with fittings upstream and downstream of the compressor. Modern compressors are either engine driven or driven by an electric motor. Occasionally, a hydraulically driven compressor is used. A typical engine driven compressor, like one found in an automobile, is in the engine compartment and operated by a drive belt. An electromagnetic clutch engages when cooling is required, which causes the compressor to operate. When cooling is sufficient, power to the clutch is cut, and the drive pulley rotates but the compressor does not. (***Figure 6-12***)

Dedicated electric motor driven compressors are also used on aircraft. Use of an electric motor allows the compressor to be located nearly anywhere on the aircraft, since wires can be run from the appropriate bus to the control panel and to the compressor. (***Figure 6-13***)

Figure 6-12. Engine driven compressor.

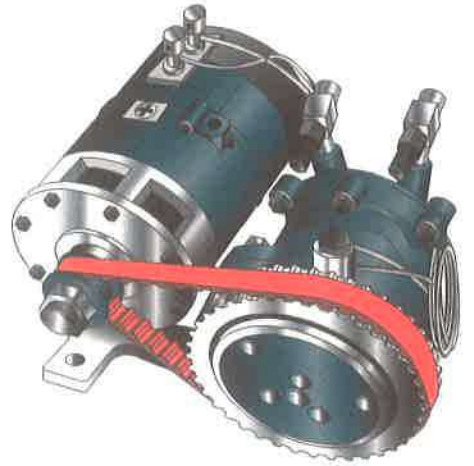


Figure 6-13. Electric motor driven compressor.

Hydraulically driven compressors can also be remotely located. Hydraulic lines from the hydraulic manifold are run through a switch activated solenoid to the compressor. The solenoid allows fluid to the compressor or bypasses it. This controls the operation of the hydraulically driven compressor. Regardless of how the vapor cycle air conditioning compressor is driven, it is usually a piston type pump. It requires use of a lightweight oil to lubricate and seal the unit. The refrigerant entrains the oil which circulates with it around the system. The crankcase of the compressor retains a supply of the oil, the level of which can be checked and adjusted by the technician. Valves exist on some compressor installations, that can be closed to isolate the compressor from the remainder of the vapor cycle system while oil servicing takes place.

Condenser

The condenser is the final component in the vapor cycle. It condenses the hot refrigerant gas coming from the compressor. It is a radiator like heat exchanger situated so that outside air flows over it and absorbs heat from the high pressure high temperature refrigerant received from the compressor. A fan is usually included to draw the air through the compressor during ground operation. On some aircraft, outside air is ducted to the compressor. On others, the condenser is lowered into the airstream from the fuselage via a hinged panel. Often, the panel is controlled by a switch on the throttle levers. It is set to retract the compressor and streamline the fuselage when full power is required. (***Figure 6-14***)

The outside air absorbs heat from the refrigerant flowing through the condenser. The heat loss causes the refrigerant to change state back into a liquid. The high

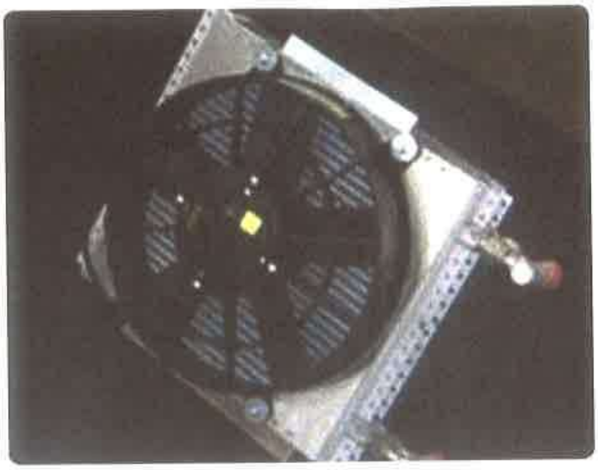


Figure 6-14. Vapor cycle air conditioning condenser.

pressure liquid refrigerant then leaves the condenser and flows to the receiver dryer. A properly engineered system that is functioning normally, fully condenses all the refrigerant flowing through the condenser.

Service Valves

All vapor cycle air conditioning systems are closed systems, however access is required for servicing. This is accomplished by means of two service valves. One valve is in the high side of the system and the other in the low side. A common type of valve used on vapor cycle systems that operate with R12 refrigerant is the Schrader valve. It is like the valve used to inflate tires. (***Figure 6-15***)

A central valve core seats and unseats by depressing a stem attached to it. A pin in the servicing hose fitting is designed to do this when screwed onto the valve's exterior threads. All aircraft service valves should be capped when not in use. R134a systems use valves that are similar to the Schrader valve in function, operation, and location. As a safety device to prevent inadvertent mixing of refrigerants, R134a valve fittings are different from Schrader valve fittings and do not attach to Schrader valve threads. The R134a valve fittings are a quick disconnect type. Another type of valve called a compressor isolation valve is used on some aircraft. It serves two purposes. Like the Schrader valve, it permits servicing the system with refrigerant. It can also isolate the compressor so the oil level can be checked and replenished without opening the entire system and losing the refrigerant charge. These valves are usually hard mounted to the inlet and outlet of the compressor.

Figure 6-15. R12 refrigerant valve.

A compressor isolation valve has three positions. When fully open, it back seats and allows the normal flow of refrigerant in the vapor cycle. When fully closed or front seated, the valve isolates the compressor from the rest of the system and servicing with oil, or even replacement of the compressor, is possible without losing the refrigerant charge. When in an intermediate position, the valve allows access to the system for servicing. The system can be operated with the valve in this position but should be back seated for normal operation. The valve handle and service port should be capped when servicing is complete. (***Figure 6-16***)

VAPOR CYCLE AIR CONDITIONING SERVICING EQUIPMENT

Special servicing equipment is used to service vapor cycle air conditioning systems. The U.S. Environmental Protection Agency has declared it illegal to release R12 refrigerant into the atmosphere. In Europe, this is controlled by regulation #1005/2009 on substances that deplete the ozone layer. Equipment has been designed to capture the refrigerant during the servicing process. Although R134a does not have this restriction, it is illegal in some locations to release it to the atmosphere. It is always good practice to capture all refrigerants for future use, rather than to waste them or by releasing them into the atmosphere. Capturing the refrigerant is a simple process with the proper servicing equipment. The technician should always be vigilant to use the approved refrigerant for the system being serviced and should follow all manufacturer's instructions.

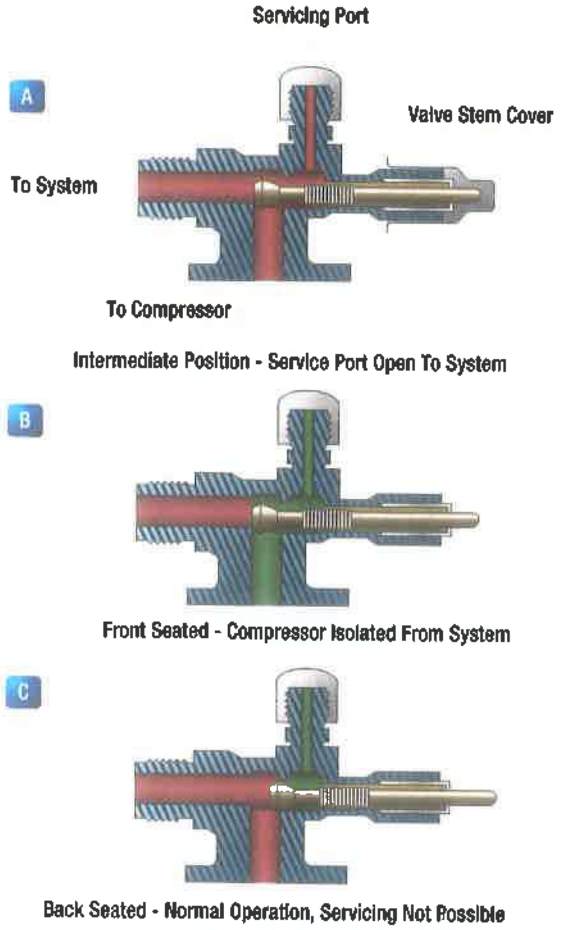


Figure 6-16. Compressor isolation valve.

Manifold Set, Gauges, Hoses, And Fittings

In the past, the main servicing device for vapor cycle air conditioning systems was the manifold set. It contains three hose fittings, two O-ring sealed valves, and two gauges. It is essentially a manifold into which the gauges, fittings, and valves are attached. The valves are positioned to connect or isolate the center hose with either fitting. Hoses attach to the right and left manifold set fittings and the other ends of those hoses attach to the service valves in the vapor cycle system. The center fitting also has a hose attached to it. The other end of this hose connects to either a refrigerant supply or a vacuum pump, depending on the servicing function to be performed. All servicing operations are performed by manipulating the valves. (***Figure 6-17***)

The gauges on the manifold set are dedicated; one for the low side of the system and the other for the high side. The low pressure gauge is a compound gauge that indicates pressures above or below atmospheric pressure (0 gauge pressure). Below atmospheric pressure, the gauge is scaled in inches of mercury down to 30 inches. This is to indicate vacuum. 29.92 inches equals an absolute vacuum (absolute zero air pressure). Above atmospheric pressure, gauge pressure is read in psi. The scale typically ranges from 0 to 60 psi, although some gauges extend up to 150 psi. The high pressure gauge usually has a range from zero up to about 500 psi gauge pressure. It does not indicate vacuum (pressure lower than atmospheric). These gauges and their scales can be seen in ***Figure 6-18***.

Special hoses are attached to the fittings of the manifold valve for servicing the system. The high pressure charging hose is usually red and attaches to the service valve located in the high side of the system. The low pressure hose, usually blue, attaches to the service valve that is located in the low side of the system. The center hose attaches to the vacuum pump for evacuating the system, or to the refrigerant supply for charging the system. Proper charging hoses for the refrigerant specific service valves must be used. When not using the manifold set, be sure the hoses are capped to prevent moisture from contaminating the valves.

Full Service Refrigerant Recovery, Recycling, Evacuation, And Recharging Units

Regulations that require capture of all vapor cycle refrigerant have limited the use of the manifold set. It can still be used to charge a system. The refrigerant container is attached to the center hose and the manifold set valves are manipulated to allow flow into the low or high side of the system as required. In any case, emptying a system of refrigerant requires a service unit designed to collect it. Allowing the refrigerant to

Figure 6-17. Manifold set for servicing a vapor cycle air conditioning system.

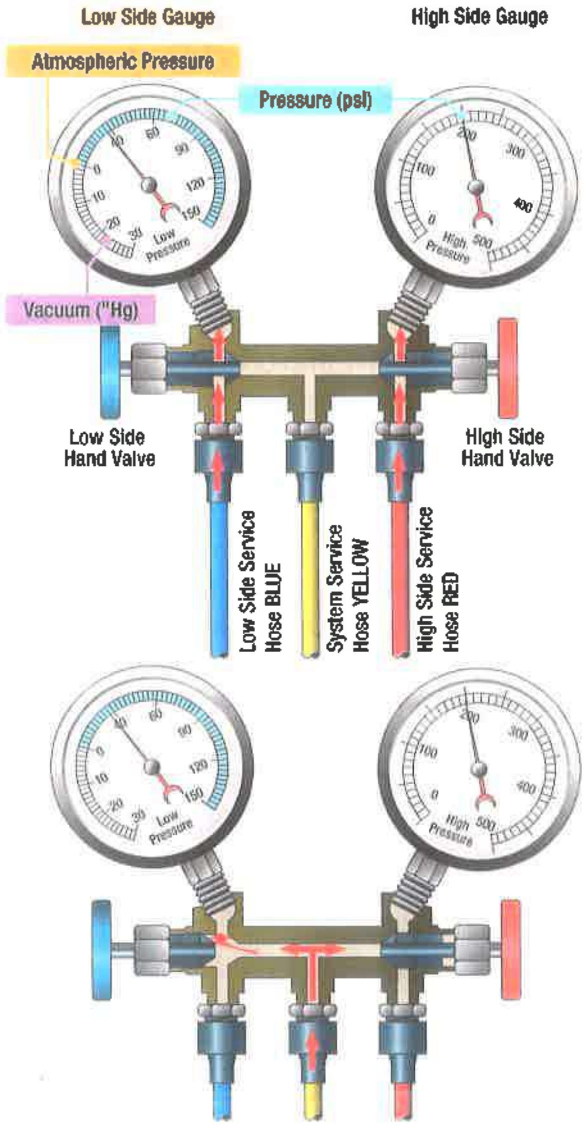


Figure 6-18. Internal working of a manifold set.

flow into a collection container attached to the center hose will not capture the entire refrigerant charge as the system and container pressures equalize above atmospheric pressure. An independent compressor and collection system is required. Modern refrigeration recharging and recovery units are available to perform all the servicing functions required for vapor cycle air conditioning systems. These all-in-one service carts have the manifold set built into the unit. As such, the logic for using a manifold set still applies. Integral solenoid valves, reservoirs, filters, and smart controls allow the entire servicing procedure to be controlled from the unit panel once the high side and low side services hoses are connected. A built-in compressor enables complete system refrigerant purging. A built-in vacuum pump performs system evacuation. A container and recycling filters for the refrigerant and the lubricating oil allow total recovery and recycling of these fluids. The pressure gauges used on the service unit panel are the same as those on a manifold set. Better quality units have an automatic function that performs all the servicing functions sequentially and automatically once the hoses are hooked up to the vapor cycle air conditioning system and the system quantity of refrigerant has been entered. (*Figure 6-19*)

REFRIGERANT SOURCE

R134a comes in containers measured by the weight of the refrigerant they hold. Small 12 ounce to 2.5 pound cans are common for adding refrigerant. Larger 30 and 50 pound cylinders equipped with shut-off valves are often used to charge an evacuated system, and they are used in shops that frequently service vapor cycle systems. (*Figure 6-20*)

These larger cylinders are also used in the full servicing carts described above. The amount of refrigerant required for any system is measured in pounds. Check the manufacturer's service data and charge the system to the level specified using only the approved refrigerant from a known source.

Figure 6-19. Refrigerant recovery-recycle-charging service unit.

Figure 6-20. R134a refrigerant container.

VACUUM PUMPS

Vacuum pumps used with a manifold set, or as part of a service cart, are connected to the vapor cycle system so that the system pressure can be reduced to a near total vacuum. The reason for doing this is to remove all the water from the system. As mentioned, water can freeze, causing system malfunction and can also combine with the refrigerant to create corrosive compounds. Once the system has been purged of its refrigerant and it is at atmospheric pressure, the vacuum pump is operated. It gradually reduces the pressure in the system. As it does, the boiling point of any water in the system is also reduced. Water boils off, or is vaporized under the reduced pressure, and is pulled from the system by the pump, leaving the system moisture free to be recharged with refrigerant. (*Figure 6-21*)

The strength and efficiency of vacuum pumps varies as does the amount of time to hold the system at reduced pressure as specified by manufacturers. Generally, the best established vacuum is held for 15-30 minutes to ensure all water is removed from the system. Follow the manufacturer's instructions when evacuating a vapor cycle air conditioning system. (*Figure 6-22*)

LEAK DETECTORS

Even the smallest leak in a vapor cycle system can cause a loss of refrigerant. When operating normally, little or no refrigerant escapes. A system that requires the addition of refrigerant should be suspected of having a leak. Electronic leak detectors are safe and effective devices. There are many types available that can detect extremely small amounts of escaped refrigerant. The detector is held close to the component and its hose connections where most leaks occur. Audible and visual alarms signal the presence of refrigerant. A detector specified for the type of refrigerant in the system should be chosen. A good leak detector is sensitive enough to detect leaks that would result in less than 1/2 ounce of refrigerant to be lost per year. (*Figure 6-23*)

Other leak detection methods exist. A soapy solution can also be applied to fittings and inspected for the formation of bubbles indicating a leak. Special leak detection dyes compatible for use with refrigerants can be injected into the vapor cycle system and be seen when they are forced out at a leak. Many of these are made visible under UV light. Occasionally, a leak can be detected with close visual inspection. Oil in the system

Inches of Vacuum on Low Side Gauge (inches Hg)	Temperature at Which Water Boils (°C)	Absolute Pressure (psi)
0	100	14.696
4.92	96.1	12.279
9.23	90	10.152
15.94	80	6.866
20.72	70	4.519
24.04	60	2.888
26.28	50	1.788
27.75	40	1.066
28.67	30	0.614
28.92	26.7	0.491
29.02	24.4	0.442
29.12	22.2	0.393
29.22	20.6	0.344
29.32	17.8	0.295
29.42	15	0.246
29.52	11.7	0.196
29.62	7.2	0.147
29.74	0	0.088
29.82	-6.1	0.0049
29.87	-14.4	0.00245
29.91	-31.1	0.00049

Figure 6-21. Boiling temperature depending on pressure.

Figure 6-22. Vacuum pump.

Figure 6-23. Infrared leak detector.

can be forced out with a leak, leaving a visible residue on the bottom side of the leaky fitting. Old hoses may become slightly porous and leak a significant amount of refrigerant over time. Because of the length and area through which the refrigerant is lost, this type of leak may be difficult to detect, even with leak detecting methods. Visibly deteriorated hoses should be replaced.

SYSTEM SERVICING

Vapor cycle air conditioning systems can give many hours of reliable, maintenance free service. Periodic visual inspections, tests, refrigerant level and oil level checks may be all that is required for some time.

Visual Inspection

All components of vapor cycle systems should be checked to ensure they are secure. Be vigilant for any damage, misalignment, or signs of leakage. The evaporator and condenser fins should be checked to ensure they are clean, unobstructed, and not folded over from an impact. Dirt and inhibited airflow through the fins can prevent effective heat exchange to and from the refrigerant. Since the condenser often has ram air ducted to it, or extends into the airstream, check for debris that may restrict airflow. Hinged units should be checked for security and wear. The mechanism to extend and retract the unit should function as specified, including the throttle position switch which is present on many systems. This switch is designed to cut power to the compressor clutch and retract the condenser at full power settings. Condensers may also have a fan to pull air over them during ground operation. This should be checked to ensure its correct functioning. (***Figure 6-24***)

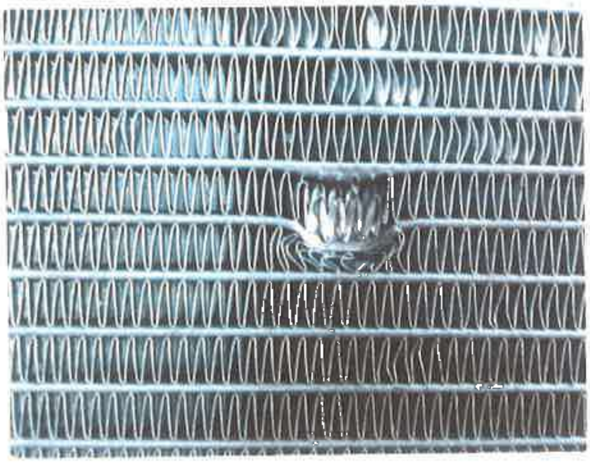


Figure 6-24. Damaged fins on a condenser.

Be sure the capillary temperature feedback sensor to the expansion valve is securely attached to the evaporator outlet. Also, if present in the system, check the security of the pressure sensors and thermostat sensors.

The fan blower should be checked to ensure it rotates freely. Depending on the system, it should run whenever the cooling switch is selected and should change speeds as the selector is rotated for more or less cooling condition. Sometimes systems low on refrigerant can cause ice on the evaporator, as can a faulty expansion valve or feedback control line. Ice formation anywhere on the outside of a vapor cycle air conditioning system should be investigated for cause and corrected. (***Figure 6-25***)

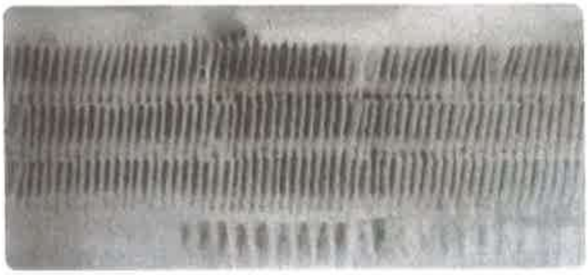


Figure 6-25. Ice on the evaporator.

Security and alignment of the compressor is critical and should be checked during inspection. Belt driven compressors need to have proper belt tension. Check the manufacturer's data for information on how to determine the condition and tension of the belt, as well as how to adjust. Oil level should be sufficient. Typically, 1/4 ounce of oil is added for each pound of refrigerant added to the system. When changing a component, additional oil may need to be added to replace that which is trapped in the replaced unit. Always use the oil specified in the manufacturer's maintenance manual.

Leak Test

As mentioned in the leak detector section above, leaks in a vapor cycle air conditioning system must be discovered and repaired. The most obvious sign of a possible leak is a low refrigerant level. While the system is operating, bubbles present in the sight glass of the receiver dryer indicate a need of more refrigerant. A system check for a leak may be in order. Note that vapor cycle systems normally lose a small amount of refrigerant each year. No action is needed if this amount is within limits. Occasionally, all the refrigerant escapes from the system. No bubbles are visible in the sight glass, but the complete lack of cooling indicates the refrigerant has leaked out.

To locate the leak point, the system needs to be partially charged with refrigerant, so that leak detection methods can be employed. About 50 psi of refrigerant in the high and low sides should be sufficient for a leak check. By inserting the refrigerant into the high side, pressure indicated on the low side gauge confirms that the orifice in the expansion valve is not clogged.

When all refrigerant is lost due to a leak, the entire system should be checked. Each fitting and connection should be inspected visually and with a leak detector. When an air conditioning system loses all its refrigerant charge, air may enter the system. Water may also enter since it is in the air. This means that a full system evacuation must be performed after the leak is found and repaired.

Performance Test

Verification of proper operation of a vapor cycle air conditioning system is often part of a performance test. This involves operating the system and checking its parameters to ensure they are in the normal range. A key indication of performance is the air temperature cooled by the evaporator. This can be measured at the air outflow from the evaporator or at a nearby delivery duct outlet. An ordinary thermometer should read 4-10°C, with the controls set to full cold after the system has been allowed to operate for a few minutes.

Manufacturer's instructions include information on where to place the thermometer, and the temperature range indicating the acceptable performance. Pressures can also be observed to indicate system performance. Depending on ambient temperature, low side pressure in a vapor cycle system operating is normally 10-50 psi. High side pressure is between 125 and 250 psi, again, depending on ambient temperature and the system design. All system performance tests are performed at a specified engine RPM (stable compressor speed) and involve a period to stabilize the operation of the vapor cycle.

Feel Test

A quick feel test can be performed on a vapor cycle air conditioning system to gauge its health. Particularly, components and lines in the high side (from the compressor to the expansion valve) should be warm to the touch. The lines on both sides of the receiver dryer should be at the same temperature. Low side lines and the evaporator should be cool. Ice should not be visible on the outside of the system. If any discrepancies exist, further investigation is needed. On hot, humid days, the cooling output of the vapor cycle system may be slightly compromised due to the volume of water condensing on the evaporator.

PURGING THE SYSTEM

Purging the system means emptying it of its refrigerant charge. Since the refrigerant must be captured, a service cart with this capability should be used. By connecting the hoses to the high and low side service valves and selecting recover mode, the cart solenoid valves position so that a purging compressor pumps the refrigerant out of the vapor cycle system and into a recovery tank.

Vapor cycle systems must be properly purged before opening for maintenance or component replacement. Once opened, precautions should be taken to prevent contaminants from entering the system. When suspicion exists that the system has been contaminated, such as when a component has catastrophically failed, it can be flushed clean. A special fluid flush formulated for vapor cycle air conditioning systems should be used. The receiver dryer is removed from the system for flushing and a new unit is installed, as it contains fresh filters. Follow the aircraft manufacturer's instructions.

CHECKING COMPRESSOR OIL

The compressor is a sealed unit that is lubricated with oil. Any time the system is purged, it is also an opportunity to check the oil quantity in the compressor crankcase. This is often done by removing a filler plug and using a dipstick. Oil quantity should be maintained within the proper range using oil recommended by the manufacturer. Be certain to replace the filler plug after checking or adding oil. (***Figure 6-26***)

EVACUATING THE SYSTEM

Only a few drops of moisture can contaminate a vapor cycle system. If this moisture freezes in the expansion valve, it could completely block the refrigerant flow. Water is removed from the system by evacuation. Anytime the system refrigerant charge falls below atmospheric pressure, the refrigerant is lost, or the system is opened, it must be evacuated before recharging.

Evacuating a vapor cycle system is also known as pumping down the system. A vacuum pump is connected and pressure inside the system is reduced to vaporize

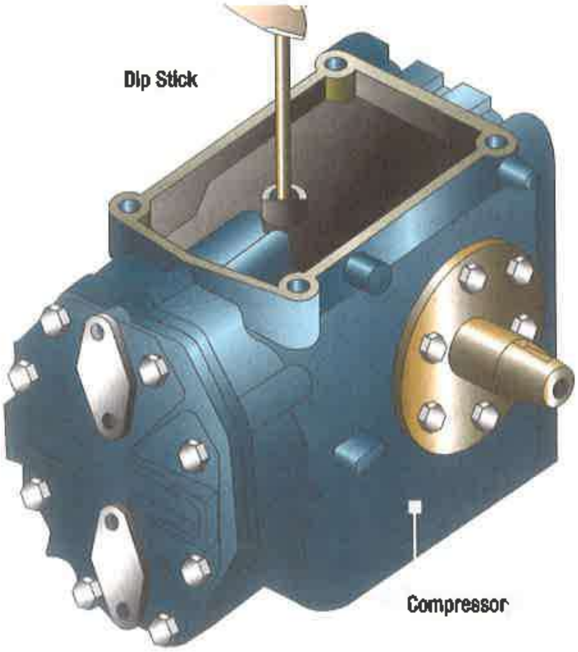


Figure 6-26. Oil checking on compressor.

any water that may exist. Continued operation of the vacuum pump draws the water vapor from the system. A typical pump used for evacuating an air conditioning system can reduce system pressure to about 29.62 "Hg (gauge pressure). At this pressure, water boils at 7.2°C. Operate the vacuum pump to achieve the recommended gauge pressure. Hold this vacuum for as long as the manufacturer specifies.

So long as a vapor cycle system retains a charge higher than atmospheric pressure, any leak forces the refrigerant out of the system. The system pressure prevents air (and water vapor) from entering. Therefore, it is permissible to recharge or add refrigerant to a system that has not dropped below atmospheric pressure without evacuating the system.

CHARGING THE SYSTEM

Charging capacity of a vapor cycle air conditioning system is measured by weight. The manufacturer's maintenance manual specifies this amount and the type of oil to be put into the system when filling. Pre-weighing the refrigerant or setting the refrigerant weight into the servicing cart ensures the system is filled to capacity. Charging a vapor cycle system should be undertaken immediately after evacuation of the system is completed. With the hoses still connected to the high and low side service valves, select the charge mode on the service cart panel so that the refrigerant supply is available. First, refrigerant is released into the high side of the system. Observe the low side gauge. When the low side gauge begins to indicate pressure, it is known that refrigerant is passing through the tiny orifice in the expansion valve. As pressure builds in the high side, the flow of refrigerant stops.

To complete the charge of the system, refrigerant needs to be drawn in by the compressor. A major concern is to avoid damaging the compressor if liquid refrigerant enters the compressor inlet. After the initial release of refrigerant into the high side, the high side service valve is closed and the remaining charge is made through the low side service valve. The engine is started and run at a specified RPM, usually a high idle speed. Full cool is selected on the air conditioning control panel in the cockpit. As the compressor operates, it draws vapor into the low side, until the correct weighed amount of refrigerant is in the system.

Charging with a manifold set is accomplished in the same way. The manifold center hose is connected to the refrigerant source that charges the system. After opening the valve on the container (or puncturing the seal on a small can), the center hose connection on the manifold set should be loosened to allow air in the hose to escape. Once the air is bled out of the hose, the refrigerant can enter the system through whichever service valve is opened.

Oil quantity added to the system is specified by the manufacturer. Refrigerant premixed with oil is available and may be permissible for use. This eliminates the need to add oil separately. Alternately, the amount of oil to be put into the system can be selected on the servicing cart. Approximately ¼ ounce of oil for each pound of refrigerant is standard, however, follow the manufacturer's specifications.

AIR-CYCLE AIR CONDITIONING

Air cycle air conditioning prepares engine bleed air to pressurize the aircraft cabin. However, its temperature and quantity must be controlled to maintain a comfortable cabin environment in all weather conditions. The air cycle system is often called the air conditioning package or pack.

SYSTEM OPERATION

Bleed air is too hot to be used in the cabin without being cooled. It enters the air cycle system and is routed through a heat exchanger where ram air cools the bleed air. This cooled bleed air is then directed into an air cycle machine. There, it is compressed before flowing through a secondary heat exchanger that cools the air again with ram air. The bleed air then flows back into the air cycle machine where it drives an expansion turbine which cools it even further. Water is then removed and the air is mixed with bypassed bleed air for a final temperature adjustment and finally sent to the cabin through the air distribution system. By examining the operation of each component during an air cycle process, a better understanding can be developed, on how bleed air is conditioned for cabin use.

The mixing of air can be done in a variety of ways. Mixing air valves, flow control valves, shutoff valves, and other various control valves are set by switches in the cockpit. One particular bleed air system in helicopters uses mini-ejectors to combine bleed air with cabin air. Bleed air conditioning systems are simple and function well, as long as the valves, ducting, and controls are in operational condition.

PNEUMATIC SYSTEM SUPPLY

During normal flight, the pneumatic system is supplied by bleed air tap-offs located within a turbine engine's compressor section. A typical pneumatic distribution system connects air supply sources from the engine to user ventilation outlets through a pneumatic manifold system and their appropriate control valves. The hot air passes through non-return valves and pilot operated control valves into jet pumps. The jet pumps draw cold air from outside which dilutes and cools the compressor bleed air. The temperature can be adjusted by the pilot by controlling the valve, and the amount of air delivered can be varied by operated butterfly valves.

COMPONENT OPERATION

Pack Valves

The pack valve is the valve that regulates bleed air from the pneumatic manifold, into the air cycle system. It is controlled with a switch from the air conditioning panel in the cockpit. Many pack valves are electrically controlled and pneumatically operated. Also known as the supply shutoff valve, the pack valve opens, closes, and modulates to allow the air cycle system to be supplied with a designed volume of hot, pressurized air. (***Figure 6-27***)

If an overheat or other abnormal condition requires that the air conditioning package is shut down, a signal is sent to the pack valve to close.

Bleed Air Bypass

All turbine aircraft are designed with a means for bypassing some of the pneumatic air supplied to the air cycle system around that system. This warm bypassed air must be mixed with the cold air produced by the air cycle system so the air delivered to the cabin is at

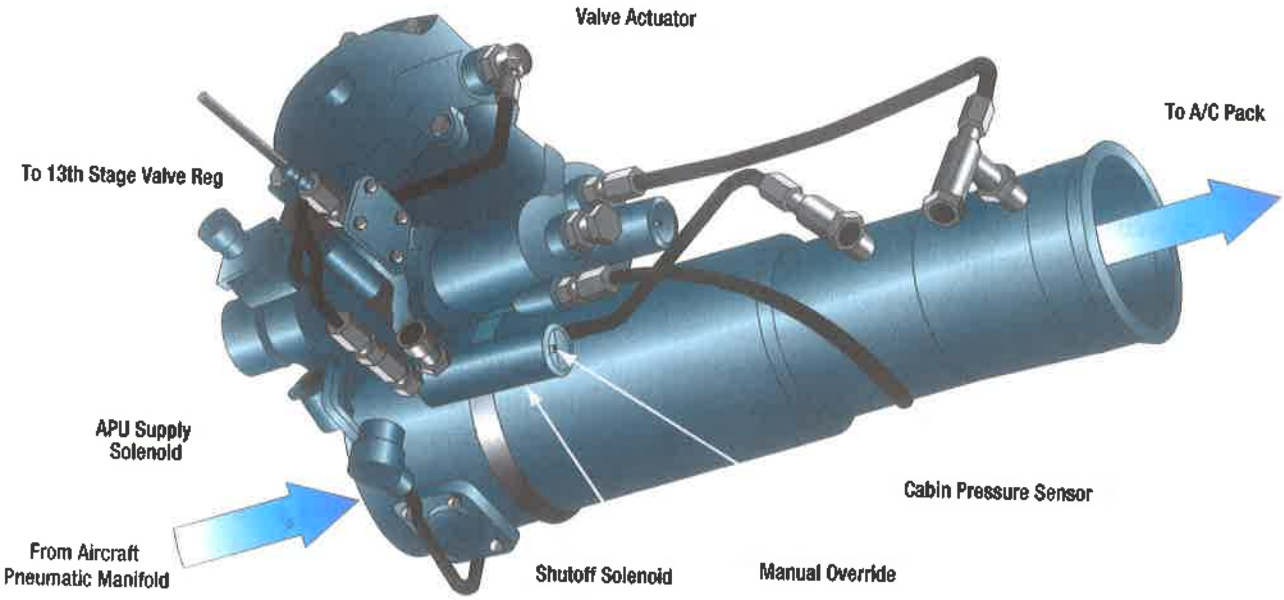


Figure 6-27. supply shutoff valve.

a comfortable temperature. This is accomplished by a mixing valve which controls the flow of bypassed air and the air to be cooled in order to meet the requirements of the auto temperature controller. It can also be controlled manually with the cabin temperature selector in manual mode. Other air cycle systems may refer to the valve that controls the air bypassed around the cooling system, such as a temperature control valve, a trim air pressure regulating valve, or a similar element.

Primary Heat Exchanger

Generally, the warm air passing through the air cycle system first passes through a primary heat exchanger. It acts similarly as a radiator in an automobile. A controlled flow of ram air is ducted over and through the exchanger, which reduces the temperature of the air inside the system. (***Figure 6-28***)

When on the ground, a fan draws air through the ram air duct so that the heat exchange is possible when the aircraft is stationary. In flight, ram air doors are modulated to increase or decrease ram air flow to the exchanger. During slow flight, the doors are open. At higher speed, the doors move toward the closed position reducing the amount of ram air going into the exchanger. Similar operation is accomplished on smaller aircraft by means of a valve.

Refrigeration Turbine Unit and Secondary Heat Exchanger

The heart of the air cycle air conditioning system is the refrigeration turbine unit, also known as the Air Cycle Machine (ACM). It comprises a compressor that is driven by a turbine on a common shaft. Air within the system flows from the primary heat exchanger into the compressor side of the ACM. As the air is compressed, its temperature rises. It is then sent to a secondary heat exchanger, similar to the primary heat exchanger located in the ram air duct. The elevated temperature of the ACM compressed air facilitates an easy exchange of heat energy to the ram air. The cooled system air, still under pressure from the continuous system air flow and the ACM compressor exits the secondary heat exchanger and is directed to the turbine side of the ACM.

The steep blade pitch angle of the ACM turbine extracts more energy from the air as it passes through and drives the turbine. Once through, the air can expand at the ACM outlet, cooling even further. The combined energy loss, first from the air driving the turbine and then expanding at the turbine outlet, lowers the system air temperature to near freezing. (***Figure 6-29***)

Figure 6-28. Heat exchanger.

The cool air from the air cycle machine can no longer hold the same water quantity as when it was warm. A water separator is used to remove the water from the saturated air before it is sent to the aircraft cabin. The separator operates with no moving parts. Foggy air from the ACM enters and is forced through a fiberglass sock that condenses and coalesces the mist into larger water drops. The convoluted interior structure of the separator swirls the air and water. The water is collected on the sides of the separator and drains down and out of the unit while the dry air passes through. A bypass valve is incorporated in case of a blockage. (***Figure 6-30***)

Refrigeration Bypass Valve

As mentioned, air exiting the ACM turbine expands and cools. It becomes so cold that it could freeze the water in the water separator, thus inhibiting or blocking airflow. A temperature sensor in the separator controls a refrigeration bypass valve designed to keep the air flowing through the water separator above freezing temperatures. The valve is also identified by other names such as a temperature control valve or an anti-ice valve. It bypasses warm air around the ACM when opened. The air enters the expansion ducting just upstream of the water separator where it heats the air just enough to keep it from freezing. Thus, the refrigeration bypass valve regulates the temperature of the ACM discharge air so it does not freeze when passing through the water separator. All air cycle air conditioning systems use at least one ram air heat exchanger and an ACM with an expansion turbine to remove heat energy from the bleed air, but variations exist.

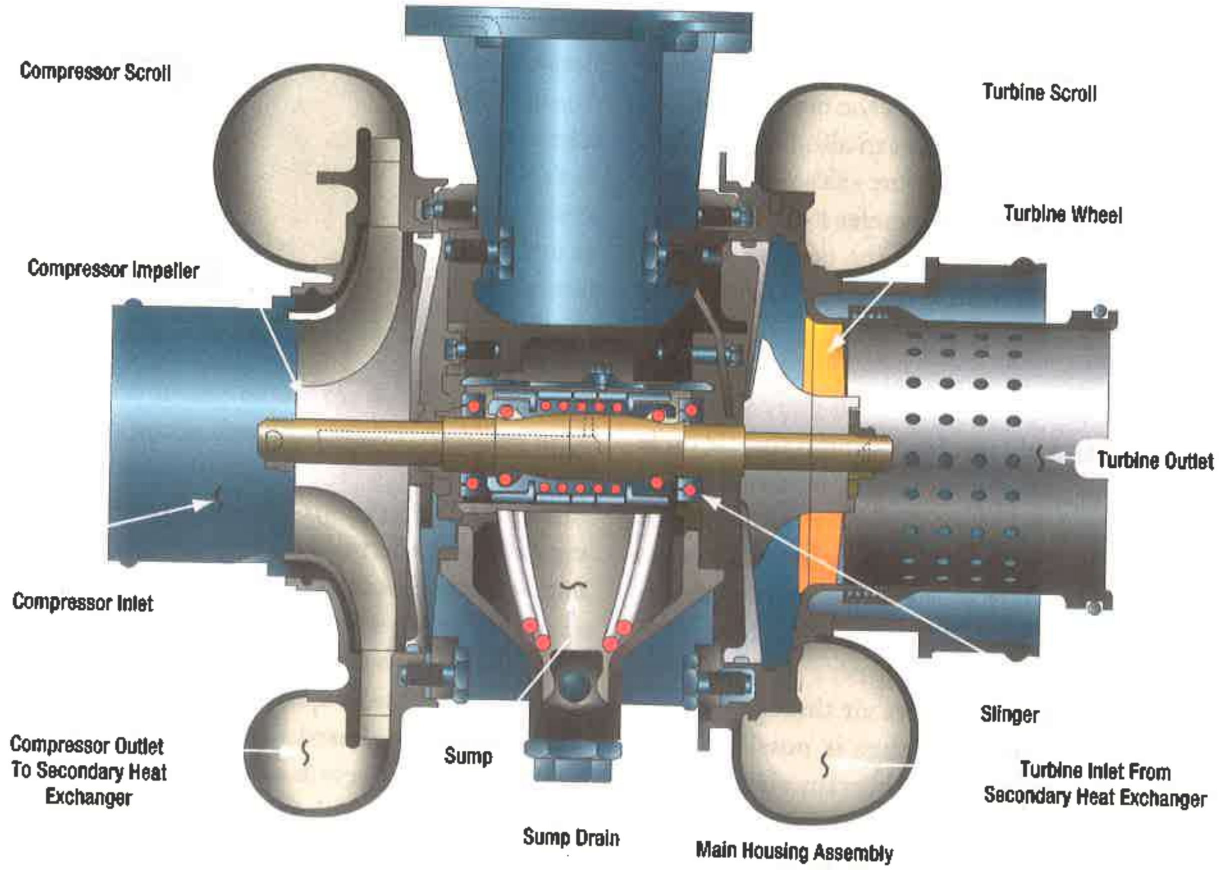


Figure 6-29. Cutaway diagram of an air cycle machine.

DISTRIBUTION SYSTEMS

The air distribution system supplies the conditioned air to the cockpit and cabin zones. Valves used for selecting ventilating air, temperature trim air, as well as in line fans and jet pumps to increase flow in certain areas of the cabin are all components of the air distribution system.

Temperature sensors, overheat switches, and check valves are also common. Mixing air conditioned air with bleed air in a duct or mixing chamber allows the crew to select the exact temperature desired for the cabin. The mixing valve is controlled in the cockpit or cabin by a temperature selector. Centralized manifolds from which air can be distributed are common.

On most aircraft the air distribution system makes provisions for ducting and circulating cooling air towards electronics equipment bays. It also contains a gasper air system. This is air ducted from the cold air manifold or from a duct to an overhead adjustable delivery nozzle at the crew position. An inline fan controlled from the cockpit supplies a steady stream of gasper air that can be regulated or shut off with the delivery nozzle(s).

Air Ducts

Ducts having circular or rectangular cross sections are most frequently used in air distribution systems. Circular ducts are used wherever possible, while rectangular ducts are generally used where circular ducts cannot because of installation or space limitations. Cabin air supply ducts are usually made from aluminum alloys, stainless steel, or plastic. Main ducts for air temperatures over 200°C are made from stainless steel. Those parts of the ducting where the air temperature does not exceed 100°C are usually constructed from soft aluminum. Plastic ducts, both rigid and flexible, are used as outlet ducts to distribute the conditioned air.

FLOW AND TEMPERATURE CONTROL SYSTEMS

FLOW CONTROL SYSTEM

Control valves in a flow control system regulate the total cabin air inflow. There are different types of flow control valves used to regulate specific volumetric airflows independent of the pneumatic supply pressure and the actual cabin pressure. In addition, all valves have an electrical shutoff function and a mechanical close and

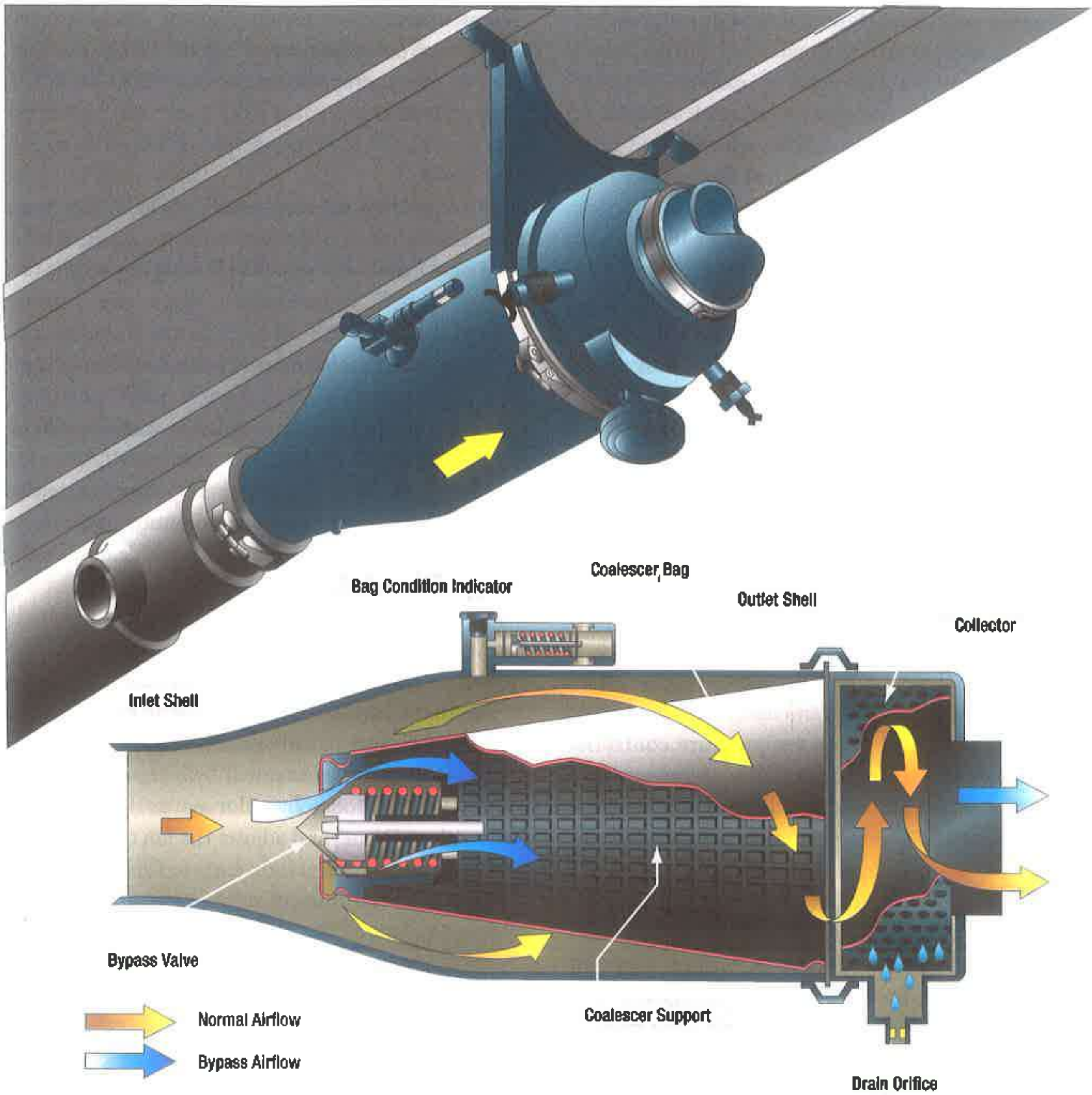


Figure 6-30. Water separator.

locking device with a visual position indicator. They are pneumatically actuated and spring loaded closed without pressure. The open pressure will be discharged to ambient air by thermostats in case of pack overheat in order to close the valve. In case of electrical power loss the flow control valves will open with pneumatic pressure.

Airflow measuring devices (venturi or electronic flow sensor) are mass flow meters. Therefore a reference signal for the air density (cabin pressure) is required to regulate a specific volumetric airflow.

There are two main types of flow control valves: those to regulate a constant airflow, and others to regulate a variable airflow. For passenger comfort and fuel saving, the flow can be regulated by some flow control valves according to a manually selected or computed demand. These valves use a torque or a stepper motor to adjust the open pressure for the valve. The regulated airflow is inversely proportional to the torque motor current. The computed flow demand may depend on various parameters such as:

- Number of packs in use.
- Number of cabin recirculating fans in use; (if cabin

fans are in use, the pack flow should be reduced, to prevent excessive cabin airflow).

- Selected number of passengers or manual flow selection; (with a higher number of passengers onboard, the pack flow should be increased to provide a minimum of fresh air for passenger comfort).
- Zone temperature cooling demand; (if a high cooling demand exists, the pack flow should be increased to provide a faster cabin cool down).
- Take-off or landing mode; (during take-off and landing, the pack flow will be reduced or shut off to unload the engines).
- Type of pneumatic source for the air conditioning packs; (during APU bleed air supply, the flow control valve is controlled fully open. The pack flow regulation in this case is performed by varying the supply pressure from the APU).

TEMPERATURE CONTROL SYSTEM

Most cabin temperature control systems operate in a similar manner. Temperature is monitored in the cabin, cockpit, conditioned air ducts, and distribution air ducts. These values are input into a temperature controller normally located in the electronics bay. A temperature selector in the cockpit can be adjusted to input the desired temperature.

The temperature controller compares the actual temperature signals received from the various sensors with the desired temperature input. Circuit logic for the selected mode processes these input signals. An output signal is sent to a valve in the air cycle system. This valve has different names depending on the aircraft manufacturer and design of the environmental control systems (i.e., mixing valve, temperature control valve, trim air valve). It mixes warm bleed air that bypassed the air cycle cooling process with the cold air produced by it. By modulating the valve in response to the signal from the temperature controller, air of the selected temperature is sent to the cabin through the air distribution system.

AIR CONDITIONER CONTROL PANEL

To control flow and temperature, different functions can be chosen on the Air Conditioner Control Panel. (*Figure 6-31*)

Temperature

A rotating potentiometer labeled "Temp" permits the pilot to select temperature between 15°C to 35°C (C = Cold / H = Hot).

Fan Speed

A three position selector switch labeled "Vent" permits the pilot to choose the fan rotation speed (Off = no rotation / Low = low speed / Hi = high speed).

Mode Selection

A three position selector switch labeled "Mode" permits to choose between:

- Vent: permits a single ventilation without cooling or heating.
- Con: permits control of the temperature according to the temperature rotating potentiometer.
- Dem: permits demisting the windshield with air at approximately 70°C.

Additional modes include:

- Forced Mode: This switch permits an override of the automatic cut-out in the event of an engine failure. A combination of different modes permits all the possibilities to be obtained.
- Ventilation Mode: The pilot moves the "Mode" switch to "Vent" and adjusts the fan speed with the "Vent" switch (High, Low or Off). With this, the ventilation shut-off valve opens, and the others remain closed. (*Figure 6-32*)

Figure 6-31. Air Conditioning Control Panel.

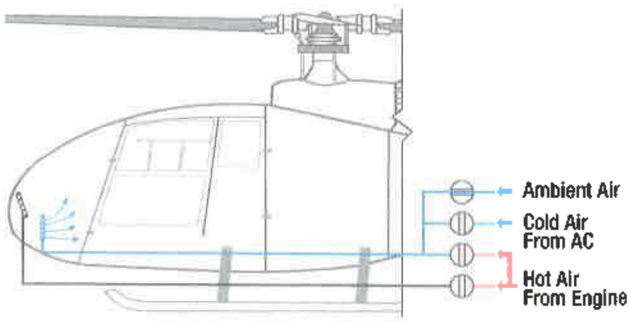


Figure 6-32. Ventilation mode.

- Cooling Mode: The pilot moves the "Mode" switch to "Con", adjusts the fan speed with the "Vent" switch and turns the temperature potentiometer on the cold position C. The consequence is that the cold air shut-off valve opens, and the others remain closed. (***Figure 6-33***)
- Heating Mode: The pilot moves the "Mode" switch to "Con" position, adjusts the fan speed with the "Vent" switch and turns the temperature potentiometer on the hot position H. The consequence is that the hot air shut-off valve opens, and the others remain closed. (***Figure 6-34***)
- Demisting Mode: The pilot moves the "Mode" switch to "Dem" position. The fan speed is automatically set to maximum speed and the temperature potentiometer is bypassed due to an automatic temperature chosen by the Electronic Control Unit. The consequence is that the demisting hot air shut-off valve opens, and the others remain closed. (***Figure 6-35***)

PROTECTION AND WARNING DEVICES

Depending on the type of helicopter, the air conditioning system can be controlled automatically or manually. In both cases, the crew must be informed in the event of a breakdown.

One of the major concerns with air conditioning systems is overheating. A leak anywhere in pneumatic system ducting can pose a fire hazard. Often, a continuous loop fire detector will be run the length of pneumatic ducting or around a bay containing pneumatic lines such as the air conditioning and APU bays. Should an overheat caused by a leak be detected, warnings occur in the cockpit on the central warning panel. Typically, the flow of air in the indicated section of the pneumatic system is shut off. In the case of an internal control unit failure, the system is automatically cut off, the fan is shut down, the control indicator light goes off and the regulation electrical valve closes. Overheat detection sensors may be installed in the nozzle or on the air routing pipes. They operate based on the bi-metal strip principle, and close the electrical circuit as soon as the critical temperature is reached (around 100°C).

In the cockpit, an amber indicator light comes on to warn the pilot that there is a system failure. (***Figure 6-36***) This is done by closing an isolation valve or shutoff valve.

On aircraft designed with such a detection system, if the indicator lights are on, in order to reduce the air conditioning system, the pilot must perform the tasks necessary to shut down the air conditioning system according to the procedures described in the manuals.

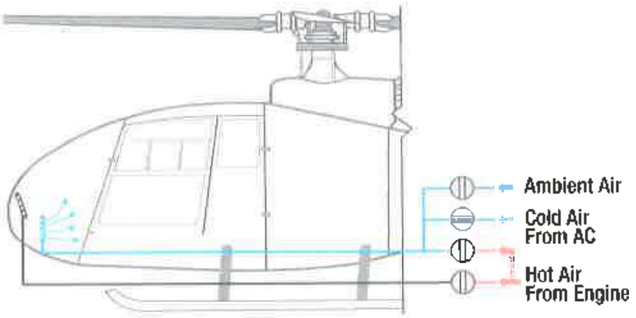


Figure 6-33. Cooling mode.

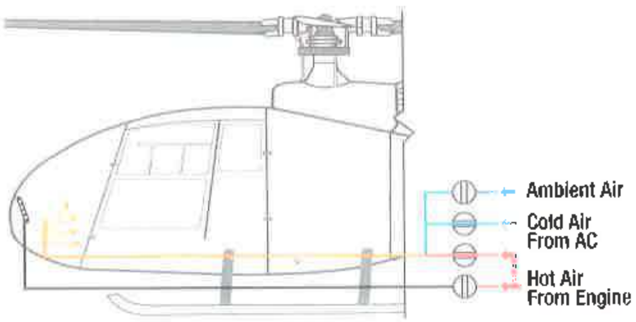


Figure 6-34. Heating mode.

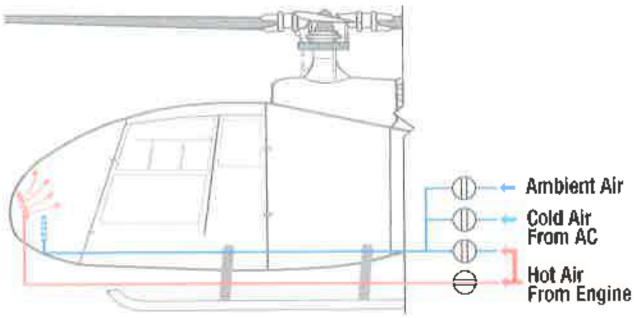


Figure 6-35. Demisting position.

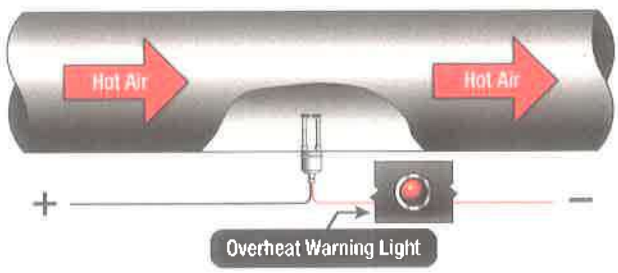
Air Conditioning Routing Pipe


Figure 6-36. Over heat detector.

In case the overheat occurred due to a failure of the automatic temperature control system, the pack can be typically reset and operated again manually.

Smoke detection is sometimes used in the cabin, cargo bays and other areas of the aircraft with controlled operating environments. Warnings are indicated on the central warning system. The warning may be aural or visual. Flight crews may immediately utilize emergency oxygen to maintain control of the aircraft and to avoid hypoxia.

Question: 6-1
Name the two possible sources of air conditioning air when a helicopter is on the ground.

Question: 6-2
What can be done to a liquid to lower its boiling point?

Question: 6-3
What should you do if you are tasked to service a vapor cycle air conditioner which you find to be running on R12 refrigerant?

Question: 6-4
Relating to the high and low side of a vapor cycle system, where is the compressor located?

Question: 6-5
When would the compressor isolation valve in a vapor cycle system be placed in its intermediate position?

Question: 6-6
What is the final cooling mechanism within an air cycle air conditioning system?

Question: 6-7
In what way is moisture removed from the cooled air in an air cycle system?

Question: 6-8
What primarily determines which type of material is used to construct air ducts in air conditioning systems?

ANSWERS

Answer: 6-1
The auxiliary Power Unit (APU) or a ground cart.

Answer: 6-2
Lower the ambient pressure.

Answer: 6-3
Service it with R12 refrigerant.

Answer: 6-4
In between the high and low side.

Answer: 6-5
When servicing the system.

Answer: 6-6
The expansion of air through the output of the expansion turbine.

Answer: 6-7
The air is swirled so that droplets form within a separator and then drain out.

Answer: 6-8
The temperature of the air flowing through the ducts.

HELICOPTER AERODYNAMICS, STRUCTURES AND SYSTEMS

INSTRUMENTS/AVIONIC SYSTEMS

SUB-MODULE 07

PART-66 SYLLABUS **LEVELS**

CERTIFICATION CATEGORY → B1.3 B1.4

Sub-Module 07

INSTRUMENTS/AVIONIC SYSTEMS

Knowledge Requirements

12.7 - Instruments/Avionic Systems

12.7.1 - Instrument Systems (ATA 31)

Pitot static: altimeter, air speed indicator, vertical speed indicator; 2

Gyroscopic: artificial horizon, attitude director, direction indicator, horizontal situation indicator, turn and slip indicator, turn coordinator;

Compasses: direct reading, remote reading;

Vibration indicating systems - HUMS;

Glass cockpit;

Other aircraft system indication.

12.7.2 - Avionic Systems

Fundamentals of system layouts and operation of: 1

Auto Flight (ATA 22);

Communications (ATA 23);

Navigation Systems (ATA 34).

12.7 - INSTRUMENTS/AVIONIC SYSTEMS

12.7.1 - INSTRUMENT SYSTEMS (ATA 31)

INTRODUCTION

Since the beginning of manned flight, it has been recognized that supplying the pilot with information about the aircraft and its operation could be useful and lead to safer flight. The Wright Brothers had very few instruments on their Wright Flyer, but they did have an engine tachometer, an anemometer (wind meter), and a stopwatch. They were obviously concerned about the aircraft engine and the progress of their flight. From that simple beginning, a wide variety of instruments have been developed to inform flight crews of different parameters. Instrument systems now exist to provide information on the condition of the aircraft, engine, components, the aircraft's attitude in the sky, weather, cabin environment, navigation, and communication. ***Figure 7-1*** shows various instrument panels from the Wright Flyer to a modern airliner.

As the size and complexity of modern aircraft has grown, so too has the need to provide information to the flight crew, without sensory overload or over cluttering the cockpit. As a result, the various individual instruments in the cockpit have evolved into a sophisticated computer controlled digital interface with flat panel display screens and prioritized messaging. A visual comparison between a conventional helicopter cockpit and a glass cockpit is shown in ***Figure 7-2***.

There are usually two parts to any instrument or instrument system. One part senses the situation and the other part displays it. In analog instruments, both functions often take place in a single unit or instrument case. These are called direct sensing instruments. Remote sensing requires the information to be sensed, or captured, and then sent to a separate display unit in the cockpit. Both analog and digital instruments make use of these methods. (***Figure 7-3***)

The relaying of bits of information can be done in various ways. Electricity is often used by way of wires that carry sensor information into the cockpit. Sometimes pneumatic lines are used.

In complex aircraft, this can lead to an enormous amount of tubing and wiring terminating behind the instrument display panel. In modern aircraft, more efficient methods of this information transfer can be accomplished via the

Figure 7-1. From top to bottom.

Figure 7-2. A conventional panel of a Bell 222 and modern glass panel of a Leonardo TH-119.

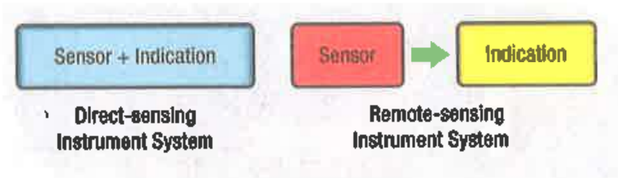


Figure 7-3. There are two parts to any instrument system.

use of digital data buses. Essentially, these are wires that share message carrying for many instruments by digitally encoding the signal for each. This reduces the number of wires and weight required to transfer remotely sensed information for the pilot's use. Flat panel computer display screens that can be controlled to show only the information desired are also lighter in weight than the numerous individual gauges it would take to display the same information simultaneously.

A bonus is the increased reliability in these solid-state systems. It is the job of the aircraft technician to understand and maintain all aircraft, including these various instrument systems. Accordingly, in this chapter, discussions begin with analog instruments and then refer to modern digital instrumentation when appropriate.

CLASSIFICATION BY TYPE

There are three basic kinds of instruments classified by the job they perform: flight instruments, engine instruments, and navigation instruments. Other miscellaneous gauges and indicators provide information that do not fall into these classifications, especially on large complex aircraft. Needs such as flight control position, cabin environmental systems, electrical power, etc. are all monitored and controlled from the cockpit via the use of instrument systems. Many are regarded as position/condition instruments since they usually report the position of a certain movable component on the aircraft, or the condition of various aircraft components or systems not included in the primary three groups.

Flight Instruments

Instruments used in controlling the aircraft's attitude are known as the flight instruments. Basic flight instruments include the altimeter that displays aircraft altitude, the airspeed indicator and a magnetic direction indicator such as a compass. Additionally, an artificial horizon, turn coordinator, and vertical speed indicator are flight instruments present in most aircraft. Much variation exists for these instruments, which is explained throughout this chapter. Over the years, flight instruments have come to be situated similarly on the panels in most aircraft. This basic T arrangement for flight instruments is shown in ***Figure 7-4***.

The top center position directly in front of the pilot and copilot is the basic position for the artificial horizon. This is so even with modern glass cockpits (those with solid state, flat screen indicating systems).

Figure 7-4. The basic T arrangement.

Original analog flight instruments are operated by air pressure and the use of gyroscopes, thus avoiding the use of electricity, which could put the pilot in a dangerous situation if the aircraft lost electrical power.

Development of sensing and display techniques, combined with advanced aircraft electrical systems, have made it possible for reliable primary and secondary instrument systems that are electrically operated. Nonetheless, a pneumatic altimeter, a gyro artificial horizon, and a magnetic direction indicator are usually retained somewhere in the instrument panel for redundancy. (***Figure 7-5***)

Engine Instruments

Engine instruments are those designed to measure operating parameters of the aircraft engine(s). These are usually quantity, pressure, and temperature indications and for measuring engine speed. The most common engine instruments are the fuel and oil quantity and pressure gauges, tachometers, and temperature gauges.

Figure 7-6 contains various engine instruments found on reciprocating and turbine-powered aircraft.

Engine instrumentation is often displayed in the center of the cockpit where it is easily visible to both the pilot and co-pilot. (***Figure 7-7***)

Multiengine aircraft often use a single gauge for a particular engine parameter, but may include multiple pointers on the same dial face.

Navigation Instruments

Navigation instruments are those providing information to guide the aircraft along a definite course. This group includes various types of compasses, some of which incorporate the use of radio signals to define a specific course while flying the aircraft from one airport to another. Other navigational instruments are specifically designed to direct the pilot in the approach to landing.

Figure 7-5. A secondary airspeed and altimeter provides redundancy for the digital displays.

Figure 7-7. Engine and other common instruments placed between pilot and copilot of this Hughes MD-500.

Reciprocating Engines	Turbine Engines
Oil Pressure	Oil Pressure
Oil Temperature	Exhaust Gas Temperature (EGT)
Cylinder Head Temperature (CHT)	Turbine Inlet Temperature (TIT) or Turbine Gas Temperature (TGT)
Manifold Pressure	Engine Pressure Ratio (EPR)
Fuel Quantity	Fuel Quantity
Fuel Pressure	Fuel Pressure
	Fuel Flow
Tachometer	Tachometer (Percent Calibrated)
	N_1 and N_2 Compressor Speeds
Carburetor Temperature	Torquemeter (On Turboprop and Turboshaft Engines)

Figure 7-6. Common engine instruments.

Traditional navigation instruments include a clock and a magnetic compass. Along with the airspeed indicator and wind information, these can be used to calculate navigational progress. In modern aircraft, instruments sending locating information via radio waves have replaced these manual calculations. Global Position Systems (GPS) use satellites to pinpoint the location of the aircraft via geometric triangulation. An example of navigational information displayed on a glass cockpit display is shown in ***Figure 7-8***.

PITOT-STATIC INSTRUMENTS

ALTIMETER, AIR SPEED INDICATOR, VERTICAL SPEED INDICATOR

Some of the most important flight instruments derive their indications by measuring air pressure. Gathering and distributing various air pressures is the function of the pitot-static system.

PITOT TUBES AND STATIC VENTS PRINCIPLE

On simple aircraft, a pitot-static system may consist of a pitot tube with impact and static air pressure ports and leak free tubing connecting these air pressure pickup points to the instruments that require them. The altimeter, airspeed indicator, and vertical speed indicator are the three most common pitot-static instruments. ***Figure 7-9*** illustrates a simple pitot-static system connected to these three instruments.

A pitot tube includes an opening which faces into the airstream to receive the full force of the impact air pressure as the aircraft moves forward. This air passes

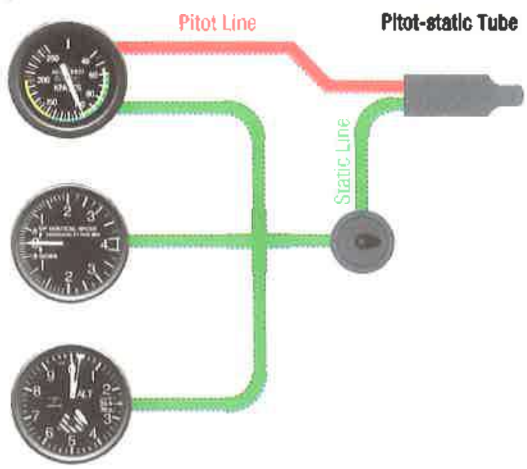


Figure 7-9. A simple pitot-static system.

through a baffled plate designed to protect the system from moisture and dirt entering the tube. Below the baffle, a drain hole is provided allowing moisture to escape. The ram air is directed aft to a chamber in the assembly. An upright tube, or riser, leads this pressurized air out of the pitot assembly to the airspeed indicator. The aft section of the pitot tube is equipped with small holes on the top and bottom surfaces that are designed to collect air pressure that is at atmospheric pressure in a static, or still, condition. (***Figure 7-10***)

Many pitot-static tube heads contain heating elements to prevent icing during flight. The pilot can send electric current to the element with a switch in the cockpit when ice forming conditions exist. Often, this switch is wired through the ignition switch so that when the aircraft is shut down, a pitot tube heater inadvertently left on does not continue to draw current and drain the battery. Caution should be exercised when near the pitot tube as these heating elements make the tube too hot to

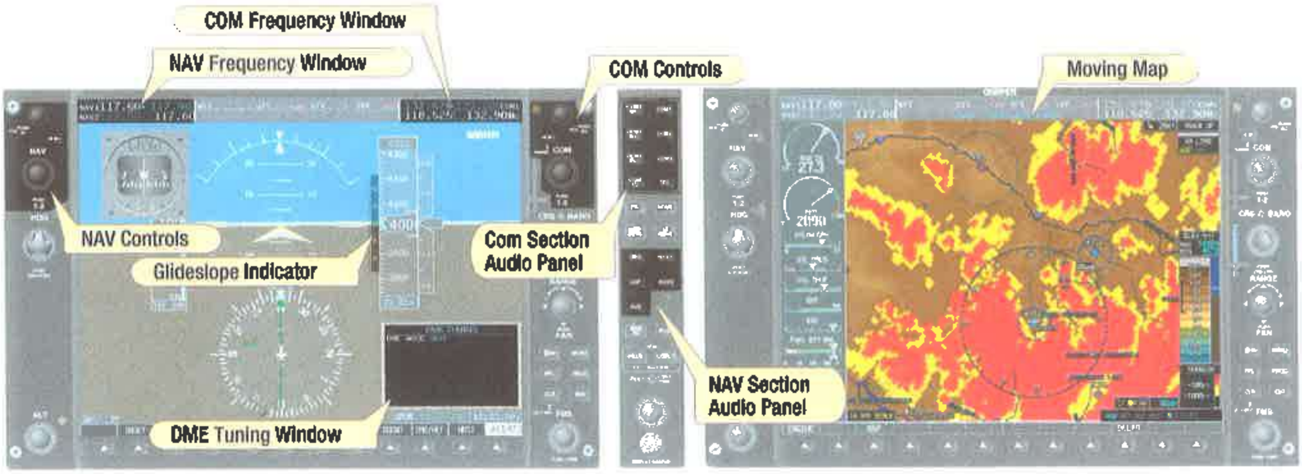


Figure 7-8. Navigation instruments.

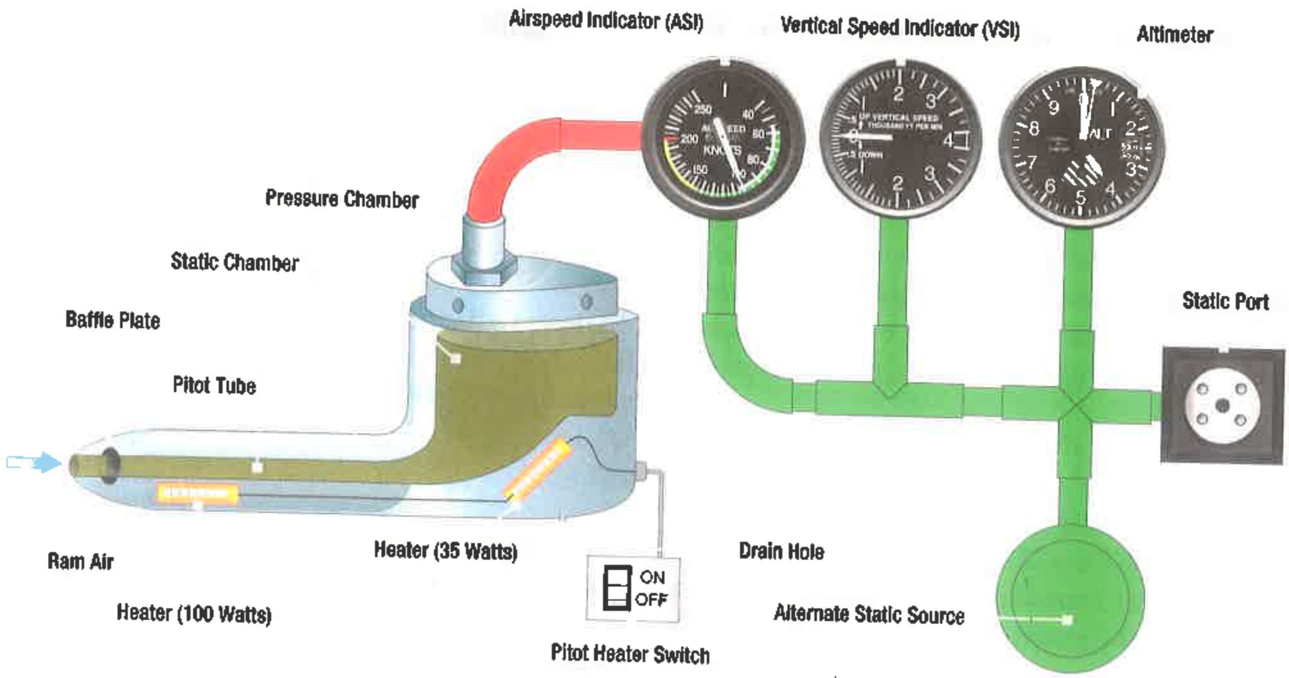


Figure 7-10. A typical pitot-static system head.

touch without receiving a burn. The pitot-static tube is mounted on the outside of the aircraft at a point where the air is least likely to be turbulent. It is pointed in a forward direction parallel to the aircraft line of flight. The location may vary. Most of them are under the cockpit, but some can be on a lateral side of the fuselage. Various designs exist but the function remains the same: to capture impact air pressure and static air pressure and direct them to the proper instruments.
(*Figure 7-11 and Figure 7-12*)

Another type of pitot-static system provides for the location of the pitot and static sources to be at separate positions on the aircraft. The pitot tube in this arrangement is used only to gather ram air pressure. Separate static vents are used to collect static air pressure information. Usually, these are located flush on the side of the fuselage. (*Figure 7-13*) There may be two or more vents. A primary and alternate vent is typical, as well as separate dedicated vents for the pilot and co-pilot instruments to ensure that there is always one set of flight instruments operable in case of a malfunction.

In some aircraft, multiple vents are located on opposite sides of the fuselage and connected with Y tubing for input to the instruments. This is done to compensate for any variations in static air pressure on the vents due to the aircraft attitude. Static vents may also be heated to prevent icing. Additionally, autopilot systems, and computers need to be connected to pitot/static air information.

Figure 7-11. Pitot static system heads, or pitot tubes-under.

Figure 7-12. Pitot static system heads, or pitot tubes-lateral.

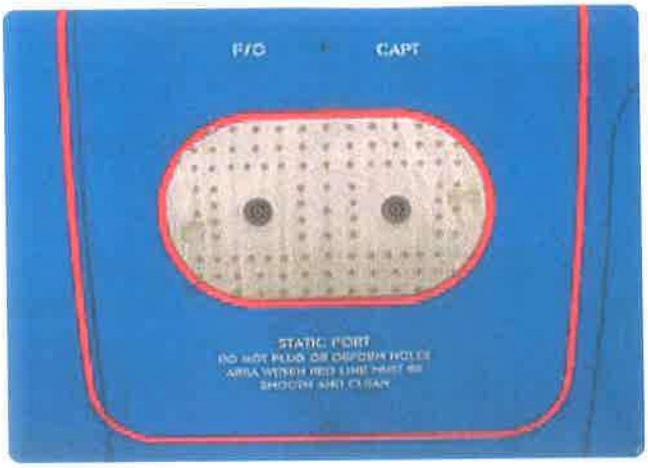


Figure 7-13. Heated primary and alternate static vents.

ALTIMETERS

An altimeter indicates the height of the aircraft above a predetermined level, such as sea level or the terrain beneath the aircraft. The most common way to measure this distance is rooted in discoveries made centuries ago. Seventeenth century work proving that the air in the atmosphere exerts pressure on the things around us led Evangelista Torricelli to the invention of the barometer. Also in that century, Blaise Pascal was able to show that a relationship exists between altitude and air pressure. As altitude increases, air pressure decreases. The amount of decrease is measurable and consistent for any given altitude change. Therefore, by measuring air pressure, altitude can be determined. (***Figure 7-14***)

Altimeters that measure altitude by measuring the ambient pressure of the atmospheric air are known as pressure altimeters. Pressure altimeters are connected to the static vent(s) via tubing. The relationship between the measured pressure and the altitude is indicated on the instrument face, which is calibrated in feet. These devices are direct reading instruments that measure absolute pressure. An aneroid bellows is the core of the pressure altimeter inner workings. Attached to this sealed diaphragm are the linkages and gears, connecting it to the indicating pointer. Static air pressure enters the airtight instrument case and surrounds the aneroid. At sea level, the altimeter indicates zero when this pressure is exerted by the ambient air on the aneroid. As air pressure is reduced by moving the altimeter higher in the atmosphere, the aneroid expands and displays altitude on the instrument by rotating the pointer. As the altimeter is lowered in the atmosphere, the air pressure around the aneroid increases and the pointer moves in the opposite direction. (***Figure 7-15***)

Atmosphere Pressure	
Altitude (ft)	Pressure (psi)
Sea Level	14.69
2 000	13.66
4 000	12.69
6 000	11.77
8 000	10.91
10 000	10.10
12 000	9.34
14 000	8.63
16 000	7.96
18 000	7.34
20 000	6.75
22 000	6.20
24 000	5.69
26 000	5.22
28 000	4.77
30 000	4.36
32 000	3.98
34 000	3.62
36 000	3.29
38 000	2.99
40 000	2.72
42 000	2.47
44 000	2.24
46 000	2.04
48 000	1.85
50 000	1.68

Figure 7-14. Air pressure.

The face, or dial of an analog altimeter is read similarly to a clock. As the longest pointer moves around the dial, it is registering the altitude in hundreds of feet. One complete revolution of this pointer indicates 1 000 feet of altitude. The second longest point moves more slowly. Each time it reaches a numeral, it indicates 1 000 feet of altitude. Once around the dial for this pointer is equal to 10 000 feet. If so equipped, a third, shortest pointer registers altitude in 10 000 foot increments. Sometimes a black and white or red and white cross hatched area is shown on the face of the instrument until the 10 000 foot level has been reached. (***Figure 7-16***)

On a helicopter, as the typical function is flight near the ground, an altimeter's precision is most important from 0 to 3 000 feet, and so the 10 000 foot scale is often omitted.

Many altimeters also contain linkages that rotate a numerical counter in addition to a moving pointer. This quick reference window allows the pilot to simply read the numerical altitude in feet. However, the motion of the rotating digits during rapid climb or descent makes it difficult or impossible to read the numbers. Thus,

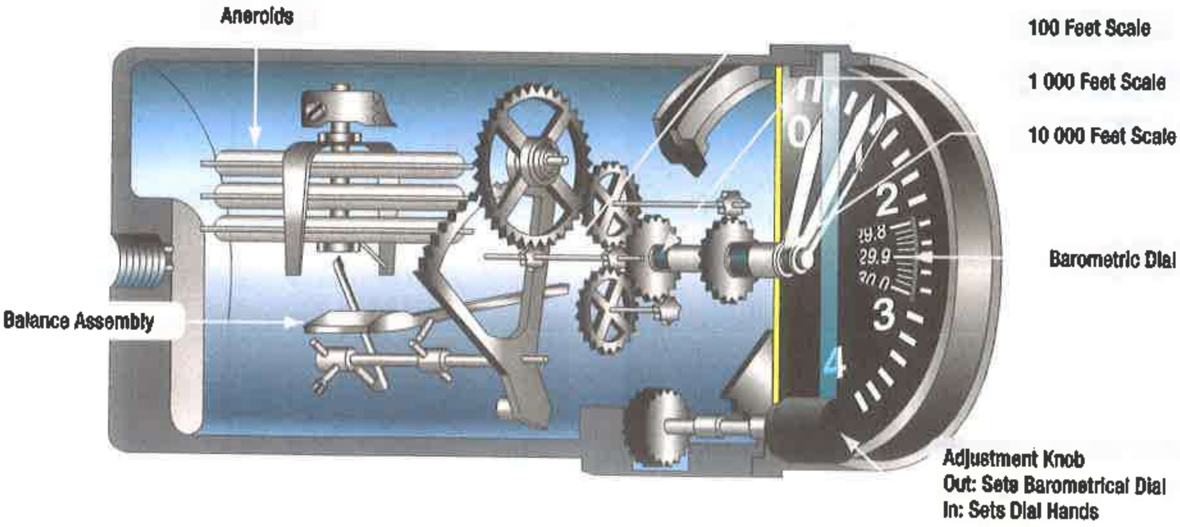


Figure 7-15. The internal arrangement of a sealed diaphragm pressure altimeter.

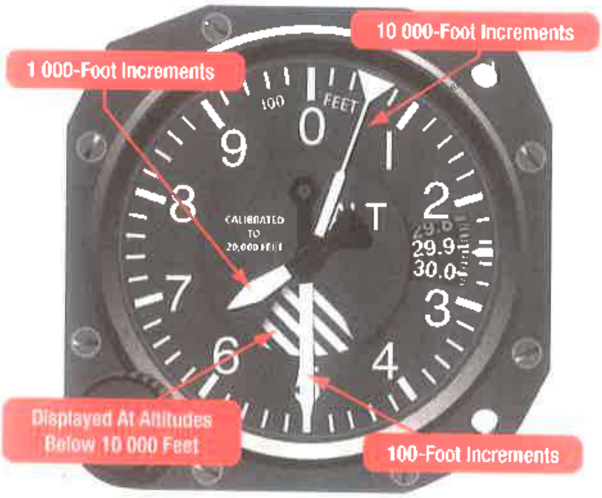


Figure 7-16. A sensitive altimeter with three pointers and a crosshatched area.

reference can then be directed to the classic clock style indication. ***Figure 7-17*** illustrates the inner workings behind this type of mechanical digital display of pressure altitude.

True digital instrument displays can show altitude in numerous ways. Use of a numerical display is most common. Often a digital display of altitude is also given electronically near the artificial horizon depiction. A linear vertical scale may also be presented to put this hard numerical value in perspective. An example of this type of display is shown in ***Figure 7-18***.

Accurate measurement of altitude is important for numerous reasons. The importance is magnified during Instrument Flight Rules (IFR) conditions. For example, avoidance of tall obstacles and rising terrain relies on precise altitude indication, as does flying at a prescribed altitude assigned by Air Traffic Control (ATC) to avoid colliding with other aircraft.

Measuring altitude with a pressure measuring device is fraught with complications. Steps are taken to refine these indications to compensate for factors that may cause inaccuracy. One major factor that affects measurements is the naturally occurring pressure variations throughout the atmosphere due to weather. Different air masses develop and move over the Earth's surface, each with inherent pressure characteristics. These air masses cause the weather we experience, especially at the boundary areas between air masses known as fronts. Accordingly, at sea level, even if the temperature remains constant, air pressure rises and falls as weather systems come and go. The values given in ***Figure 7-14*** are averages for theoretical purposes.

To maintain altimeter accuracy despite varying pressure, a method for setting the altimeter was devised. An adjustable pressure scale on the face of an analog altimeter is known as a barometric or Kollsman window. This is adjusted to read the existing atmospheric pressure when the pilot rotates the knob on the instrument. This adjustment is linked through gears inside the altimeter to move the altitude indicating pointers on the dial. A setting of 29.92"Hg indicates standard pressure at sea level. However by putting the current known air pressure (also known as altimeter setting) in the window, the instrument indicates the actual altitude. This altitude, adjusted for atmospheric pressure changes,

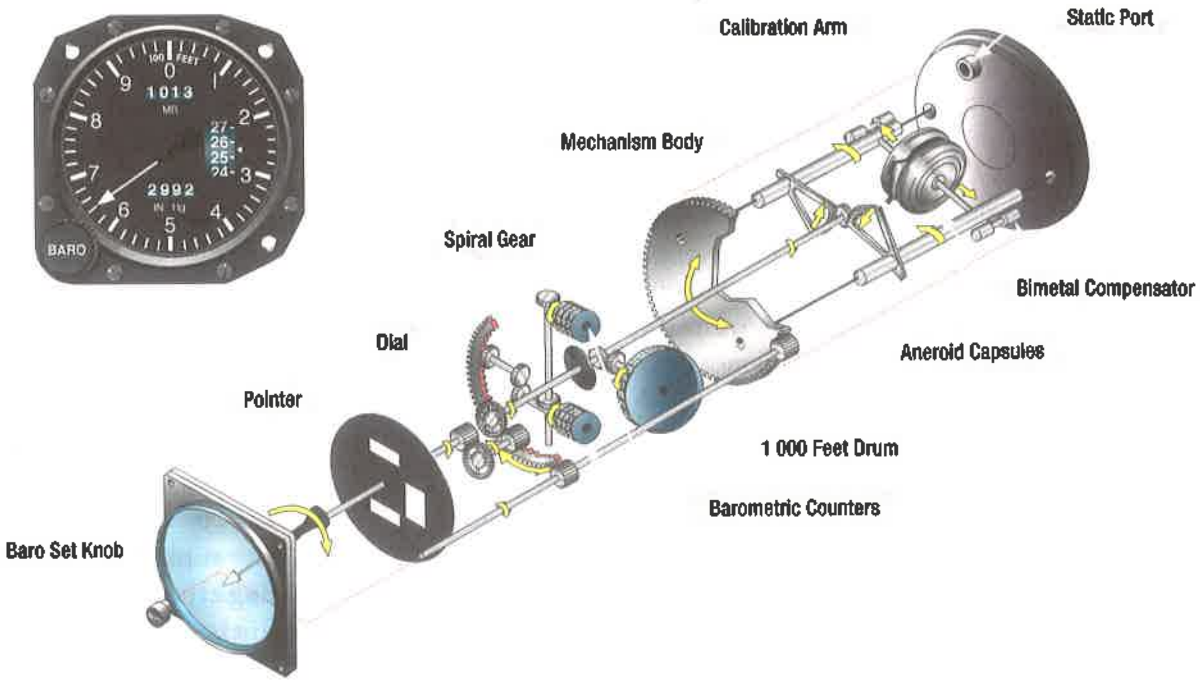


Figure 7-17. A drum-type counter.

is known as the indicated altitude. In flight the altimeter setting is often changed to match that of the closest available weather reporting station or airport. This keeps the altimeter accurate as the flight progresses.

When all aircraft reference this standard pressure level of 29.92"Hg, vertical separation between aircraft assigned to different altitudes by ATC can be assured, of course assuming that all altimeters are functioning

Figure 7-18. This primary flight display unit.

properly, and pilots hold their assigned altitudes. Note that the true altitude or actual height of an aircraft above sea level is only the same as the pressure altitude when standard conditions exist. Otherwise, all aircraft with altimeters set to 29.92"Hg could have true altitudes higher or lower than the pressure altitude indicated.

Temperature also affects the accuracy of an altimeter. As the aneroid diaphragms used in altimeters are usually made of metal, their elasticity changes as its temperature changes. This can lead to a false indication, especially at high altitudes when the ambient air is very cold. To correct this, a bimetallic compensating device is built into many sensitive altimeters to compensate for these varying temperatures.

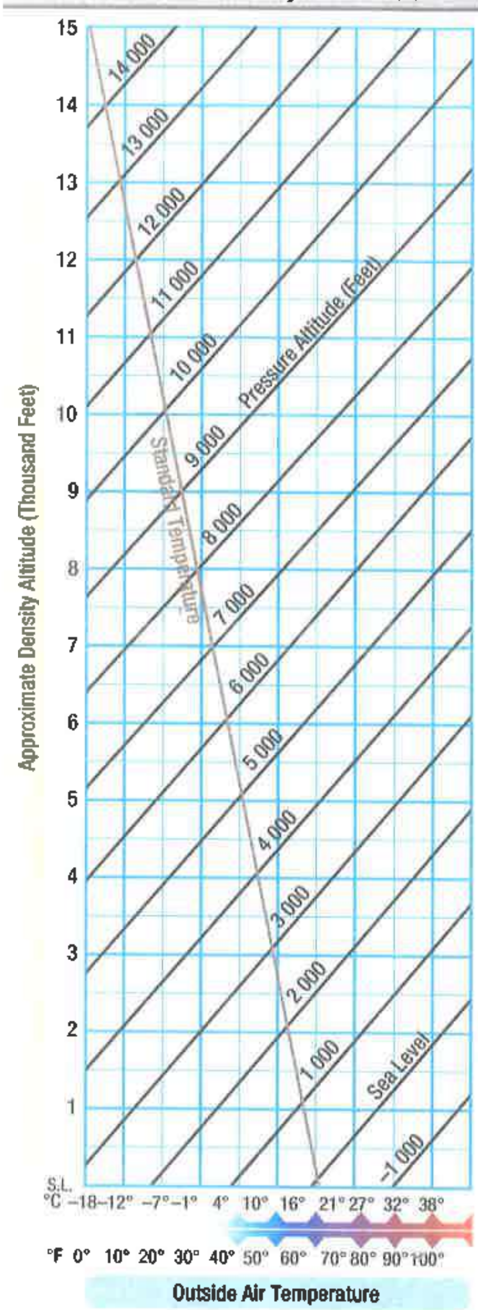
Figure 7-17 shows such a device on a drum type altimeter. Temperature also affects air density, which has a great impact on the performance of an aircraft. Although this does not cause the altimeter to produce an errant reading, flight crews must be aware that performance changes with temperature variations in the atmosphere. The term density altitude describes altitude corrected for nonstandard temperature. That is, the density altitude is the pressure altitude at which an aircraft would experience similar performance as it would on the actual day being experienced. For example, on a very cold day, the air is more dense than on a standard day, so an aircraft performs as though it is at a lower altitude. On an extremely hot day the reverse is true. An aircraft performs as though it were at a higher elevation where the air is less dense. Conversion factors and charts have been produced so pilots can calculate the density altitude on any particular day. Inclusion of nonstandard air pressure due to humidity can also be factored.

So, while the effects of temperature do not cause an altimeter to indicate falsely, an altimeter indication can be misleading in terms of aircraft performance if these effects are not considered. (*Figure 7-19*)

Other factors can cause an inaccurate altimeter indication. Scale error is a mechanical error whereby the scale of the instrument is not accurately aligned with the pointers. Periodic testing and adjustment by trained technicians using calibrated equipment ensures this is kept to a minimum.

Position error, or installation error, is an inaccuracy caused by the location of the static vent that supplies the altimeter. While every effort is made to place static vents in undisturbed air, airflow over the airframe changes with the speed and attitude of the aircraft. The amount of this air pressure collection error is measured in test flights, and a correction table showing the variances can be included with the altimeter for the pilot's use. (*Figure 7-20*) In some modern aircraft a position error can be removed by the Air Data Computer (ADC), so the pilot needs not be concerned about this inaccuracy.

Density Altitude Chart



Altimeter Setting ("Hg)	Pressure Altitude Conversion Factor
28.0	1 824
28.1	1 727
28.2	1 630
28.3	1 533
28.4	1 436
28.5	1 340
28.6	1 244
28.7	1 148
28.8	1 053
28.9	957
29.0	863
29.1	768
29.2	673
29.3	579
29.4	485
29.5	392
29.6	298
29.7	205
29.8	112
29.9	20
29.92	0
30.0	−73
30.1	−165
30.2	−257
30.3	−348
30.4	−440
30.5	−531
30.6	−622
30.7	−712
30.8	−803

Figure 7-19. The effect of air temperature on aircraft performance.

Static system leaks can affect the static air input to the altimeter resulting in inaccurate altimeter indications. It should also be understood that analog altimeters are mechanical devices that often reside in a hostile environment. The significant vibration and temperature range swings encountered by the instruments and the pitot-static system (the tubing connections and fittings) can sometimes create damage or a leak, leading to instrument malfunction. Proper care upon installation is the best preventive action. In all cases, static system maintenance includes leak checks every 24 months, regardless of whether any discrepancy has been noticed.

In addition, the mechanical nature of the altimeter diaphragm pressure measuring apparatus has limitations. The diaphragm itself is elastic when responding to static air pressure changes. Hysteresis is a function of a material caused by the natural reluctance of a pressure sensing material such as a diaphragm to return to its original position and shape after being mechanically deformed during long periods such as level flight. If followed by an abrupt altitude change, the indication lags or responds slowly while expanding or contracting. While temporary, this limitation does cause an inaccurate altitude indication.

AIRSPEED INDICATORS

The airspeed indicator is also a differential pressure gauge. Ram air pressure from the aircraft pitot tube is directed into a diaphragm in the analog airspeed instrument's case. Static air pressure from the aircraft static vent(s) is directed into the case surrounding the diaphragm. As the speed of the aircraft varies, the ram air pressure varies, expanding or contracting the diaphragm. Linkage attached to the diaphragm causes a pointer to move over the instrument face, which is calibrated in knots or miles per hour. (***Figure 7-21***)

Figure 7-20. The location of the static vent.

The relationship between ram air pressure and static air pressure produces what is known as indicated airspeed. As with the altimeter, there are other factors that must be considered in measuring airspeed throughout all phases of flight. These can cause inaccurate readings or indications that are not useful to the pilot in a particular situation. In analog airspeed indicators, the factors are often compensated for with mechanisms inside the case and on the instrument dial.

Digital flight instruments can have calculations performed in the ADC so the desired accurate indication is displayed. While the relationship between ram air pressure and static air pressure is the basis for most airspeed indications, it can be even more accurate. Calibrated airspeed considers errors due to the position of the pitot-static pickups. It also corrects for the nonlinear nature of the pitot-static pressure differential when it is displayed on a linear scale. Analog airspeed indicators come with a correction chart that allows cross referencing of indicated airspeed to calibrated airspeed for various flight conditions. These differences are typically small and often ignored.

Additionally, indicated airspeed does not consider temperature and air pressure differences needed to indicate true airspeed. True airspeed is the same as indicated airspeed when standard conditions exist. But when atmospheric temperature or pressure varies, the relationship between the ram and static pressure alters. Analog airspeed instruments often include bimetallic temperature compensating devices that can alter the linkage movement between the diaphragm

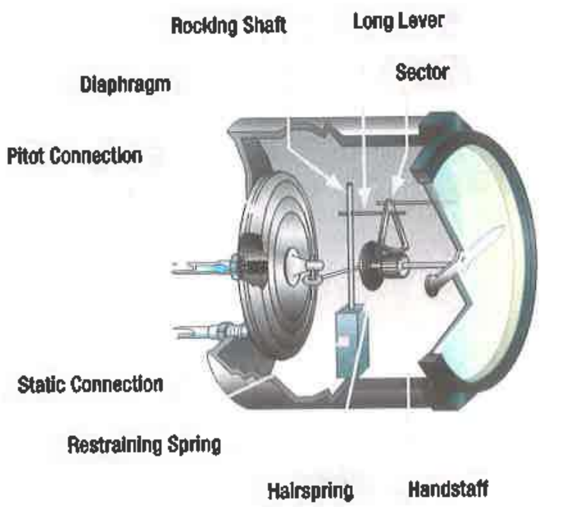


Figure 7-21. An airspeed indicator is a differential pressure gauge.

and the pointer movement. An aneroid inside the airspeed indicator case can also be used to compensate for non-standard pressures. Alternatively, airspeed indicators exist that allow the pilot to set temperature and pressure variables manually with external knobs on the instrument. The knobs rotate the dial face and internal linkages to compensate for nonstandard temperature and pressure, resulting in a true airspeed indication. (***Figure 7-22***)

Digital flight instrument systems perform these calculations for true airspeed in the ADC. Ram air from the pitot tube, static air from the static vent(s) and temperature information are run into the sensing portion of the computer. The ADC then calculates these variables so a true airspeed value can be digitally sent to the display. See ***Figure 7-18*** for the display of airspeed information on the primary flight display on a light aircraft. Note that the placement of the airspeed indicator in the standard T configuration is just left of the artificial horizon display.

VERTICAL SPEED INDICATOR

An analog Vertical Speed Indicator (VSI) may also be referred to as a Vertical Velocity Indicator (VVI), or rate-of-climb indicator. It is a direct reading differential pressure gauge that compares static pressure from the aircraft static system with static pressure surrounding the diaphragm in the instrument case. Air is free to flow unrestricted in and out of the diaphragm but is made to flow in and out of the case through a calibrated orifice. A pointer attached to the diaphragm indicates zero vertical speed when the pressure inside and outside the diaphragm are the same. The dial is usually graduated in 100s of feet per minute. A zeroing adjustment screw, or knob, on the face of the instrument is used to center the pointer exactly on zero while the aircraft is on the ground. (***Figure 7-23***)

As the aircraft climbs, the unrestricted air pressure in the diaphragm lowers as the air becomes less dense. The case air pressure surrounding the diaphragm lowers more slowly, having to pass through the restriction created by the orifice. This causes unequal pressure inside and outside the diaphragm, which in turn causes the diaphragm to contract and so its pointer indicates a climb. The process works in reverse in a descent. If a steady climb or descent is maintained, a steady pressure differential is established between the diaphragm and the case pressure surrounding it, resulting in an accurate indication of the rate of climb via markings on the instrument face. (***Figure 7-24***)

A shortcoming of the rate-of-climb mechanism as described, is that there is a lag of six to nine seconds before a stable differential pressure can be established, indicating the climb or descent of the aircraft. An Instantaneous Vertical Speed Indicator (IVSI) has a built-in mechanism to reduce this lag. A lightly sprung dashpot, or piston, reacts to the direction change of an abrupt climb or descent. As this accelerometer does so, it pumps air into or out of the diaphragm, hastening the establishment of the pressure differential and so causing an appropriate indication. (***Figure 7-25***)

Figure 7-22. An analog true airspeed indicator.

Figure 7-23. A vertical speed indicator.

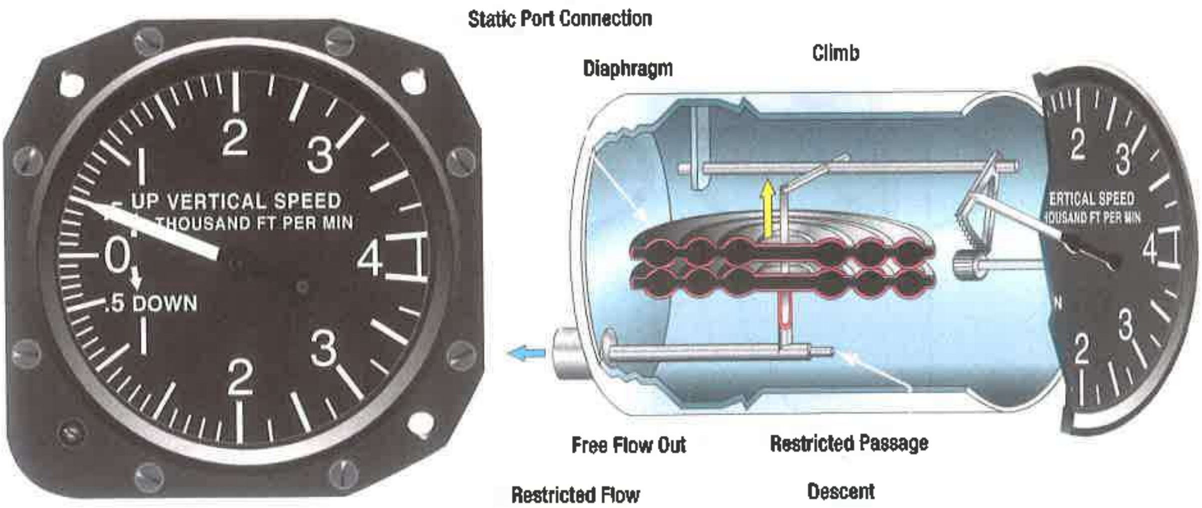


Figure 7-24. The VSI is a differential pressure gauge.

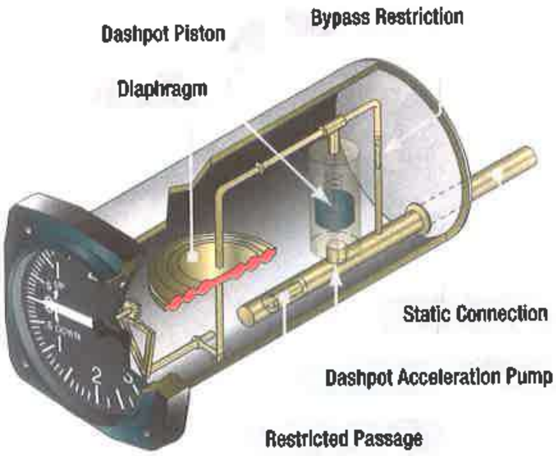


Figure 7-25. The small dashpot in this IVSI.

The rate-of-climb indication in a digitally displayed instrument system is computed from static air input to the ADC. An aneroid, or solid state pressure sensor continuously reacts to changes in static pressure. The digital clock within the ADC replaces the calibrated orifice of an analog instrument. As the static pressure changes, the computer clock can develop a rate for the change. Using the known lapse rate conversion for air pressure as altitude increases or decreases, a figure for climb or descent in Feet Per Minute (FPM) can be calculated. The vertical speed is often displayed near the altimeter information on the primary flight display. (***Figure 7-18***)

GYROSCOPIC INSTRUMENTS

ARTIFICIAL HORIZON, ATTITUDE DIRECTOR, DIRECTION INDICATOR, HORIZONTAL SITUATION INDICATOR, TURN AND SLIP INDICATOR, TURN COORDINATOR

Gyroscopic instruments are essential instruments used on all aircraft. They provide the pilot with critical attitude and directional information and are particularly important while flying under IFR. On many small aircraft, electric turn-and-bank or turn coordinators are combined with vacuum powered attitude and directional gyro instruments as a means for redundancy. On more complex aircraft, reliable and redundant electrical systems make all-electric powered gyro instruments possible. Note that electric gyro instruments have some sort of indicator on the dial to show when the instrument is not receiving power. Usually, this is in the form of a red flag with the word OFF written on it. Three of the most common flight instruments, the attitude indicator, heading indicator, and turn needle of the turn-and-bank indicator, are gyroscopic.

GYROSCOPIC PRINCIPLES

To understand how these instruments operate, knowledge of gyroscopic principles and instrument power systems is required. A mechanical gyroscope is comprised of a wheel or rotor with its mass concentrated around its perimeter. The rotor has bearings to enable it to spin at high speeds. (***Figure 7-26A***) Different mounting configurations are available for the rotor and axle, which allow the rotor assembly to rotate about one

Figure 7-26. Gyroscopes.

or two axes perpendicular to its axis of spin. To suspend the rotor for rotation, the axle is first mounted in a supporting ring. (***Figure 7-26B***) If brackets are attached 90° around the supporting ring from where the spin axle is attached, the supporting ring and rotor can both move freely 360°. In this configuration, the gyro is said to be a captive gyro. It can rotate about only one axis that is perpendicular to the axis of spin. (***Figure 7-26C***)

The supporting ring can also be mounted inside an outer ring. The bearing points are the same as the bracket just described, 90° around the supporting ring from where the spin axle is attached. Attachment of a bracket to this outer ring allows the rotor to rotate in two planes while spinning. Both are perpendicular to the spin axis of the rotor. The plane that the rotor spins in due to its rotation about its axle is not counted as a plane of rotation. A gyroscope with this configuration, two rings plus the mounting bracket, is said to be a free gyro because it is free to rotate about two axes that are both perpendicular to the rotor spin axis. (***Figure 7-26D***)

As a result, the supporting ring with a spinning gyro mounted inside is free to turn 360° inside the outer ring. Unless the rotor of a gyro is spinning, it has no unusual properties; it is simply a wheel universally mounted. When the rotor is rotated at a high speed, the gyro exhibits a couple of unique characteristics. The first is called gyroscopic rigidity or rigidity in space. This means that the rotor of a free gyro always points in the same direction no matter which way the base of the gyro is positioned. (***Figure 7-27***)

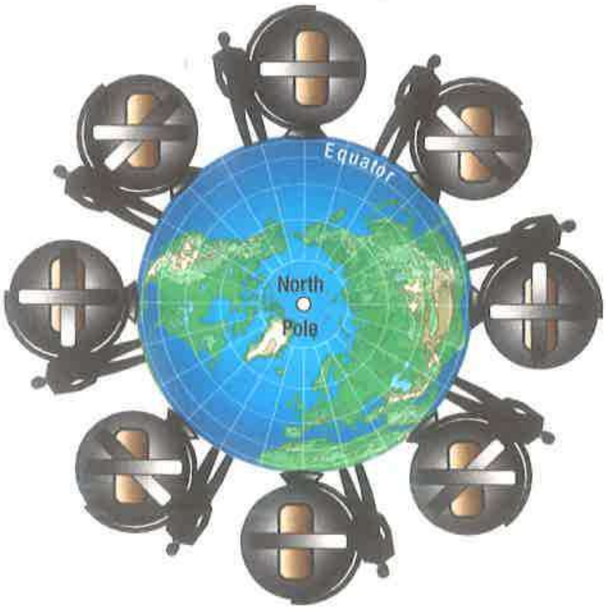


Figure 7-27. Once spinning, a free gyro rotor stays oriented in the same position.

Gyroscopic rigidity depends upon several design factors:

- Weight for a given size - a heavy mass is more resistant to disturbing forces than a light mass.
- Angular velocity - the higher the rotational speed, the greater the rigidity or resistance to deflection.
- Radius at which the weight is concentrated - maximum effect is obtained when its principal mass is concentrated near the rim and rotating at high speed.
- Bearing friction - any friction applies a deflecting force to a gyro. Minimum bearing friction keeps these deflecting forces at a minimum.

The characteristic of gyros to remain rigid in space is exploited in the attitude and the directional indicators that use gyros.

Precession is a second characteristic of gyroscopes. When applying a force to the horizontal axis of the gyro, the applied force is resisted. Instead of responding to the force by moving about the horizontal axis, the gyro moves in response about its vertical axis. Stated another way, an applied force to the axis of the spinning gyro does not cause the axis to tilt. Rather, the gyro responds as though the force was applied 90° from the direction of rotation of the gyro rotor. The gyro rotates rather than tilts. (***Figure 7-28***) This predictable precession is utilized in a turn and bank instrument.

Ring Laser Gyro

Improved attitude and direction information is always a goal in aviation. Modern aircraft make use of highly accurate solid state attitude and directional devices with no moving parts resulting in high reliability and low maintenance.

The Ring Laser Gyro (RLG) is widely used in commercial aviation. The basis for RLG operation is that it takes time for light to travel around a stationary, non-rotating circular path. Light takes longer to complete the journey if the path is rotating in the same direction as the light is traveling and takes less time to complete the loop if the path is rotating in the direction opposite to that of the light. Essentially, the path is made longer or shorter by the rotation of the path. (***Figure 7-29***) This is known as the Sagnac effect.

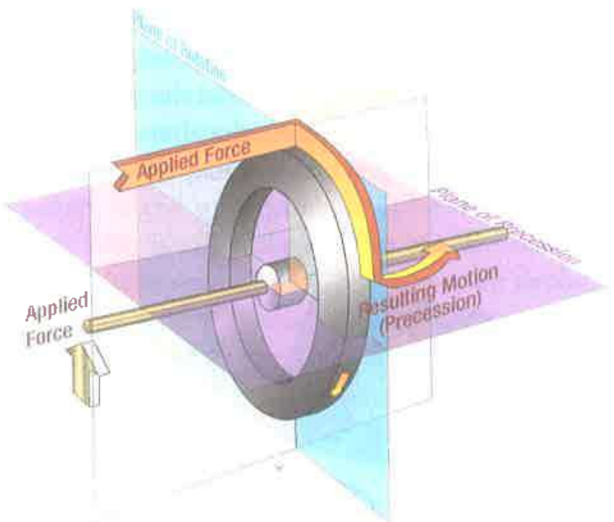


Figure 7-28. When a force is applied to a spinning gyroscope.

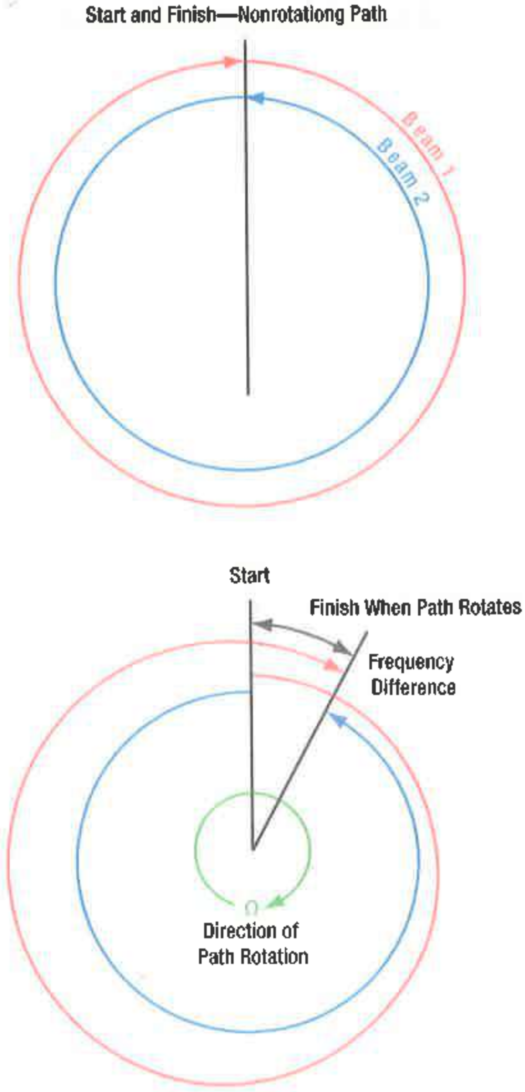


Figure 7-29. Light traveling in opposite directions around a non-rotating path.

Laser is an acronym for Light Amplification by Stimulated Emission of Radiation. A laser operates by exciting atoms in plasma to release electromagnetic energy, or photons. An RLG produces laser beams that travel in opposite directions around a closed triangular cavity. The wavelength of the light traveling around the loop is fixed. As the loop rotates, the path the lasers must travel lengthens or shortens. The light wavelengths then compress or expand to complete travel around the loop as the loop changes its effective length. As the wavelengths change, the frequencies also change.

By examining the difference in the frequencies of the two counter rotating beams of light, the rate at which the path is rotating can be measured. A piezoelectric dithering motor in the center of the unit vibrates to prevent lock-in of the output signal at low rotational speeds. It causes units installed on aircraft to hum when operating. (*Figure 7-30*)

An RLG is remotely mounted so the cavity path rotates around one of the axes of flight. The rate of frequency phase shift detected between the counter rotating lasers is proportional to the rate that the aircraft is moving about that axis. On aircraft, an RLG is installed for each axis of flight. Output can be used in analog instrumentation and autopilot systems. It is also compatible for use by digital display computers and for digital autopilot computers. RLGs are rugged and have a long service life with little maintenance due to their lack of moving parts. They are extremely accurate and generally are considered superior to mechanical gyroscopes.

Micro Electro Mechanical System (MEMS)

On aircraft, MEMS devices save space and weight. With solid state MEMS devices, reliability is increased primarily due to the lack of moving parts. The development of MEMS technology integrates with the use of air data computers. This newest improvement in technology is low cost and promises to proliferate through all forms of aviation.

Tiny vibration based units with resistance and capacitance measuring pick-offs are accurate and reliable and only a few millimeters in length and width. They are normally integrated into a complete micro-electronic solid state chip designed to yield an output after various conditioning processes are performed. The chips, which are analogous to tiny circuit boards, can be packaged for installation inside a dedicated computer or module that is installed on the aircraft.

While a large mechanical gyroscope spins in a plane, its rigidity in space is used to observe and measure the movement of the aircraft. The basis of operation of many MEMS gyroscopes is the same despite their tiny size. The difference is that a vibrating or oscillating piezoelectric device replaces the spinning, weighted ring of the mechanical gyro. Once set in motion, any out of plane motion is detectable by varying micro-voltages or capacitances detected through geometrically arranged pickups. Since piezoelectric substances have a relationship between movement and electricity, micro-electrical stimulation sets a piezo electric gyro in motion and the tiny voltages produced via the movement in the piezo are extracted. They are then input as the required variables needed to compute attitude or direction information. (*Figure 7-31*)

In many aircraft, Attitude Heading Reference Systems (AHRS) have taken the place of the gyroscope and other individual instruments. While MEMS devices provide part of the attitude information for the system, GPS, solid state magnetometers, solid state accelerometers, and digital air data signals are all combined in an AHRS to output highly reliable information for display on a cockpit panel. (*Figure 7-32*)

ARTIFICIAL HORIZON

The attitude indicator, or artificial horizon, is one of the most essential flight instruments. It gives the pilot pitch and roll information that is especially important when flying without outside visual references. The attitude indicator operates with a gyroscope rotating in the horizontal plane. Thus, it mimics the actual horizon through its rigidity in space. As the aircraft pitches and rolls in relation to the actual horizon, the gyro gimbals

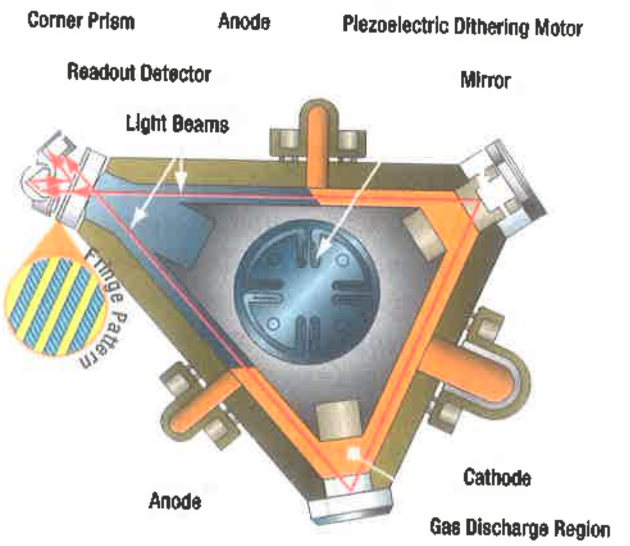


Figure 7-30. The ring laser gyro.

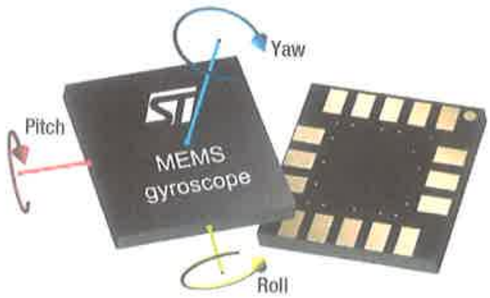


Figure 7-31. A MEMS piezo-electric gyroscope.

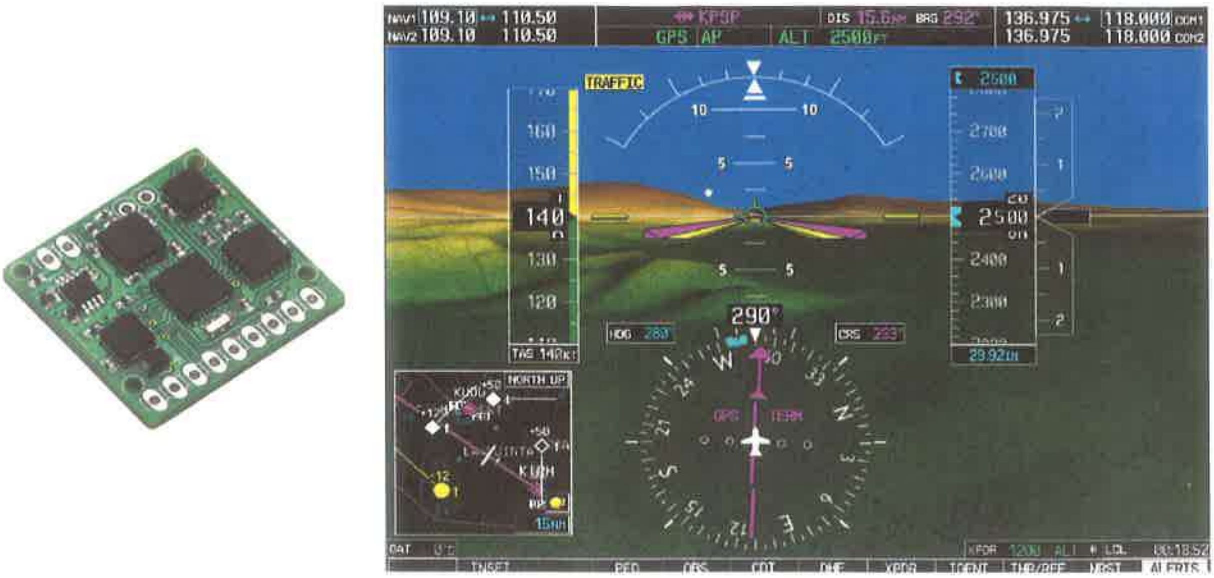
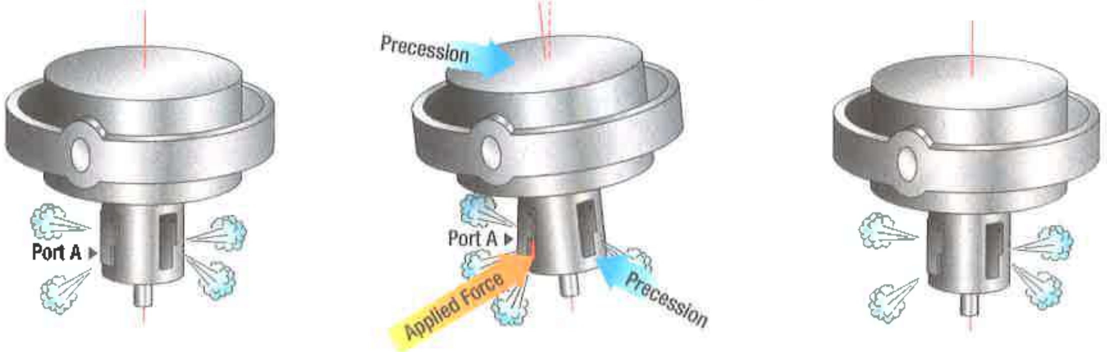


Figure 7-32. This complete AHRS circuit board (about 3x3 cm) combines three gyro axes, three accelerometers, three magnetic compass axes and data processing capacity to produce a variety of visual displays such as here.

Figure 7-33. A vacuum-driven attitude indicator.

allow the aircraft and instrument housing to pitch and roll around the gyro rotor that remains parallel to the ground. A horizontal representation of the airplane is fixed to the instrument housing. A painted semi-sphere simulating the horizon, the sky, and the ground is attached to the gyro gimbals. The sky and ground meet at what is called the horizon bar. The relationship between the horizon bar and the miniature airplane are the same as those of the aircraft and the actual horizon. Graduated scales reference the degrees of pitch and roll. Often, an adjustment knob allows pilots of varying heights to place the horizon bar at an appropriate level. (*Figure 7-33*)

In a typical vacuum driven gyro system, air is sucked through a filter and then through the attitude indicator in a manner that spins the gyro rotor inside. An erecting mechanism is built into the instrument to assist in keeping the gyro rotor rotating in the intended plane. Precession caused by bearing friction makes this necessary. After air engages the scalloped drive on the rotor, it flows from the instrument to the vacuum pump through four ports. These ports all exhaust the same amount of air when the gyro is rotating in plane. When the gyro rotates out of plane, air tends to port out of one side more than another. Vanes close to prevent this, causing more air to flow out of the opposite side. The force from this unequal venting of the air re-erects the gyro rotor. (*Figure 7-34*)



Exhaust air equal in all directions gyro erect.

Gyro precesses, increasing exhaust from port A.

Precessing force at port A erects gyro, exhaust air again equal at all ports.

Figure 7-34. The erecting mechanism on a vacuum-driven gyro.

Early vacuum driven attitude indicators were limited in how far the aircraft could pitch or roll before the gyro gimbals contacted stops, causing abrupt precession and tumbling of the gyro. Many early gyros include a caging device. It is used to erect the rotor to its normal operating position prior to flight or after tumbling. A flag indicates that the gyro must be uncaged before use. Modern gyroscopic instruments are built so they do not tumble, regardless of the angular movement of the aircraft about its axes.

In addition to the contamination potential introduced by the air drive system, other shortcomings exist in the performance of vacuum driven attitude indicators. Some are induced by the erection mechanism. The pendulous vanes that move to direct airflow out of the gyro respond not only to forces caused by a deviation from the intended plane of rotation, but also centrifugal force experienced during turns causes the vanes to allow asymmetric porting of the gyro vacuum air. The result is an inaccurate display of the aircraft attitude, especially in skids and steep banked turns. Also, abrupt acceleration and deceleration imposes forces on the gyro rotor. Suspended in its gimbals, it acts like an accelerometer, resulting in a false nose up or nose down indication. Pilots must learn to recognize these errors and adjust accordingly.

ATTITUDE DIRECTOR

Electric attitude directors are like vacuum driven gyro indicators (artificial horizon). The main difference is in the drive mechanism. Inside the gimbals of an electric gyro, a small squirrel cage electric motor is the rotor. It is typically driven by 115 volt, 400 cycle AC turning at approximately 21 000 RPM. Other characteristics of the vacuum driven gyro are shared by the electric gyro. The rotor is still oriented in the horizontal plane. The free gyro gimbals allow the aircraft and instrument case to rotate around the gyro rotor that remains rigid in space. A miniature airplane fixed to the instrument case indicates the aircraft attitude against the moving horizon bar behind it. Electric attitude indicators address some of the shortcomings of vacuum driven indicators. Since there is no air flowing through an electric attitude indicator, air filters, regulators, plumbing lines, and vacuum pumps are not needed. Contamination from dirt in the air is not an issue, resulting in longer bearing life and less precession. Erection mechanism ports are not needed, so pendulous vanes responsive to centrifugal forces are also eliminated.

It is still possible that this gyro may experience precession and need to be erected. This is done with magnets rather than vent ports. A magnet attached to the top of the gyro shaft spins at approximately 21 000 RPM. Around this magnet, but not attached to it, is a sleeve that is rotated by magnetic attraction at approximately 44 to 48 RPM. Steel balls are free to move around the sleeve. If the pull of gravity is not aligned with the axis of the gyro, the balls fall to the low side. The resulting precession re-aligns the axis of rotation vertically. Typically, electric attitude indicator gyros can be caged manually by a lever and cam mechanism to provide rapid erection. When the instrument is not getting sufficient power for normal operation, an OFF flag appears in the upper right hand face of the instrument. (***Figure 7-35***)

DIRECTION INDICATOR

The gyroscopic direction indicator or Directional Gyro (DG) is often the primary instrument for direction.

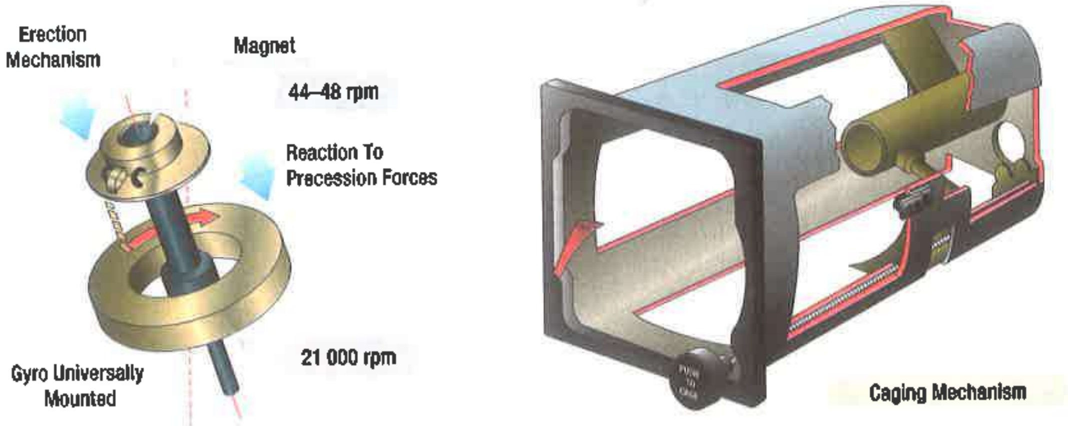


Figure 7-35. Erecting and caging mechanisms.

Because a magnetic compass fluctuates so much, a gyro aligned with the magnetic compass gives a much more stable heading indication. Gyroscopic direction indicators are located at the center base of the instrument panel basic T. A vacuum powered DG is common on many light aircraft. Its basis for operation is its gyro rigidity in space. The gyro rotor spins in the vertical plane and stays aligned with the direction to which it is set. The aircraft and instrument case moves around the rigid gyro. This causes a vertical compass card that is geared to the rotor gimbal to move. It is calibrated in degrees, usually with every 30 degrees labeled. The nose of a small, fixed airplane on the instrument glass indicates the aircraft heading. (***Figure 7-36***) Vacuum driven direction indicators have many of the same basic gyroscopic issues as attitude indicators. Built-in compensation for precession varies and a caging device is usually found. Periodic manual realignment with the magnetic compass by the pilot is required during flight.

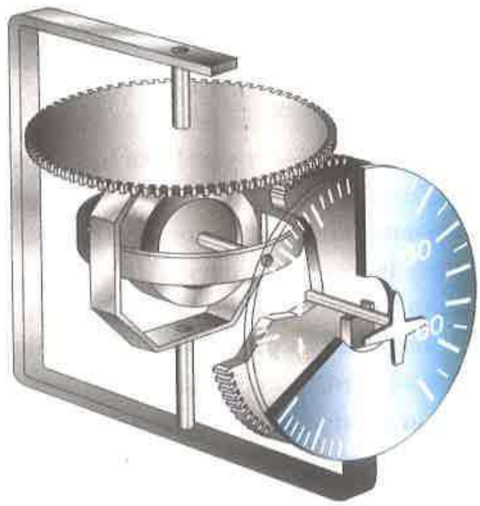


Figure 7-36. A typical vacuum-powered gyroscopic direction indicator.

HORIZONTAL SITUATION INDICATOR

The Horizontal Situation Indicator (HSI) (***Figure 7-37***) is an aircraft flight instrument normally mounted below the artificial horizon in place of a conventional heading indicator. It combines a heading indicator with a VHF Omnidirectional Range Instrument Landing System (VOR-ILS) display. The advantage of combining two instruments in one is to reduce the number of elements in the pilot's instrument scan. On an HSI, the aircraft is represented by a schematic figure in the center of the instrument. The VOR-ILS display is shown in relation to this figure. The heading indicator is usually connected to a remote compass and the HSI is frequently interconnected with an autopilot capable of following the heading select bug and of executing an ILS approach by following the localizer and glide slope. It is easier to follow the information of a HSI than a VOR indicator. When an HSI is tuned to a VOR station, left and right always mean left and right and TO/FROM is indicated by a simple triangular arrowhead pointing to the VOR. The most modern HSI displays are electronic and often integrated with electronic flight instrument systems into so-called "glass cockpit" systems.

TURN AND SLIP INDICATOR

The turn and slip indicator is also referred to as a turn-and-bank indicator, or needle and ball indicator. Regardless, it shows the correct execution of a turn while banking the aircraft and indicates movement about the vertical axis of the aircraft (yaw). Most turn-and-slip indicators are located below the airspeed indicator of the instrument panel basic T, just to the left of the direction indicator. The turn and slip indicator is two separate devices built into the same instrument housing: a turn indicator pointer and slip indicator ball. The turn pointer is operated by a gyro that can be

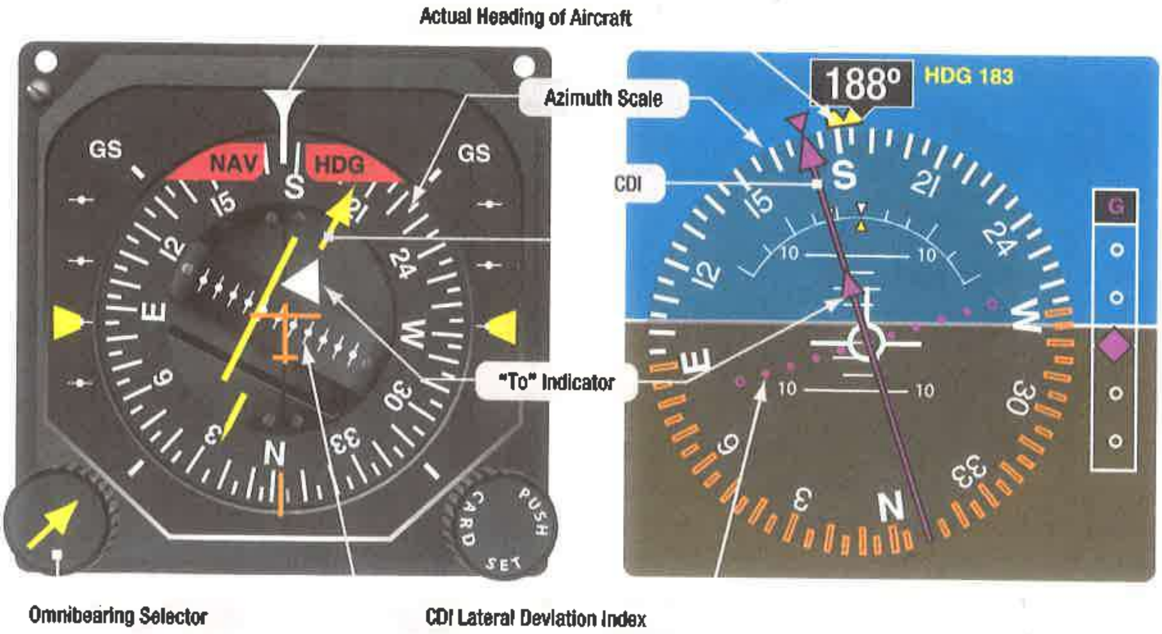


Figure 7-37. A mechanical HSI (left) and an electronic HSI (right).

driven by vacuum, air pressure, or electrically. The ball is a completely independent device. It is a round agate, or steel ball, in a glass tube filled with dampening fluid. It moves in response to gravity and centrifugal force experienced in a turn. Turn indicators indicate the rate at which the aircraft is turning. Three degrees of turn per second cause an aircraft to turn 360° in two minutes. This is considered a standard turn. This rate can be indicated with marks right and left of the pointer which normally rests in the vertical position. Sometimes, no marks are present, and the width of the pointer is used as the calibration device. In this case, one pointer width deflection is equal to the 3° per second standard 2 minute turn rate. Faster aircraft tend to turn more slowly and have graduations or labels that indicate 4 minute turns. In other words, a pointer width or alignment with a graduation mark indicates that the aircraft is turning a 1.5° per second and completes a 360° turn in 4 minutes. It is customary to placard the instrument face with words indicating whether it is a 2 or 4 minute turn indicator. (***Figure 7-38***)

The turn pointer indicates the rate at which an aircraft is turning about its vertical axis. It does so by using the precession of a gyro to tilt a pointer. The gyro spins in a vertical plane aligned with the longitudinal axis of the aircraft. When the aircraft rotates about its vertical axis during a turn, the force experienced by the spinning gyro is exerted about the vertical axis. Due to precession, the reaction of the gyro rotor is 90° further around the gyro in the direction of spin. This means the reaction to the force around the vertical axis is movement around the longitudinal axis of the aircraft. This causes the top of the rotor to tilt to the left or right. The pointer is attached with linkage that makes the pointer deflect in the opposite direction, which matches the direction of turn. So, the aircraft turn around the vertical axis is indicated around the longitudinal axis on the gauge. This is intuitive to the pilot since the pointer indicates in the same direction as the turn. (***Figure 7-39***)

The slip indicator (ball) part of the instrument is an inclinometer. The ball responds only to gravity during coordinated straight and level flight. Thus, it rests in the

Figure 7-38. A turn-and-slip indicator.

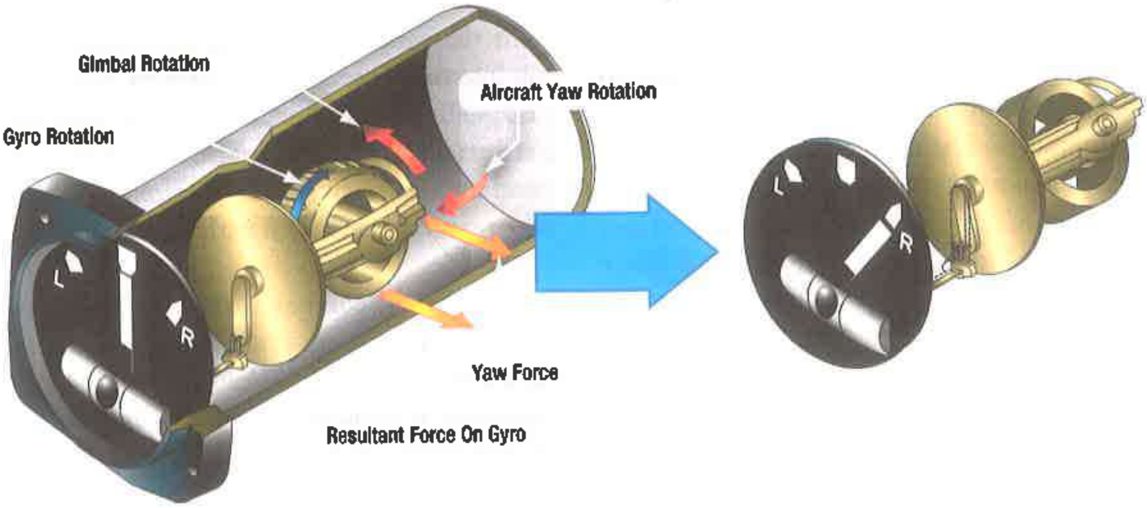


Figure 7-39. The turn-and-slip indicator's gyro reaction.

lowest part of the curved glass between the reference wires. When a turn is initiated and the aircraft is banked, both gravity and the centrifugal force of the turn act upon the ball. If the turn is coordinated, the ball remains in place. Should a skidding turn exist, the centrifugal force exceeds the force of gravity on the ball, and it moves in the direction of the outside of the turn.

During a slipping turn, there is more bank than needed, and gravity is greater than the centrifugal force acting on the ball. The ball moves in the curved glass toward the inside of the turn. Often power for the turn and slip indicator gyro is electrical if the attitude and direction indicators are vacuum powered. This allows limited operation off battery power should the vacuum system and the electric generator fail. The directional and attitude information from the turn-and-slip indicator, combined with information from the pitot-static instruments allow continued safe emergency operation of the aircraft. Electrically powered turn-and-slip indicators are usually DC powered. Vacuum powered turn-and-slip indicators are usually run on less vacuum (+/- 2" Hg) than fully gimbaled attitude and direction indicators. Regardless, proper vacuum must be maintained for accurate turn rate information to be displayed.

TURN COORDINATORS

Many aircraft make use of a turn coordinator. The rotor of the gyro in a turn coordinator is canted upwards 30°. As such, it responds not only to movement about the vertical axis, but also to roll movements about the longitudinal axis. This is useful because it is necessary to roll an aircraft to turn it about the vertical axis. Instrument indication of roll is the earliest possible warning of a departure from straight and level flight. Typically, the face of the turn coordinator has a small airplane symbol. The wing tips of the airplane provide the indication of level flight and the rate at which the aircraft is turning. (***Figure 7-40***)

COMPASSES

DIRECT READING, REMOTE READING

A myriad of techniques and instruments exists to aid the pilot in navigation. An indication of direction is part of this navigation. This section discusses some of the magnetic direction indicating instruments.

DIRECT READING

It is a requirement that all certified aircraft have some sort of magnetic direction indicator. The magnetic compass is a direction finding instrument that has been

Figure 7-40. A turn coordinator.

used for navigation for hundreds of years. It is a simple instrument that takes advantage of the Earth's magnetic field. ***Figure 7-41*** shows the Earth and the magnetic field that surrounds it. The magnetic north pole is close to the geographic north pole of the globe, but they are not the same. An ordinary permanent magnet that is free to do so, aligns itself with the direction of the magnetic pole. Upon this principle, permanent magnets are attached under a float that is mounted on a pivot, and so free to rotate in the horizontal plane. As such, the compass magnets align with the earth magnetic field.

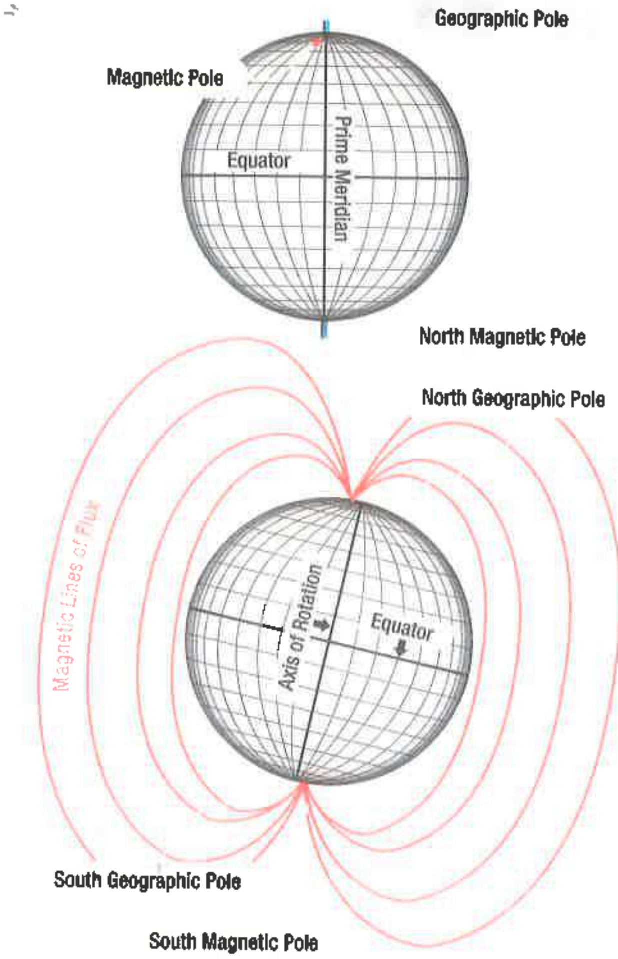


Figure 7-41. The earth and its magnetic field.

A numerical compass card, usually graduated in 5° increments, is constructed around the perimeter of the float. It serves as the instrument dial. The entire assembly is enclosed in a sealed case that is filled with a liquid similar to kerosene. This dampens vibration and oscillation of the moving float assembly and decreases friction. On the front of the case, a glass face allows the numerical compass card to be referenced against a vertical lubber line. The magnetic heading of the aircraft is read by noting the graduation on which the lubber line falls. Thus, direction in any of 360° can be read off the dial as the compass card assembly holds its alignment with magnetic north while the aircraft changes direction. The liquid that fills the compass case expands and contracts as altitude changes and temperature fluctuates. A bellows diaphragm expands and contracts to adjust the volume of the space inside the case and thus it remains full. (***Figure 7-42***)

Magnetic Deviation

There are accuracy issues associated with magnetic compasses. The main magnets of a compass align not only with the Earth's magnetic field, but also align with the composite field made up of local electromagnetic influences from metallic structures near the compass and the operation of the aircraft electrical system. This is

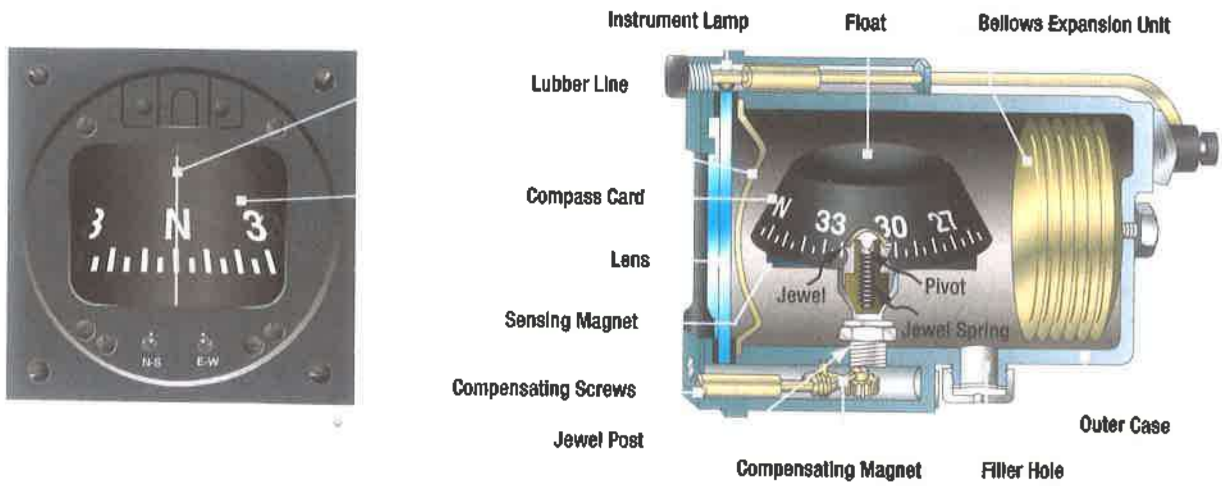


Figure 7-42. The parts of a typical magnetic compass.

called magnetic deviation. To correct this, compensating screws are turned which move small magnets in the compass case to correct for this deviation. The two setscrews are on the face of the instrument and are labeled N-S and E-W. They position the small magnets to counterbalance the local magnetic forces acting on the main magnets.

The process of adjusting for magnetic deviation is known as swinging the compass. It is described in the instrument maintenance pages near the end of this chapter. Magnetic deviation should never be more than 10 degrees. Using nonferrous mounting screws and shielding or twisting the wires running to the compass illuminating lamp are additional steps taken to keep deviation to a minimum.

Magnetic Variation

Another compass error is called magnetic variation. It is caused by the difference in location between the Earth's magnetic poles and the geographic poles. There are only a few places on the planet where a compass pointing to magnetic north is also pointing to geographic north. A line drawn through these locations is called the Agonic line. At all other points, there is some variation between that which a magnetic compass indicates is north and true geographic north. Isogonic lines drawn on aeronautical charts indicate points of equal variation. Depending on the location of the aircraft, airmen must add or subtract degrees from the magnetic indication to obtain true geographic location information. (***Figure 7-43***)

Dip Error

The Earth's magnetic field exits the poles vertically and arches around to extend past the equator horizontally or parallel to the Earth's surface. (***Figure 7-41***) Operating an aircraft near the magnetic poles causes what is known as dip error. The compass magnets pull downward toward the pole, rather than horizontally, as is the case near the equator. This downward motion causes inaccuracy in the indication. Although the compass float mechanism is weighted to compensate, the closer the aircraft is to the north or south magnetic poles, the more pronounced the errors.

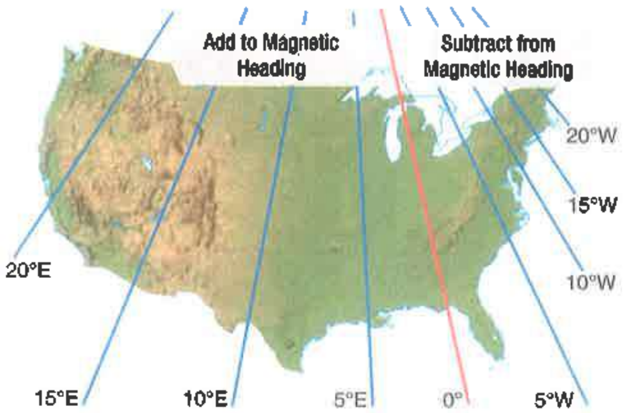


Figure 7-43. Aircraft located along the agonic line.

Dip errors manifest themselves in two ways. The first is called acceleration error. If an aircraft is flying on an east-west path and simply accelerates, the inertia of the float mechanism causes the compass to swing to the north. Rapid deceleration causes it to swing southward. Second, if flying toward the North Pole and a banked turn is made, the downward pull of the magnetic field initially pulls the card away from the direction of the turn. The opposite is true if flying south from the north pole and a banked turn is initiated. In this case, there is initially a pull of the compass indicator toward the direction of the turn. These kinds of movements are called turning errors.

Vertical Magnetic Compasses

Another peculiarity exists with the magnetic compass that is not dip error. If flying north or toward any indicated heading, turning the aircraft to the left causes a steady decrease in the heading numbers. But, before the turn is made, the numbers to the left on the compass card are increasing. The numbers to the right of the lubber line rotate behind it on a left turn, so, the compass card rotates opposite to the direction of the intended turn. This is because, from the pilot seat, you are looking at the back of the compass card. While not a major problem, it is more intuitive to see the 360° of direction oriented as they are on an aeronautical chart or a hand held compass.

The vertical magnetic compass is a variation of the magnetic compass that eliminates the reverse rotation of the compass card just described. By mounting the main indicating magnets on a shaft rather than a float, through a series of gears, a compass card can be made to turn about a horizontal axis. This allows the numbers for a heading to be oriented correctly on the indicating card. In other words, when turning right, increasing numbers are to the right; when turning left, decreasing numbers rotate from the left. (***Figure 7-44***)

Figure 7-44. A vertical magnetic direction indicator.

Many vertical magnetic compasses have also replaced the liquid filled housing with a dampening cup that uses eddy currents to dampen oscillations. Note that a vertical magnetic compass and a directional gyro look similar and are often in the lower center position of the instrument panel basic T. Both use the nose of an aircraft as the lubber line against which a rotating compass card is read. Vertical magnetic compasses are characterized by the absence of the hand adjustment knob found on Digital Gyros, which is used to align the gyro with a magnetic indication.

REMOTE READING

Magnetic deviation is compensated for by swinging the compass and adjusting compensating magnets in the instrument housing. By using a synchro remote indicating system, the magnetic compass's float assembly can act as the rotor of the synchro system. As the float mechanism rotates to align with magnetic north in the remotely located compass, a varied electric current can be produced in the transmitter. This alters the magnetic field produced by the coils of the indicator, and a magnetic indication relatively free from deviation is displayed. Many of these systems are of the Magnesyn type.

A Magnesyn transmitter is an elaborate and fully accurate method of direction indication that combines the use of a gyro, a magnetic compass, and a remote indicating system. A gyroscopic direction indicator is augmented by magnetic direction information from a remotely located compass. The type of compass used is called a flux valve or flux gate compass. It consists of a magnetically permeable circular segmented core frame or spider. The Earth's magnetic field flows through this iron core and varies its distribution through segments of the core as the flux valve is rotated via the movement of the aircraft. Pickup coil windings are located on each of the core spider legs that are positioned 120° apart. (***Figure 7-45***)

The distribution of the magnetic field flowing through the legs is unique for every directional orientation of the aircraft. A coil is placed in the center of the core and is energized by AC current. As the AC flow passes through zero while changing direction, the Earth's magnetic field can flow through the core. Then, it is blocked or gated as the magnetic field of the core current flow builds to its peak again. The cycle is repeated at the frequency of the AC supplied to the excitation coil. The result is repeated flow and non-flow of the flux across the pickup coils.

During each cycle, a unique voltage is induced in each of the pickup coils reflecting the orientation of the aircraft in the Earth's magnetic field. The electricity that flows

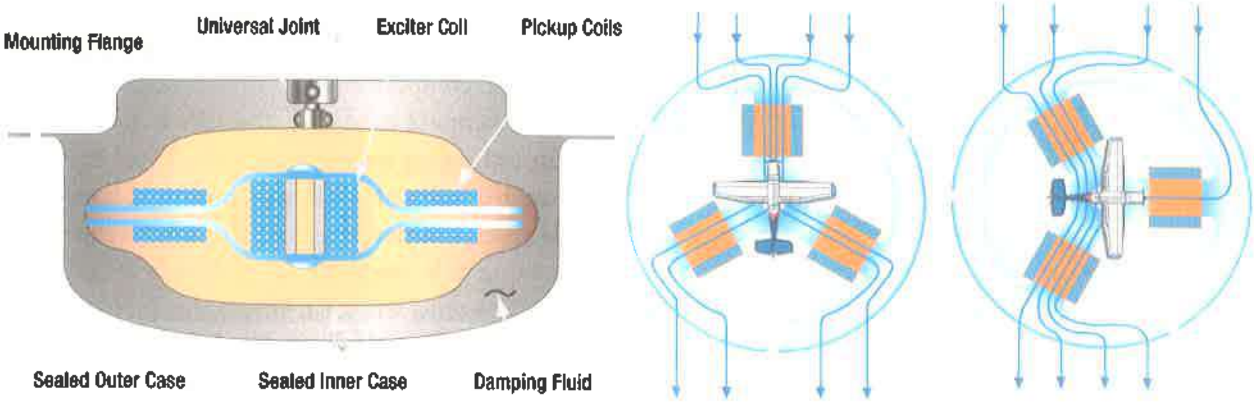


Figure 7-45. The simplified mechanism of a flyweight type mechanical tachometer.

from each of the pickup coils is transmitted out of the flux valve via wires into a second unit. It contains an Autosyn transmitter, directional gyro, an amplifier, and a triple wound stator like that found in the indicator of a synchro system. Unique voltage is induced in the center rotor of this stator which reflects the voltage received from the flux valve pickup coils sent through the stator coils. It is amplified and used to augment the position of the DG. The gyro is wired to be the rotor of an Autosyn synchro system, which transmits the position of the gyro into an indicator unit located in the cockpit. In the indicator, a vertical compass card is rotated against a small airplane lubber line like that in a vertical magnetic compass. (***Figure 7-46 and Figure 7-47***)

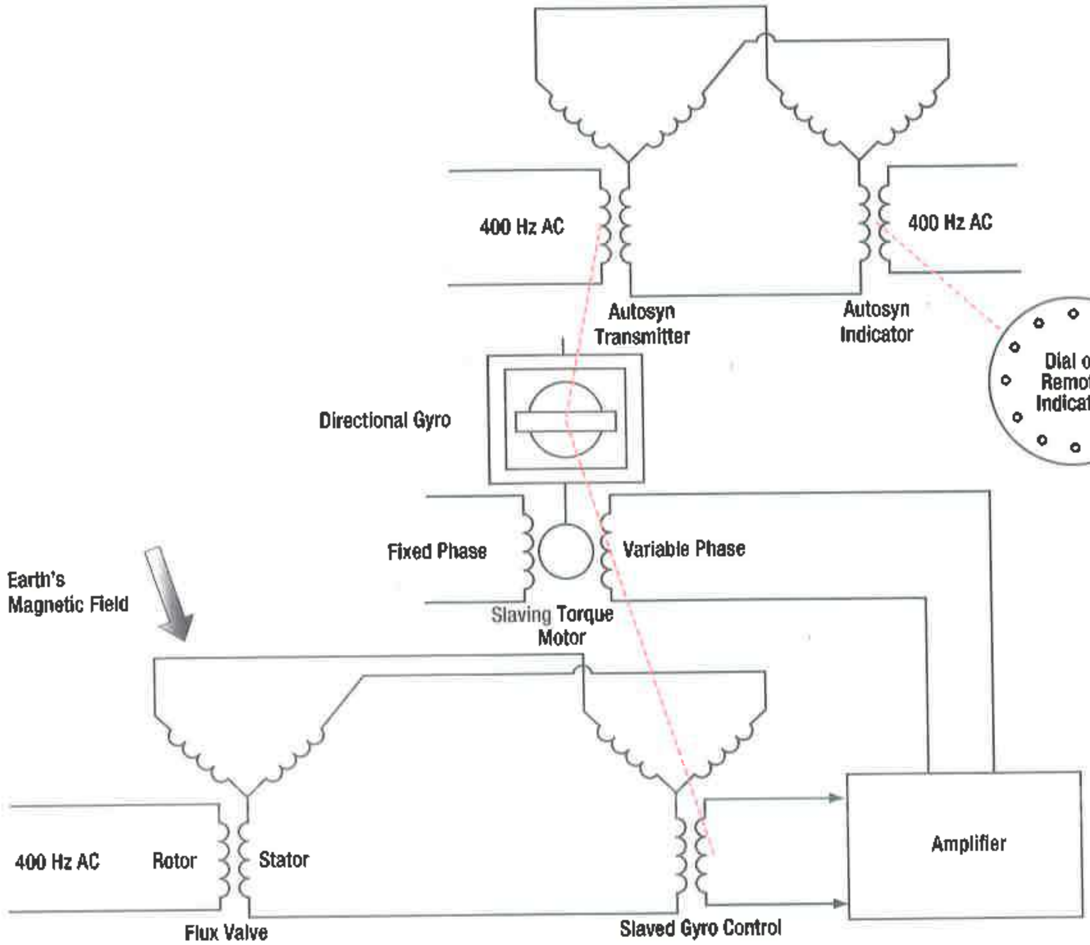


Figure 7-46. A simplified schematic of a flux gate.

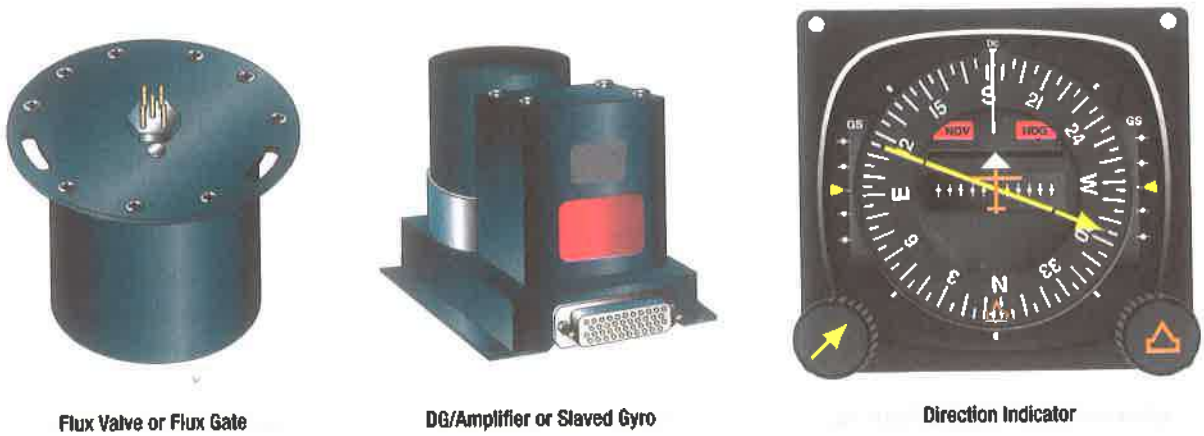


Figure 7-47. Solid state magnetometer units.

Further enhancements to direction finding systems of this type involve the integration of radio navigation aids. The Radio Magnetic Indicator is one such variation. (*Figure 7-48*) In addition to the rotating direction indicator of the slaved gyro compass, it contains two pointers. One indicates the bearing to a VHF VOR station and the other indicates the bearing to a nondirectional ADF beacon. It should be noted that integration of slaved gyro direction indicating system information into auto pilot systems is also possible.

Figure 7-48. A radio magnetic indicator (RMI).

Solid state magnetometers are used on many modern aircraft. They have no moving parts and are extremely accurate. Tiny layered structures react to magnetism on a molecular level resulting in variations in electron activity. These low power consuming devices can sense not only the direction to the Earth's magnetic poles, but also the angle of the flux field. They are free from oscillation that plagues a standard magnetic compass. They feature integrated processing algorithms and easily integrate with digital systems. (*Figure 7-49*)


Figure 7-49. Solid state magnetometer units.

VIBRATION INDICATING SYSTEMS (HUMS)

Health and Usage Monitoring Systems (HUMS) cover the topic of improving safety through data collection and analysis techniques, in particular for preventative maintenance. HUMS is an additional means of observing the functioning of systems to identify trends or anomalies and so allowing them to be assessed and corrected.

All on-board systems are focused on the collection, processing and interpretation of data generated by the various components within an aircraft's drive train, including engines, gearboxes, shafts, fans, and rotor systems. Collected data can be viewed at the aircraft, within the test cell or any other platform. Onboard vibration monitoring systems enhance safety through early detection of mechanical faults, and so reducing failures. It reduces maintenance hours, provides maximum flexibility and supports system growth with proven reliability.

Advanced engine diagnostics and automated engine performance calculations, such as a max power check and health indicator test, round out this feature. The system can connect to flight data recorders providing operators with crash survivable data storage. HUMS provides inflight drive train data acquisition, processing and diagnostics for complex aircraft. This reduces flight test data acquisition time. Data signaling potential problems on one aircraft can be used to comprehensively analyze an entire fleet. Better maintenance planning means less unplanned downtime, faster turnaround and increased mission readiness.

A helicopter has many vibrations. Changes to these vibrations due to an impending or partial failure may pass without being noticed. Many engines are therefore fitted with vibration indicators that continually monitor the vibration level. The indicator is usually a milli-ammeter that receives signals through an amplifier from the structure mounted transmitters.

Numerous sensors located on all primary mechanical and airframe parts relay information about the performance of these parts to the flight data retrieval unit. In flight, pilots can view and monitor this data on their Multi Function Displays and make informed decisions.

The vibration sensing element is usually an electromagnetic transducer that converts the rate of vibration into electrical signals and so causes the indicator pointer to move proportional to the vibration level. A warning lamp on the instrument panel is included to warn the pilot if an unacceptable level of vibration is approached, enabling the engine to be shut down and so reduce the risk of damage.

A more accurate method differentiates between the frequency ranges of each rotating assembly and so enables the source of vibration to be isolated. This is particularly important on rotorcraft due to their many gearboxes. A system of filters in the electrical circuit to the gauge makes it possible to compare the vibration obtained against a known frequency range and so locate the vibration source. A multiple selector switch enables the pilot to select a specific area to obtain a reading of the level of vibration.

HUMS performs and assists in the following functions:

- Built-in Tests
- Mechanical Diagnostics
- Usage Monitoring
- Exceedance/Event Monitoring
- Engine Vibration Monitoring
- Rotor Track and Balance

GLASS COCKPIT

To increase safety when operating complicated aircraft, computer systems have been incorporated. Flight instrumentation and engine and airframe monitoring are areas well suited to gain advantages from the use of computers. They contribute to reduce instrument panel clutter and focusing the pilot's attention only on matters of imminent importance.

"Glass cockpit" refers to the use of flat-panel display screens in cockpit instrumentation and the use of computer produced images to replace individual mechanical gauges. Moreover, computer systems monitor the processes and components of an operating aircraft beyond human ability while relieving the pilot of the stress from having to do so. In addition, the solid state nature of the components increase reliability. Microprocessors, data buses, and LCDs all save space and weight. Technician's frequently interface with Engine Indicating and Crew Alerting System (EICAS) and Electronic Centralized Aircraft Monitoring systems (ECAM) through control panels to gather operating and maintenance data. (***Figure 7-50***) Details on the operation and use of these glass cockpit maintenance aids are in the manufacturer's maintenance manuals.

OTHER AIRCRAFT SYSTEM INDICATIONS

REMOTE SENSING AND INDICATIONS

It is often impractical to utilize direct reading gauges for information needed to be read in the cockpit. Placing sensors at suitable locations on the airframe or engine and then transmitting the collected data electrically to displays in the cockpit is a widely used method of remote sensing and indicating on aircraft.

Remote sensing instrument systems operate with high reliability and accuracy. Small electric motors inside the instrument housings are used to position the pointers instead of mechanical linkages. They receive current from the output of the Air Data Computer or other computers. By varying the electric signal, the motors are turned to the precise location needed to reflect the correct indication. This type of remote sensing is both reliable and relatively simple.

Note that digital displays receive their input from Digital Air Data Computers via a digital data bus and do not use electric motors. The data packages are transmitted via the bus and contain instructions on how to illuminate the display screen.

Synchro Type Remote Indicating Instruments

A synchro system is an electric system used for transmitting information from one point to another. The word 'synchro' means synchronous, and refers to a two unit electrical system capable of measuring, transmitting, and indicating a certain parameter of the aircraft. Most position indicating instruments, such as flap position, are designed around a synchro system, Fluid pressure indicators, landing gear, autopilots, radar, and many other remote applications also often operate with a synchro system. The most common types of synchro systems are the Autosyn, Selsyn, and Magnesyn synchro systems.

Synchro systems are similar in construction and all operate by exploiting the consistent relationship between electricity and magnetism. The fact that

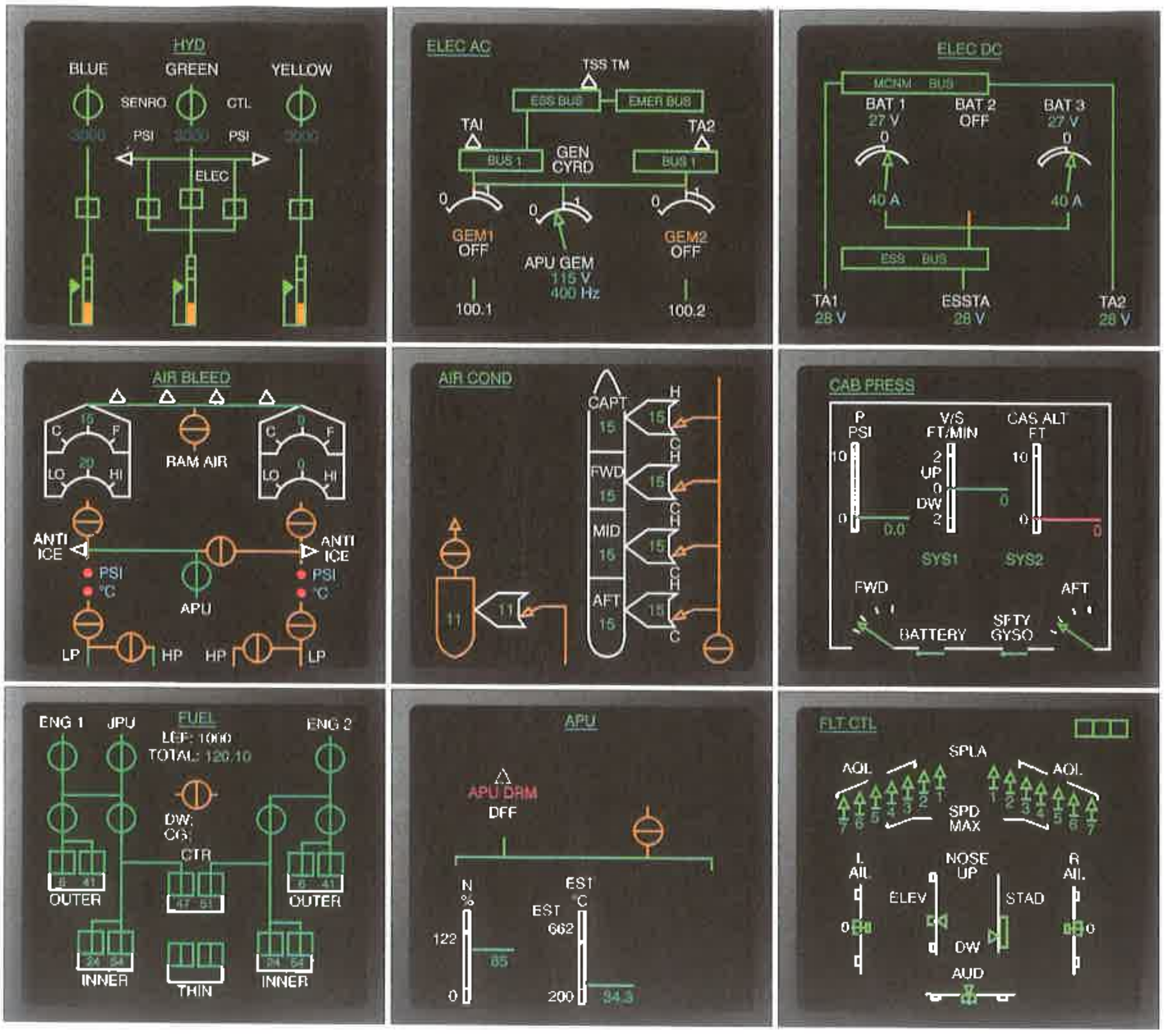


Figure 7-50. EICAS and ECAM cockpit displays.

electricity can be used to create magnetic fields that have a definite direction and that magnetic fields can interact with other electromagnetic fields, is the basis of their operation.

DC Selsyn Systems

On aircraft with DC electrical systems, DC Selsyn systems are widely used. A Selsyn system consists of a transmitter, an indicator, and connecting wires. The transmitter consists of a circular resistance winding and a rotatable contact arm. The rotatable arm turns on a shaft in the center of the resistance winding. The two ends of the arm are brushes and always touch the winding on opposite sides. (***Figure 7-51***)

On position indicating systems, the shaft to which the contact arm is fastened protrudes through the end of transmitter housing and is attached to the unit whose position is to be transmitted such as flaps or landing gear.

A transmitter is often connected to the moving unit through a mechanical linkage. As the unit moves it causes the transmitter shaft to turn. As this occurs voltage is applied through the brushes to any two points around the circumference of the resistance winding. The rotor shaft of a DC Selsyn system operates the same way, but instead of protruding outside of the housing may be located inside the housing instead.

Referring to ***Figure 7-51***, note that the resistance winding of the transmitter is tapped off in three fixed places, usually 120° apart. These taps distribute current through the toroidal windings of the indicator motor. When current flows through these windings, a magnetic field is created. Like all magnetic fields, a definite north and south direction to the field exists.

As the transmitter's rotor shaft is turned, the voltage supplied to the contact arm changes. Because it contacts the transmitter resistance winding in different positions, the resistance between the supply arm and the various tap-offs change. This causes the voltage flowing through the tap-offs to change as the resistance of sections of the winding become longer or shorter. The result is that varied current is sent via the tap-offs to the three windings in the indicator motor.

The resultant magnetic field and its direction created by current flowing through the indicator coils also changes as each receives varied current from the tap-offs. Thus, the direction of the field across the indicating element corresponds to the moving arm in the transmitter.

A permanent magnet is attached to the centered rotor shaft in the indicator. The magnet, and so the pointer aligns itself with the direction of the magnetic field.

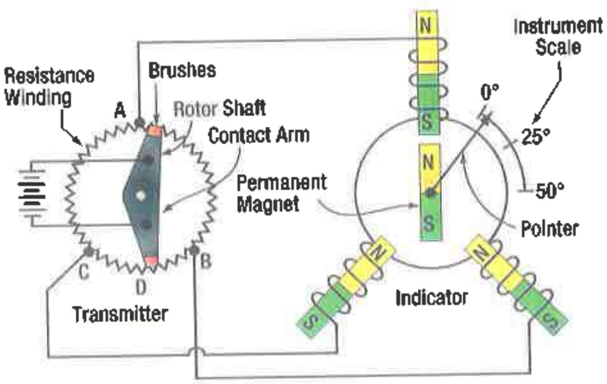


Figure 7-51. A schematic of a DC Selsyn Synchro remote indicating system.

Whenever the magnetic field changes direction, the permanent magnet and pointer realign with the new position of the field. Thus, the position of the aircraft device is indicated.

Landing gears contain mechanical devices that lock the gear up, called an up-lock, or down, called a down-lock. When the DC Selsyn system is used to indicate the position of the landing gear, the indicator can also show that the up-lock or down-lock is engaged. This is done by again varying the current flowing through the indicator coils. Switches on the mechanical locking devices close when the locks engage.

In a landing gear system, current from the Selsyn system flows through the switch to an additional circuit which adds a resistor to one of the transmitter windings created by the rotor arm and a tap-off. This changes the total resistance of that section resulting in a change of current flowing through one of the indicator motor coils. This in turn, changes the magnetic field around that coil. Therefore, the combined magnetic field created by all three motor coils is affected, causing a shift in the direction of the indicator field. The permanent magnet and pointer align with the new direction and shift to the lock/unlock position on the indicator. ***Figure 7-52*** shows a simplified diagram of a lock switch in a three wire Selsyn system and an indicator dial.

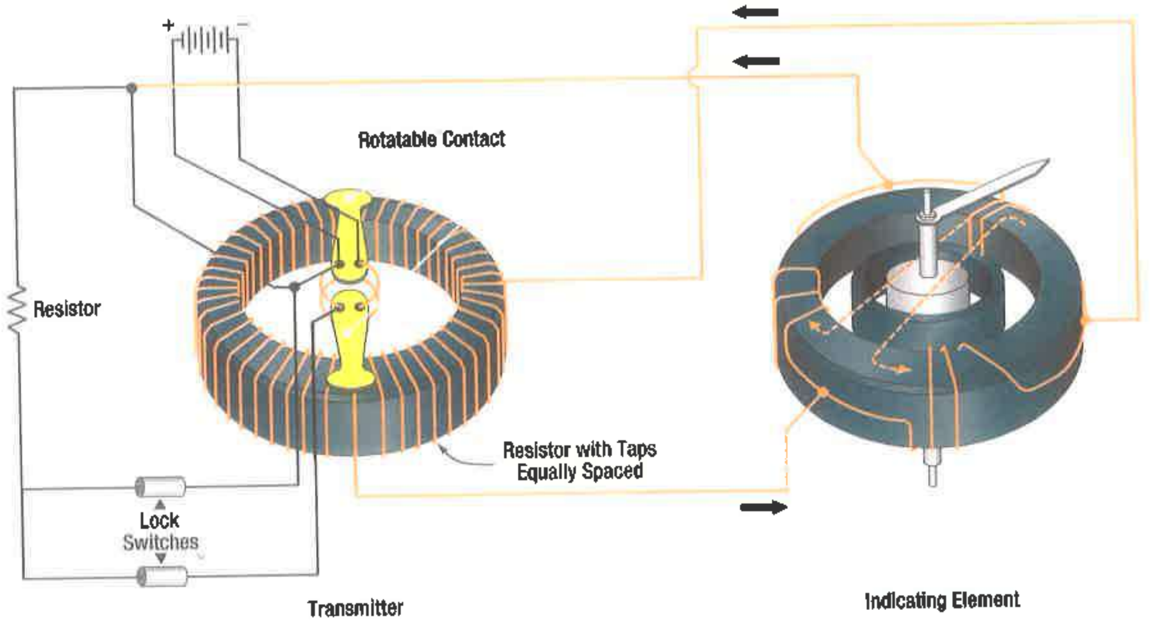


Figure 7-52. A lock switch circuit.

AC Synchro Systems

Aircraft with AC power systems make use of Autosyn or Magnesyn remote indicating systems. Both operate in a similar way to DC Selsyn, except that AC power is used and so make use of electric induction, rather than resistance current flows as with DC systems. Magnesyn systems use permanent magnet rotors such as those found in the DC Selsyn system. Usually, the transmitter magnet is larger than the indicator magnet, but the electromagnetic response of the indicator rotor magnet and pointer remains the same. It aligns with the magnetic field set up by the coils, adopting the same angle of deflection as the transmitter rotor. (***Figure 7-53***)

Autosyn systems are further distinguished by the fact that the transmitter and indicator rotors are electromagnets rather than permanent magnets. Nonetheless, like a permanent magnet, an electromagnet aligns with the direction of the magnetic field created by current flowing through the stator coils in the indicator. Thus, the indicator pointer's position mirrors the transmitter rotor position. (***Figure 7-54***)

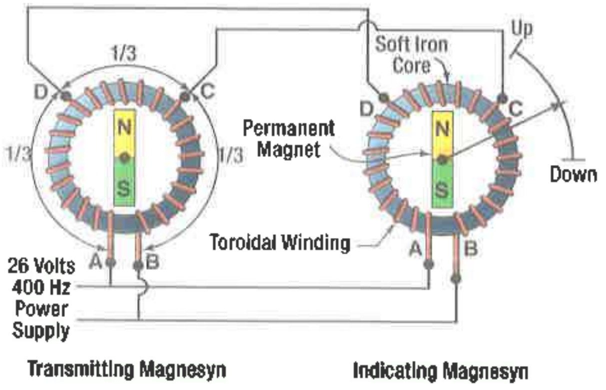


Figure 7-53. A magnasysn synchro remote-indicating system.

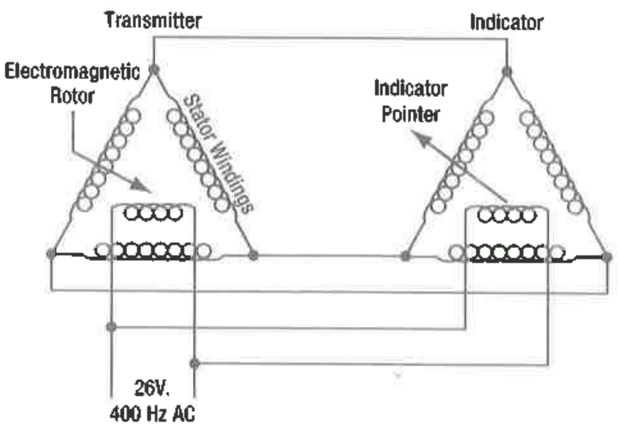


Figure 7-54. An autosyn remote-indicating system.

AC synchro systems are wired differently than DC systems. The varying current flows through the transmitter, and indicator stator coils are induced as the AC cycles through zero and the rotor field flux can flow.

The important characteristics of all synchro systems are maintained by both Autosyn and Magnesyn systems. The position of the transmitter rotor is mirrored by the rotor in the indicator. These systems are used in many of the same applications as the DC systems and more.

REMOTE INDICATING

Fuel And Oil Pressure Gauges

Fuel and oil pressure indications can be conveniently obtained using synchro systems. As stated previously, running fuel and oil lines into the cabin to direct-reading gauges is not desirable due to increased risk of fire in the cabin and the additional weight of the lines. By locating the transmitter of a synchro system remotely, fluid pressure can be directed into it without a long tubing run. Inside the transmitter, the motion of a pressure bellows can be geared to the transmitter rotor in such a way as to make the rotor turn. (***Figure 7-55***)

As in all synchros, the transmitter rotor turns proportional to the pressure sensed, which varies the voltages set up in the resistor windings of the synchro stator. These voltages are transmitted to the indicator coils that alter the magnetic field positioning the pointer. Often on twin-engine aircraft, synchro mechanisms for each engine can be used to drive separate pointers on the same indicator. By placing the coils one behind the other, the pointer shaft from the rear indicator motor can be sent through the hollow shaft of the forward indicator motor. Thus, each pointer responds with the magnet alignment in its own motor magnetic field while sharing the same gauge housing. Labeling the pointers engine 1 or 2 removes any doubt about which indicator pointer is being observed.

A similar principle is employed in an indicator that has side-by-side indications for different parameters, such as oil pressure and fuel pressure in the same indicator housing. Each parameter has its own synchro motor for positioning its pointer. Aircraft with digital instrumentation make use of pressure sensitive solid-state sensors that output signals for collection and processing by dedicated engine and airframe computers. Others

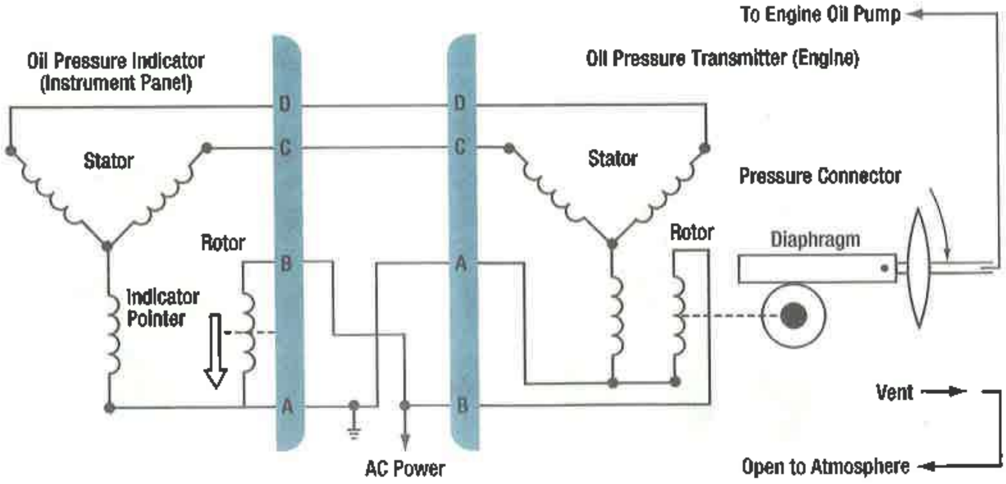


Figure 7-55. Remote pressure sensing indicators.

may retain their analog sensors, but may forward this information through an analog to digital converter unit from which the appropriate computer can obtain digital information to illuminate a digital display.

MECHANICAL MOVEMENT INDICATORS

Tachometers

The tachometer (tach) indicates the rotational speed of an engine in Revolutions Per Minute (RPM). It can be a direct or remote indicating. On reciprocating engines, the tach is used to monitor engine power and to ensure the engine is operated within certified limits. On gas turbine engines it monitors the speed(s) of the compressor section(s) of the engine. Turbine engine tachometers are calibrated in percentage of RPM with 100% corresponding to optimum turbine speed. This allows similar operating procedures despite the actual varied RPM of each engine. ***(Figure 7-56)***

Along with the speed of rotation of the engine, it is also necessary on a rotorcraft to know the rotor speed of rotation. During autorotation, the only needed information for control is the rotation of the rotor. If the rotor spins too fast, hinges can break, and the consequence could be catastrophic like losing a blade creating an obvious crash. On the other hand, if the rotational speed decreases too much, the lift decreases at the same time. If the pilot does not realize this in time, the consequences could be irreversible. Rotor tachometers are calibrated in RPM for a linked turbine engine and in percentage of RPM for free turbine engines. ***(Figure 7-57)***

Figure 7-56. Example tachometers for a reciprocating engine.

Figure 7-57. A tachometer for the main rotor.

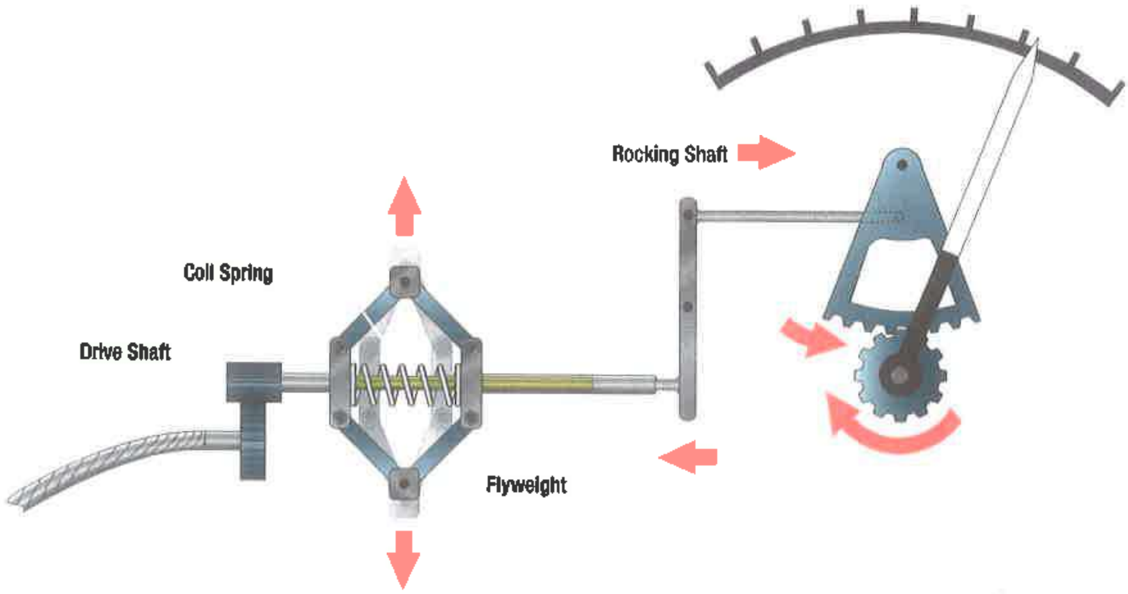


Figure 7-58. The simplified mechanism of a flyweight type mechanical tachometer.

It should also be noted that many reciprocating engine tachometers also have built in numeric drums that are geared to the rotational mechanism inside. They consist of an indicator connected to the engine by a flexible drive shaft. The drive shaft is geared to the engine so that when the engine turns, so does the shaft. The indicator contains a flyweight assembly coupled to a gear mechanism that drives a pointer. As the drive shaft rotates, centrifugal force acts on the flyweights and moves them to an angular position. This angular position varies with the RPM of the engine or gearbox. The amount of movement of the flyweights is transmitted through the gear mechanism to the pointer. (*Figure 7-58*)

A common variation of a mechanical tachometer uses a magnetic drag cup to move the pointer in the indicator. As the drive shaft turns, it rotates a permanent magnet in a close tolerance aluminum cup. A shaft attached to the indicating pointer is attached to the exterior center of the cup. As the magnet is rotated by the engine flex drive cable, its magnetic field cuts through the conductor surrounding it creating eddy currents in the aluminum cup. This current flow creates a magnetic field, which interacts with the rotating magnet's field. The result is that the cup tends to rotate, and with it the indicating pointer. A calibrated restraining spring limits the cup's rotation to the arc of the pointer across the instrument face. (*Figure 7-59*)

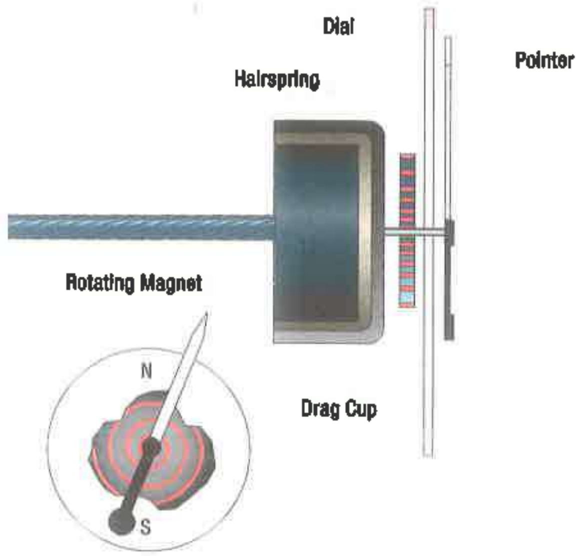


Figure 7-59. A simplified magnetic drag cup tachometer indicating device.

Electric Tachometers

It is not practical to use a mechanical linkage between the engine and RPM indicators on rotorcraft due to the distance between the engine and the instrument panel. Greater accuracy with lower maintenance is achieved using electric tachometers. As a wide variety of these systems can be employed, the manufacturer's instructions should be consulted for each specific tachometer system. A popular electric tachometer makes use of a small AC generator mounted on a reciprocating engine gear case or on the accessory drive section of a turbine engine. As the engine turns, so does the generator. The frequency output of the generator is directly proportional to the speed of the engine. It is connected via wires to a

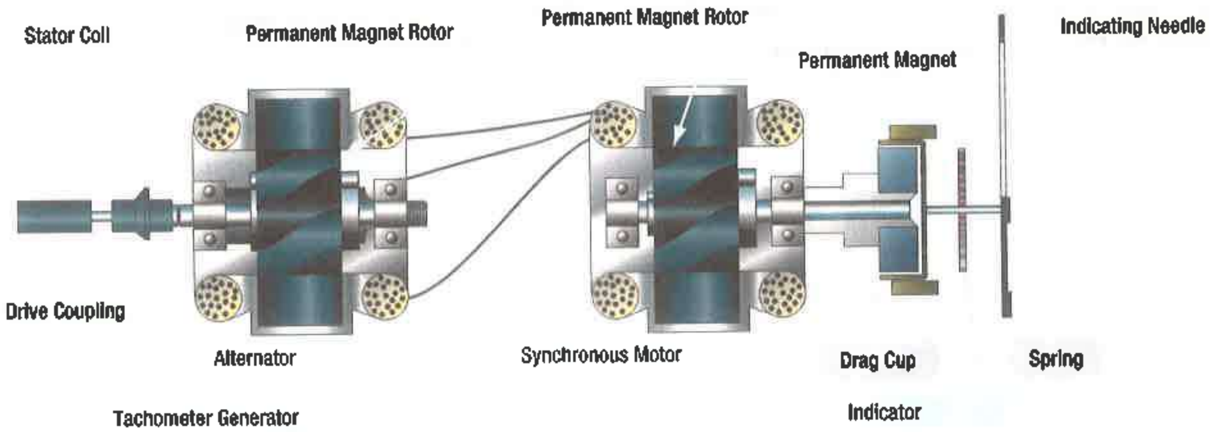


Figure 7-60. An electric tachometer system.

synchronous motor in the indicator that mirrors this output. A drag cup, or disk link, is used to drive the indicator as in mechanical tachometers. (***Figure 7-60***)

Two different types of generator units, distinguished by their type of mounting system, are shown in ***Figure 7-61.*** The dual tachometer consists of two tachometer indicator units housed in a single case. The indicator pointers show simultaneously, on one or two scales, the RPM of two engines. A dual tachometer on a helicopter often shows the RPM of the engine and of the main rotor. A comparison of the voltages produced by the two generators of this type of helicopter indicator gives information concerning clutch slippage. A third indication showing this slippage is sometimes included in a helicopter's tachometer. (***Figure 7-62***)

Some turbine engines use tachometer probes for RPM indication, rather than a tachometer generator system. This provides an advantage in that there are no moving parts. They are sealed units that are mounted on a flange and protrude into the compressor section of the engine. A magnetic field is set up inside the probe that extends through pole pieces and out the end of the probe. A rotating gear wheel, which moves at the same speed as the engine compressor shaft, alters the magnetic field flux density as it moves past the pole pieces at close proximity. This generates voltage signals in coils inside the probe. The amplitude of the Electro Magnetic Field signals vary directly with the speed of the engine. The tachometer probe outputs signals to be processed in a remotely located module. They must then be amplified to drive a servo motor indicator in the cockpit. They may also be used as input for an automatic power control system or a flight data acquisition system. (***Figure 7-63***)

Collective Pitch Transmitter

The collective pitch transmitter is connected to the collective pitch indicator to transmit the collective pitch value of the main rotor. The transmitter is connected under the Main Rotor Head's (MRH) spherical joint which moves only in the collective's displacement of the flight controls. (***Figure 7-64***) As the pilot raises and lowers the stick, the ball joint moves the position of the transmitter. The retraction/extension of the transmitter modifies the position of a magnet which together

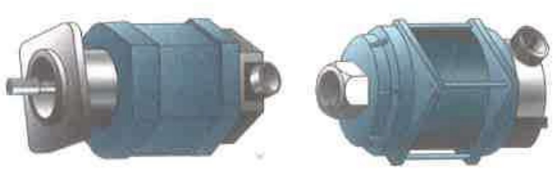


Figure 7-61. Different types of tach generators.

Figure 7-62. A helicopter tachometer.

with its magnetic field, modifies an electrical signal as analyzed by the step indicator to convert the signal into an indication.

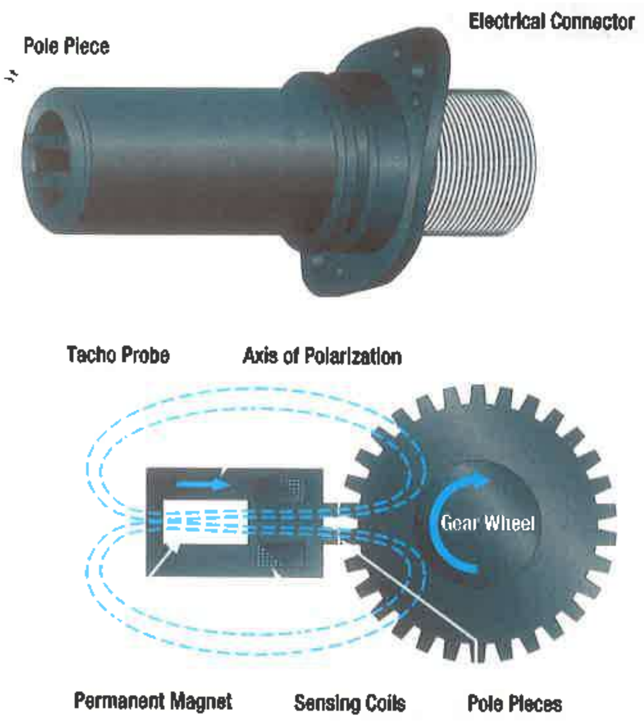


Figure 7-63. A tacho probe.

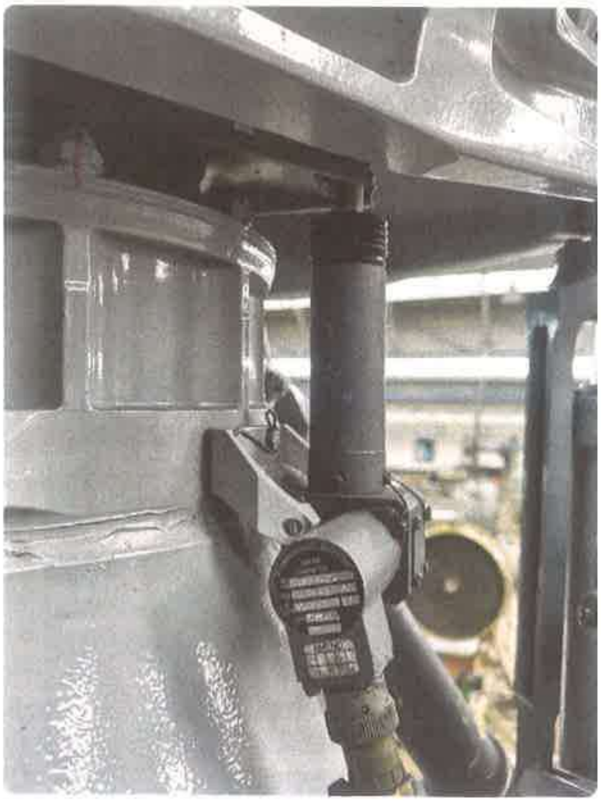


Figure 7-64. A collective pitch transmitter.

TEMPERATURE MEASURING INSTRUMENTS

For an aircraft to operate properly, the temperature of numerous items must be known. Engine oil, carburetor mixture, inlet air, free air, engine cylinder heads, heater ducts, and various turbine engine temperatures all require monitoring. Many other temperatures must also be known. Different types of thermometers are used to collect and display temperature information.

Non-Electric Temperature Indicators

The physical characteristics of most materials change with changes in temperature. The changes are consistent, such as the expansion or contraction of solids, liquids, and gases. While not used in aviation, most everyone is familiar with the liquid mercury thermometer. As the temperature of the mercury increases, it expands up a narrow passage that has a graduated scale upon it to read the temperature associated with that expansion.

A bimetallic thermometer is especially useful in aviation. The temperature sensing elements of a bimetallic thermometer are made of two dissimilar metal strips bonded together. Each metal expands and contracts at a different rate when temperature changes. One end of the bimetallic strip is fixed, and the other end is coiled. A pointer is attached to the coiled end which is set in the instrument housing. When the bimetallic strip is heated, the two metals expand. Since their expansion rates differ and they are attached to each other, the effect is that the coiled end tries to uncoil as that metal expands faster than the other. This moves the pointer across the dial face of the instrument. When the temperature drops, the metals contract at different rates, which tightens the coil to move the pointer in the opposite direction.

Direct reading bimetallic temperature gauges are often used in light aircraft to measure free air or outside air temperature. In this application, a collecting probe protrudes through the windshield of the aircraft to be exposed to the outside air. The coiled end of the bimetallic strip in the instrument head is just inside the windshield where the pilot can read it.
(***Figure 7-65 and Figure 7-66***)

A Bourdon tube gauge is also a simple direct reading nonelectric temperature gauge. By calibrating the dial face of a Bourdon tube gauge with a temperature scale, the temperature can be indicated. The basis for operation

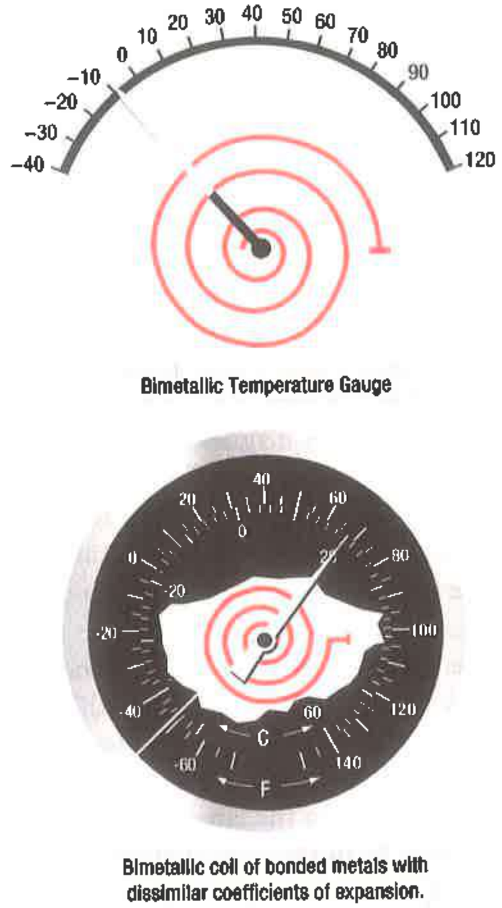


Figure 7-65. A bimetallic temperature gauge.

Figure 7-66. A bimetallic outside air temperature gauge.

is the consistent expansion of the vapor produced by a volatile liquid in an enclosed area. This vapor pressure changes directly with temperature. By filling a sensing bulb with a volatile liquid and connecting it to a Bourdon tube, the tube causes an indication of the rising or falling vapor pressure due. Calibration of the dial face is in degrees Fahrenheit or Celsius, rather than pounds Per Square Inch (psi), and so providing a temperature reading. In this type of gauge, the sensing bulb is placed in the area needing to have its temperature measured. A long capillary tube connects the bulb to the Bourdon tube in the instrument housing. The narrow diameter of the capillaries ensures that the volatile liquid is lightweight and stays in the sensor bulb. Oil temperature is sometimes measured this way.

Electric Temperature Indicators

Electrical indicators are common in aviation. The following measuring and indication systems can be found on many types of aircraft. Certain temperature ranges are more suitably measured by one or another type of system. The principal parts of the electrical resistance thermometer are the indicating instrument, the temperature sensitive element (or bulb), and the connecting wires and connectors. Electrical resistance thermometers are used widely to measure carburetor air, oil, free air, and more in low and medium temperatures from -70°C to 150°C.

For most metals, electrical resistance changes as the temperature of the metal changes. Typically, resistance of a metal increases as its temperature rises. Various alloys have a high temperature resistance coefficient, meaning their resistance varies significantly. This makes them suitable for temperature sensing. The metal resistor is subjected to the fluid or area in which temperature needs to be measured. It is connected by wires to a resistance measuring device inside the cockpit indicator. The dial is calibrated in degrees Fahrenheit or Celsius, rather than ohms. As the temperature changes, the resistance of the metal changes, and the indicator shows to what extent.

A typical electrical resistance thermometer looks like any other temperature gauge. Most indicators are self compensating for changes in cockpit temperature. The heat sensitive resistor is manufactured to have definite resistance for each temperature value within its working range. The temperature sensitive resistor is a length of a nickel/manganese wire or other suitable alloy in an insulating material. It is protected by a closed end metal tube attached to a threaded plug with a hexagonal head. (***Figure 7-67***) The two ends of the winding are brazed or welded, to an electrical receptacle designed to receive the prongs of the connector plug.

The indicator contains a resistance measuring instrument. Sometimes it uses a modified form of a Wheatstone bridge circuit. The Wheatstone bridge meter operates by balancing one unknown resistor against other known resistances. Three equal values of resistance are connected into a diamond shaped circuit. The resistor with an unknown value is also part of the circuit. The unknown resistance represents the resistance of the temperature bulb of the thermometer system. A galvanometer is attached across the circuit at points X and Y.

When the temperature causes the resistance of the bulb to equal that of the other resistances, no potential difference exists between points X and Y. Therefore, no current flows in the galvanometer leg of the circuit. If the temperature of the bulb changes, its resistance then changes and the bridge becomes unbalanced, causing current to flow through the galvanometer in one direction or the other. The galvanometer pointer is the temperature gauge pointer moving against the dial face calibrated in degrees. Many indicators are provided with a zero adjustment screw on the face of the instrument. This adjusts the zeroing spring tension of the pointer when the bridge is balanced so that no current flows through the meter.

Ratiometer Electrical Resistance Thermometers

Because a Wheatstone bridge indicator is subject to errors from line voltage fluctuation, another way of electrically indicating temperature is with a ratiometer. The ratiometer is more stable and can deliver higher accuracy. As its name suggests, the ratiometer measures a ratio of current flows. The resistance bulb sensing portion of the ratiometer is essentially the same as a Wheatstone bridge. The circuit contains a variable resistance and a fixed resistance to provide an indication. It contains two branches for current flow. Each has a coil mounted on either side of the pointer assembly that is mounted within the magnetic field of a large permanent magnet. Varying current flow through the coils causes different magnetic fields to form, which in turn react with the larger magnetic field of the permanent magnet. This interaction rotates the pointer against the dial face that is calibrated in degrees giving a temperature indication. (***Figure 7-68***)

The magnetic pole ends of the permanent magnet are closer at the top than they are at the bottom. This causes the magnetic field lines of flux between the poles to be more concentrated at the top. As the two coils produce their magnetic fields, the stronger field pivots downward into the weaker less concentrated part of the permanent field, while the weaker magnetic field shifts upward toward the more concentrated flux field of the large magnet. This provides a balancing effect that changes as the coil field strengths vary with temperature and so the resultant current flowing through the coils.

For example, if the resistance of the temperature bulb is equal to the value of the fixed resistance (R), equal values of current flow through the coils. The torques, caused by the magnetic field each coil creates, are the same and cancel any movement in the larger field. The indicator pointer will then be in the vertical position. If the bulb

Figure 7-67. An electric resistance thermometer sensing bulb.

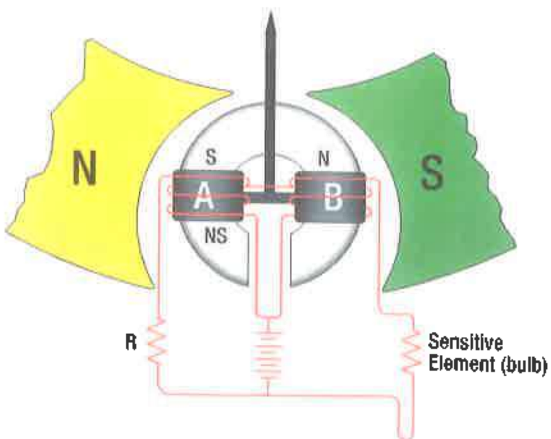


Figure 7-68. A ratiometer temperature measuring indicator.

temperature increases, its resistance also increases. This causes the current flow through the coil A circuit branch to increase and creates a stronger magnetic field at coil A than at coil B.

Consequently, the torque on coil A increases, and is pulled downward into the weaker part of the large magnetic field. At the same time, less current flows through the sensor bulb resistor and through coil B, causing coil B to form a weaker magnetic field that is pulled upward into the stronger flux area of the permanent magnet field. The pointer stops rotating when the fields reach a new balance point that is causally related to the resistance in the sensing bulb.

Ratiometer temperature measuring systems are used to measure engine oil, outside air, carburetor air, and other temperatures in many types of aircraft. They are especially useful to measure temperatures where accuracy is important, or large variations of supply voltages are encountered.

Thermocouple Temperature Indicators

A thermocouple is a circuit or connection of two unlike metals. The metals are touching at two separate junctions. If one of the junctions is heated to a higher temperature than the other, an electromotive force is produced in the circuit. This voltage is directly proportional to the temperature. So, by measuring the amount of electromotive force, temperature can be determined. A voltmeter is placed across the colder of the two thermocouple junctions. The hotter the high temperature junction (hot junction) becomes, the greater the electromotive force produced, and the higher the temperature indication on the meter. (***Figure 7-69***)

Thermocouples are used to measure high temperatures. Two common applications are Cylinder Head Temperature (CHT) in reciprocating engines and Exhaust Gas Temperature (EGT) in turbine engines. Thermocouple leads are made from a variety of metals, depending on the maximum temperature to which they are exposed. Iron and constantan, or copper and constantan, are common for CHT measurement. Chromel and alumel are used for turbine EGT thermocouples. The amount of voltage produced by the dissimilar metals when heated is measured in millivolts. Therefore, thermocouple leads are designed to provide a specific amount of resistance in the thermocouple circuit.

The material, length, or cross-sectional size of the thermocouple leads cannot be altered without compensation for the change in total resistance that would result. Each lead that makes a connection back to the voltmeter must be made of the same metal as the part of the thermocouple to which it is connected.

The hot junction of a thermocouple also varies in shape depending on its application. Two common types are the gasket and the bayonet. In the gasket type, two rings of the dissimilar metals are pressed together to form a gasket that can be installed under a spark plug or cylinder hold down nut. In the bayonet type, the metals come together inside a perforated protective sheath.

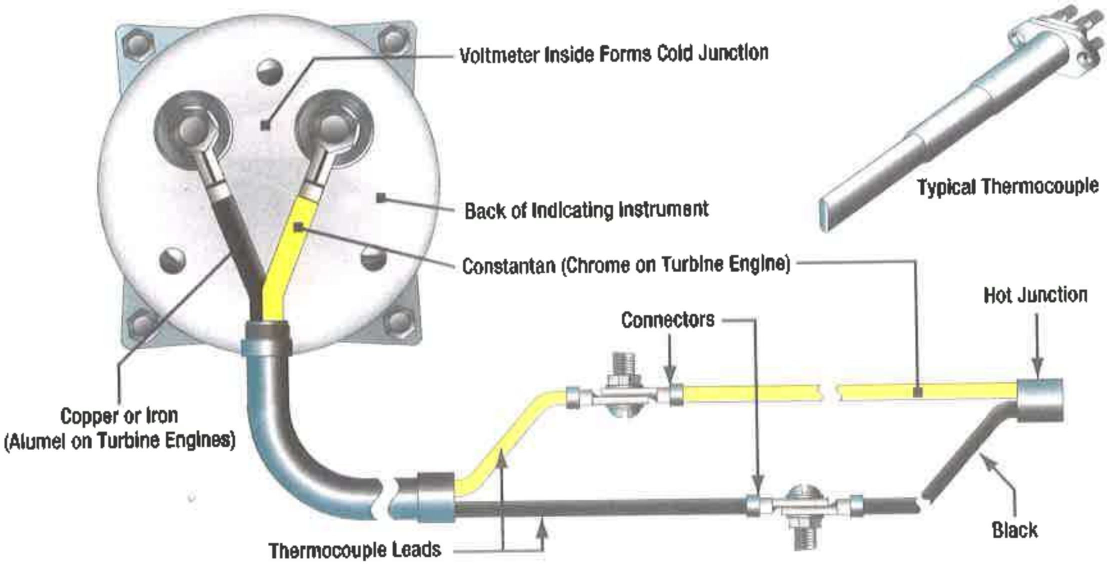


Figure 7-69. Thermocouples combine two unlike metals.

Bayonet thermocouples fit into a hole, or in a cylinder head. On turbine engines, they are found mounted on the turbine inlet or outlet case and extend through the case into the gas stream.

Note that for a CHT indication, the cylinder chosen for the thermocouple installation is the one that runs the hottest under most operating conditions. The location of this cylinder varies with different engines. (*Figure 7-70*)

The cold junction of the thermocouple circuit is inside the instrument case. Since the electromotive force in the circuit varies with the difference in temperature between the junctions, it is necessary to compensate the indicator mechanism for changes in cockpit temperature, which affect the cold junction. This is accomplished by using a bimetallic spring connected to the indicator mechanism. This works the same as the bimetallic thermometer described previously. When the leads are disconnected from the indicator, the temperature of the cockpit area around the instrument panel can be read on the indicator dial. (*Figure 7-71*) Numeric indicators for CHT are also common.

Turbine Gas Temperature Indicating Systems

EGT is a critical variable of turbine engine operation. The EGT indicating system provides a visual indication in the cockpit of the exhaust gases as they leave the turbine section. In certain turbine engines, EGT is measured at the entrance to the turbine section. This is then referred to as a Turbine Inlet Temperature (TIT) indicating system.

Several thermocouples are used to measure EGT or TIT. They are spaced at intervals around the perimeter of the turbine casing or exhaust duct. The tiny thermocouple voltages are amplified and used to energize a servomotor that drives the indicator pointer. Gearing a digital drum indication off the pointer motion is common. (*Figure 7-72*)

The EGT indicator shown is a hermetically sealed unit. The instrument scale ranges from 0°C to 1 200°C, with a vernier dial in the upper right hand corner and a power off warning flag on the lower portion of the dial.

Numerous thermocouples can be used with the average voltage representing the TIT. Dual thermocouples exist containing two electrically independent junctions within a single probe. One set of thermocouples is paralleled to transmit signals to the indicator. The other set provides temperature signals to engine monitoring and control systems. Each circuit is electrically independent, providing dual reliability.

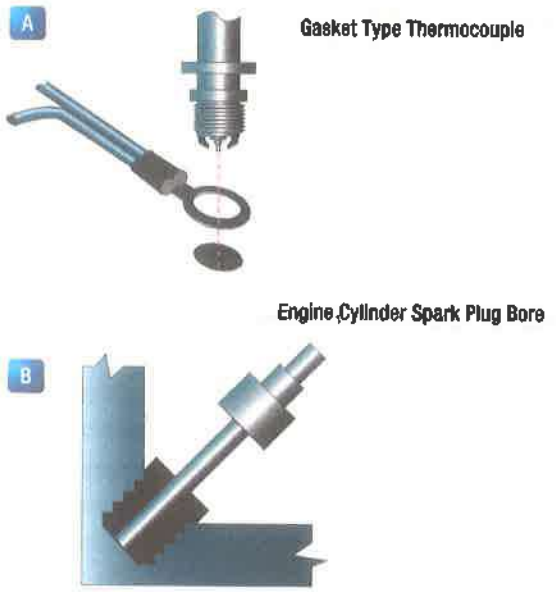


Figure 7-70. A cylinder head temperature thermocouple.

Figure 7-71. Typical thermocouple temperature indicators.

A schematic for the TIT system for one engine of a four engine turbine airplane is shown in ***Figure 7-73***. Circuits for the other three engines are identical to this system. The indicator contains a bridge circuit, a chopper circuit, a two-phase motor to drive the pointer, and a feedback potentiometer. Also included are a voltage reference circuit, an amplifier, a power-off flag, a power supply, and an over temperature warning light.

Output of the amplifier energizes the variable field of the two phase motor that positions the indicator main pointer and a digital indicator. The motor also drives the feedback potentiometer to provide a humming signal to stop the drive motor when the correct pointer position, relative to the temperature signal, has been reached. The voltage reference circuit provides a closely regulated reference voltage in the bridge circuit to preclude error from input voltage variation to the indicator power supply.

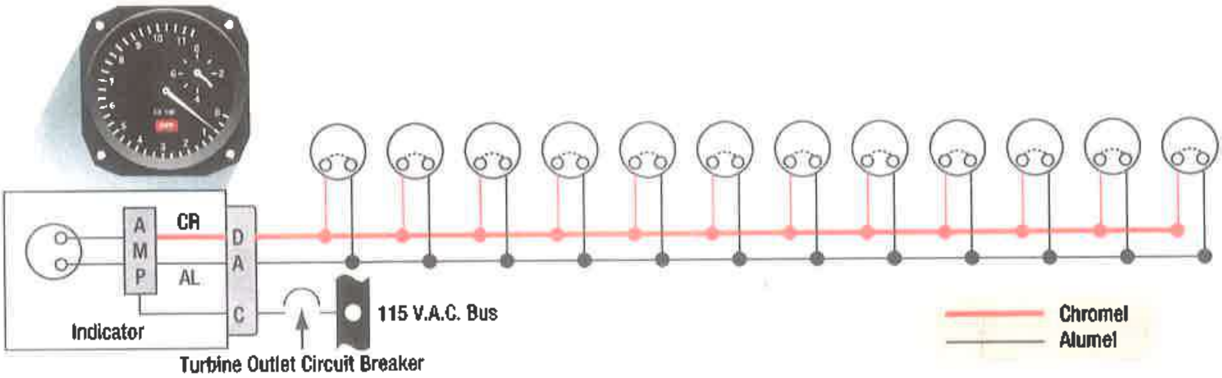


Figure 7-72. A typical exhaust gas temperature thermocouple system.

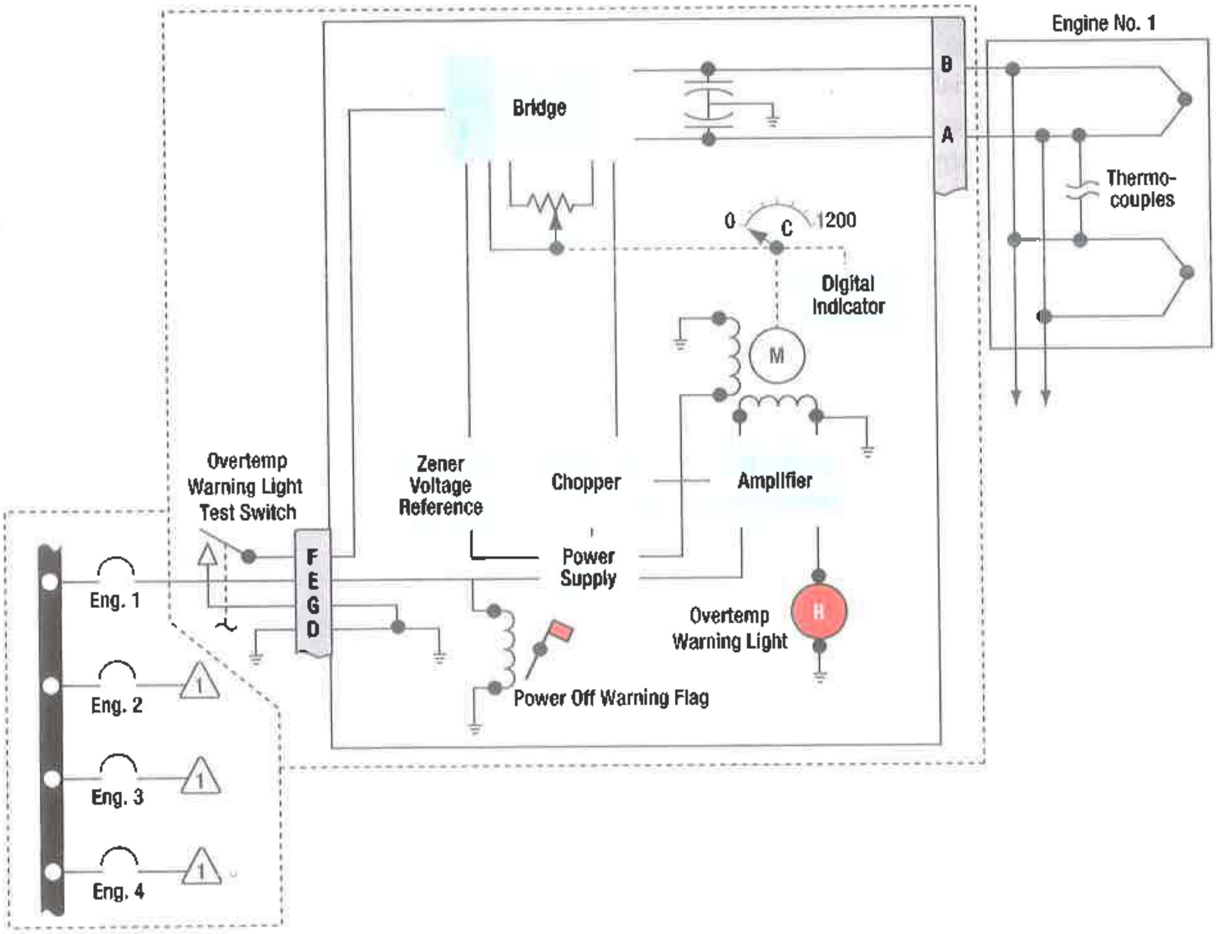


Figure 7-73. A typical analog turbine inlet temperature indicating system.

The over temperature warning light in the indicator illuminates when the TIT reaches a predetermined limit. An external test switch is usually installed so that over temperature warning lights can be tested. When the test switch is operated, an over temperature signal is simulated in each indicator's control circuit.

Digital cockpit instrumentation systems need not employ resistance indicators or adjusted servo thermocouple gauges to provide the pilot with temperature information. Sensor resistance and voltage values are input to the appropriate computer where they are adjusted, processed, monitored, and output for display on cockpit panels. They are also sent for use by other computers requiring temperature information for the control and monitoring of various integrated systems.

Total Air Temperature (TAT) systems include a sensor and an indicator with a built in resistance balance circuit. Airflow through the sensor is designed so that air with the precise temperature impacts a platinum alloy resistance element. The sensor is engineered to capture temperature variations in terms of varying the resistance of the element. When placed in the bridge circuit, the indicator pointer moves in response to the imbalance caused by the variable resistor.

More complex systems use signal correction technology and amplified signals sent to a servo motor to adjust the indicator in the cockpit. These systems include a closely regulated power supply and failure monitoring. They often use numeric drum readouts but can also be sent to an LCD driver to illuminate LCD displays. Many LCD displays are multifunctional, capable of displaying static air temperature and true airspeed. In fully digital systems, the correction signals are input into the ADC. There, they can be manipulated appropriately for cockpit display or for whichever system requires temperature information. (*Figure 7-74*)

TAT sensor/probe design is complicated by the potential of ice forming. Left unheated, a probe may cease functioning properly. The inclusion of a heating element threatens accurate data collection. Heating the probe must not affect the resistance of the sensor element. (*Figure 7-75*) Some TAT sensors channel bleed air through the units to affect the flow of outside air, so that it flows directly onto the platinum sensor without gaining added energy from the probe heater.

Figure 7-74. Different cockpit TAT displays.

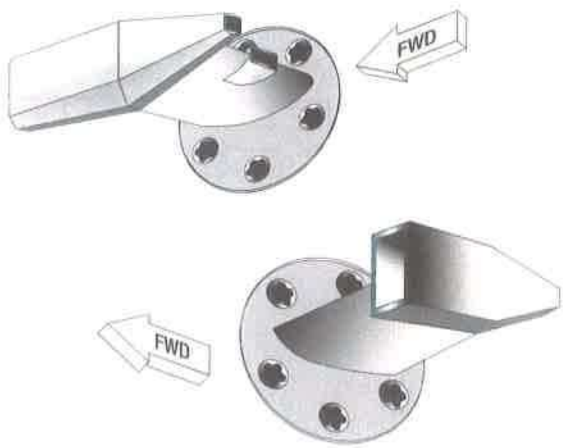


Figure 7-75. Total air temperature (TAT) probes.

PRESSURE MEASURING INSTRUMENTS

Several instruments inform the pilot of changes of pressure regarding both in flight and powerplant operations. They can be either direct reading or remote sensing. These are some of the most critical instruments on the aircraft and must inform the pilot accurately to maintain safe operations.

Pressure measurement involves some sort of mechanism able to sense changes in pressure. To inform the pilot, a technique for calibration and display of the information is then added. The type of pressure needed to be measured often makes one sensing mechanism more suited for that particular instance. The three fundamental mechanisms used in aircraft systems are the Bourdon tube, the diaphragm or bellows, and the solid state sensing device.

A Bourdon tube is illustrated in ***Figure 7-76***. The open end of a coiled tube is fixed in place and the other end is sealed and free to move. When a fluid to be measured is directed into the open end of the tube, the unfixed portion of the coiled tube tends to straighten out. The higher the pressure of the fluid, the more the tube straightens. When the pressure is reduced, the tube recoils. A pointer is attached to this moving end of the tube, usually through a linkage of shafts and gears. By calibrating the motion of the straightening tube, a face and dial of the instrument can be created.

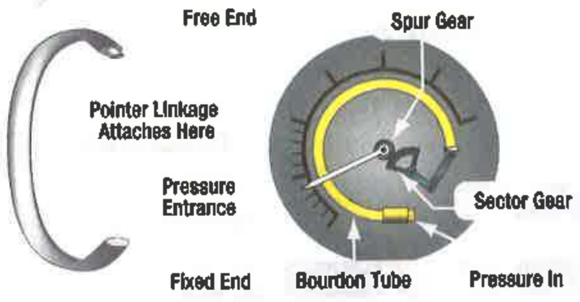


Figure 7-76. The Bourdon tube.

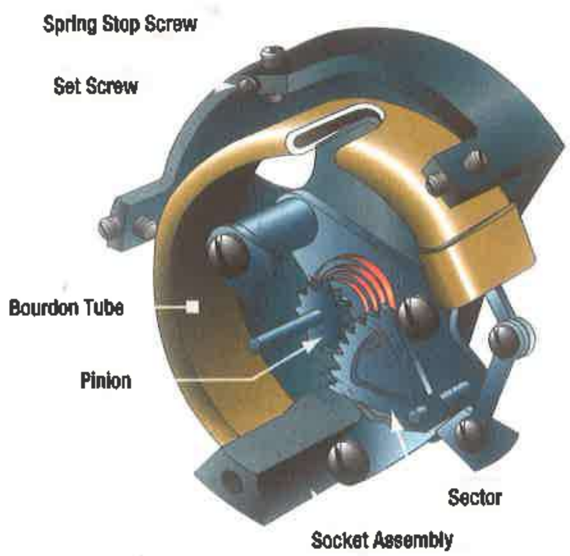


Figure 7-77. The Bourdon tube mechanism can be used to measure pressure or temperature.

The Bourdon tube is the internal mechanism for many pressure gauges used on aircraft. When high pressures need to be measured, the tube is designed to be stiff. Gauges used for lower pressures use a more flexible tube that uncoils and coils more readily. Most Bourdon tubes are made from brass, bronze, or copper. Alloys of these metals can be made to coil and uncoil consistently numerous times.

Bourdon tube gauges are simple and reliable. Some of the instruments using Bourdon tube mechanisms include the engine oil pressure and hydraulic pressure. Since the pressure of the vapor produced by a heated liquid or gas increases as temperature increases, Bourdon tube mechanisms can also be used to measure temperature. This is done by calibrating the pointer and relabeling the face of the gauge with a temperature scale. Oil temperature gauges often use Bourdon tube mechanisms. (***Figure 7-77***)

Since the sensing and display of pressure or temperature information using a Bourdon tube mechanism usually occurs in a single instrument housing, they are most often direct reading gauges. The Bourdon tube sensing device can also be used remotely. Here, the Bourdon tube motion is converted to an electrical signal and carried to the cockpit display via a wire. This is lighter and more efficient, plus eliminates the possibility of leaking fluids into the passenger compartment.

The diaphragm and bellows are two other basic sensing mechanisms used for pressure measurement. The diaphragm is a hollow, thin walled metal disk and usually corrugated. When pressure is introduced through an opening on one side of the disk, the entire disk expands. By placing linkage in contact against the other side of the disk, the movement of the pressurized diaphragm can be transferred to a pointer that registers the movement against the scale on the instrument face. (***Figure 7-78***)

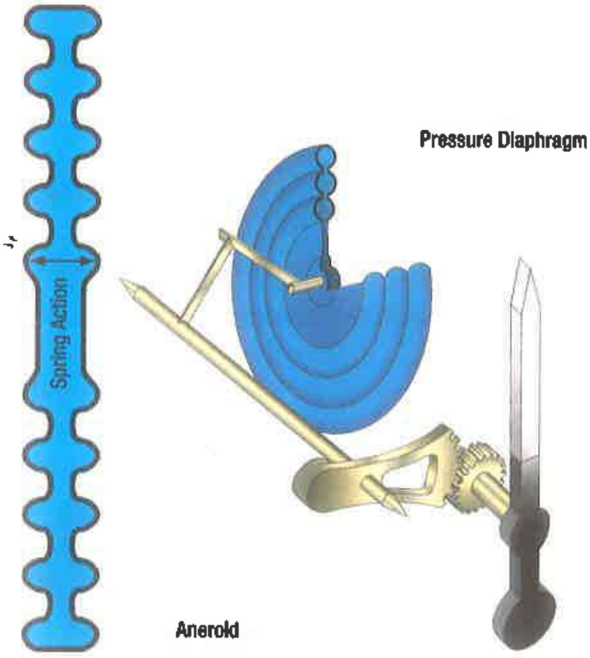


Figure 7-78. A diaphragm.

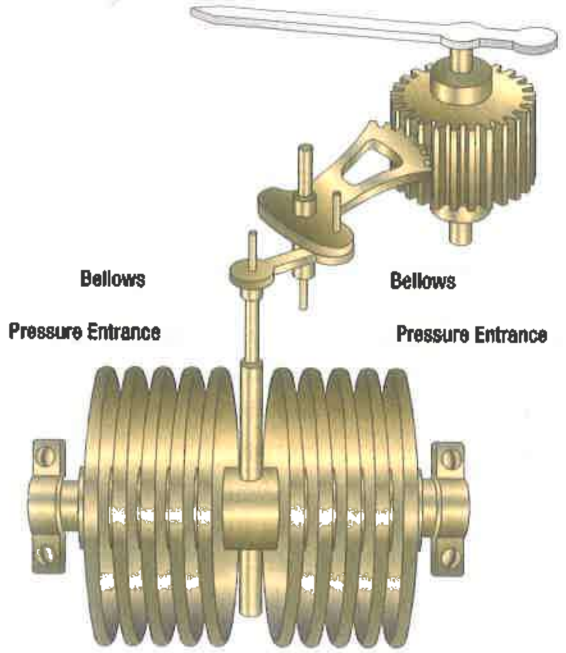


Figure 7-79. A bellows unit in a differential pressure gauge.

Diaphragms can also be sealed and evacuated before sealing, retaining absolutely nothing inside. When this is done, the diaphragm is called an aneroid. Aneroids are used in many flight instruments.

A diaphragm can also be filled with a gas to standard atmospheric pressure and then sealed. Each of these diaphragms has their uses, which are described in the next section. The common feature to all of them is that the expansion and contraction of the diaphragm side wall is the movement that correlates to increasing and decreasing pressure.

When several diaphragm chambers are connected altogether, the device is called a bellows. This accordion-like assembly of diaphragms can be especially useful when measuring the difference in pressure between two gases, called differential pressure. Just as with a single diaphragm, it is the movement of the side walls of the bellows that correlates with changes in pressure and to which a pointer linkage and gearing is attached to inform the pilot. (*Figure 7-79*)

Diaphragms, aneroids, and bellows pressure sensing devices are often located inside a single instrument housing that contains the pointer and instrument dial. Thus, many instruments that make use of these sensitive and reliable mechanisms are direct reading gauges. However, many remote sensing instrument systems also make use of the diaphragm and bellows. A transducer converts the pressure into an electrical signal and sends the signal to the gauge in the cockpit, or to a computer, for processing and subsequent display.

Examples of instruments that use a diaphragm or bellows in a direct reading or remote sensing gauge are the altimeter, the vertical speed indicator, and the manifold pressure gauge.

Solid state micro sensors are used in modern aircraft to determine the critical pressures needed for safe operation. Many of these have digital outputs ready for processing by flight instrument and other onboard computers. As with the analog sensors described above, the key to the function of solid state sensors is their consistent property changes as pressure changes. Solid state sensors used in most aviation applications exhibit varying electrical output or resistance changes when pressure changes occur. Crystalline piezoelectric, piezo-resistor, and semiconductor sensors are most common. In the typical sensor, tiny wires are embedded in the crystal or pressure sensitive semiconductor chip. When pressure deflects the crystal(s), a small amount of electricity is created or, in the case of a semiconductor chip and some crystals, the resistance changes. Since the current and resistance changes vary directly with the amount of deflection, outputs can be calibrated and used to display pressure values.

Nearly all the pressure information needed for engine, airframe, and flight instruments can be captured and calculated by solid state pressure sensors in combination with temperature sensors. Solid-state pressure sensing systems are remote-sensing systems. The sensors are mounted on the aircraft at effective locations.

Types Of Pressure

Pressure is a comparison between two forces. Absolute pressure exists when a force is compared to a total vacuum, or absolutely no pressure. It is necessary to define absolute pressure because the air in the atmosphere always exerts pressure on everything. Even when it seems there is no pressure being applied, like when a balloon is deflated, there is still atmospheric pressure inside and outside of the balloon. Thus to measure atmospheric pressure, it is necessary to compare it to a total absence of pressure, such as in a vacuum. Many aircraft instruments make use of absolute pressure values, such as the altimeter, the rate-of-climb indicator, and the manifold pressure gauge. This is usually done with an aneroid.

The most common type of pressure measurement is gauge pressure. This is the difference between the pressure to be measured and the atmospheric pressure. The gauge pressure inside the deflated balloon is therefore 0 psi. Gauge pressure is measured by ignoring the fact that the atmosphere is always exerting its pressure on everything. For example, a tire is filled with air to 32 psi at sea level and checked with a gauge to read 32 psi. The approximately 14.7 psi of air pressing on the outside of the tire is ignored. The absolute pressure in the tire is 32 psi plus the 14.7 psi that is needed to balance the 14.7 on the outside of the tire. So, the tire's absolute pressure is approximately 46.7 psi. If the same tire is inflated to 32 psi at a location 10 000 feet above sea level, the air pressure on the outside of the tire would only be approximately 10 psi, due to the thinner atmosphere. The pressure inside the tire required to balance this would be 32 psi plus 10 psi, making the absolute pressure of the tire 42 psi. So, the same tire with the same inflation and performance characteristics has different absolute pressure values. Gauge pressure, however, remains the same and is the most more useful in informing us of the condition of the tire.

Gauge measurements are simple and useful. They eliminate the need to measure varying atmospheric pressure to monitor a particular pressure situation. Gauge pressure should be assumed, unless otherwise indicated. In many instances in aviation, it is desirable to compare the pressures of two different elements to obtain useful information for operating the aircraft. When two pressures are compared in a gauge, the measurement is described as differential pressure. An aircraft airspeed indicator is a differential pressure gauge. It compares ambient air pressure to ram air pressure for determining how fast the aircraft is moving through the air. In aviation, there is also a commonly used pressure known as standard pressure. Standard pressure refers to a standard value that has been created for atmospheric pressure. This standard pressure value at sea level at 15°C is 29.92 inches of mercury ("Hg), 1 013.2 hectoPascal (hPa), or 14.7 psi.

Specific standard day values have also been established for air density, volume, and viscosity. All these values are developed averages since the atmosphere is continuously fluctuating. They are used by engineers when designing instrument systems and are sometimes used by technicians and pilots. Often, using a standard value for atmospheric pressure is more desirable than using the actual value. For example, at 18 000 feet and above, all aircraft use 29.92"Hg as a reference to indicate altitude, resulting in altitude indications in all cockpits being identical. Therefore, an accurate means is established for maintaining vertical separation of aircraft at these high altitudes.

Engine Oil Pressure Instruments

The most important instrument to perceive the health of an engine is the oil pressure gauge. Oil pressure is usually indicated in psi. The normal operating range is typically represented by a green arc on the circular gauge. For an exact acceptable operating range, consult the manufacturer operating and maintenance data. In reciprocating and turbine engines, oil is used to lubricate and cool bearing surfaces where parts are rotating or sliding past each other at high speeds. A loss of pressurized oil to these areas would rapidly cause excessive friction and over temperature conditions leading to catastrophic engine failure. Aircraft using analog instruments often use direct reading Bourdon tube oil pressure gauges. ***Figure 7-80*** shows the face of a typical oil pressure gauge of this type.

Digital instrument systems use an analog or digital remote sensing unit that outputs to the computer driving the display of the oil pressure value(s) on the cockpit display screens. Oil pressure may be displayed in a circular or linear gauge fashion and may include a numerical value on screen. Often, oil pressure is grouped with other engine parameters on the same page on the display. ***Figure 7-81*** shows this grouping on a Garmin G1000 digital display system for general aviation aircraft.

Figure 7-80. An analog oil pressure gauge.

Manifold Pressure Instruments

In reciprocating engine aircraft, the manifold pressure gauge indicates the pressure of the air in the engine induction manifold. This is an indication of power being developed by the engine. The higher the pressure of the fuel/air mixture going into the engine, the more power it can produce. For normally aspirated engines, this means that an indication near atmospheric pressure is the maximum. Turbocharged or supercharged engines pressurize the air being mixed with the fuel, so full power indications are above atmospheric pressure. Most manifold pressure gauges are calibrated in inches of mercury ("Hg), although digital displays may display in a different scale. A typical analog gauge makes use of an aneroid. When atmospheric pressure acts on the aneroid inside the gauge, the connected pointer indicates

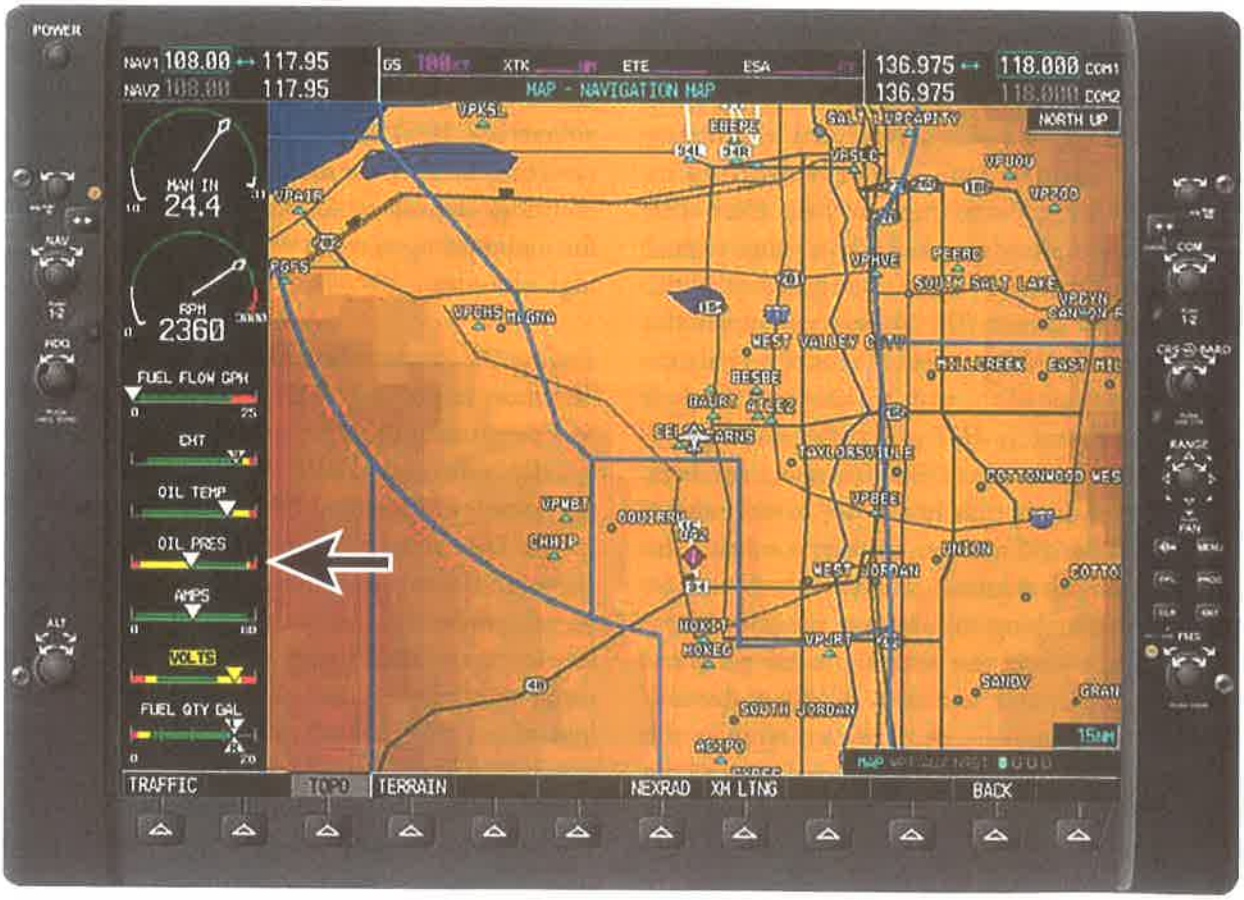


Figure 7-81. Oil pressure indication.

the current air pressure. A line running from the intake manifold into the gauge presents this pressure to the aneroid, so the gauge indicates the absolute pressure in the manifold.

Fuel Pressure Instruments

Fuel pressure gauges also provide critical information to the pilot. (*Figure 7-82*) Typically, fuel is pumped out of various fuel tanks for use by the engines. A defective fuel pump, or a tank that has been emptied beyond when there is not enough fuel entering the pump to maintain desired output pressure, is a condition that requires immediate attention. While direct-sensing fuel pressure gauges using Bourdon tubes, diaphragms and bellows sensing arrangements exist. It is undesirable to run a fuel line into the cockpit due to the potential for fire should a leak develop. Therefore, the preferred arrangement is to have a sensing mechanism as a transmitter device that uses electricity to send a signal to the indicator in the cockpit.

Hydraulic Pressure Instruments

Numerous other pressure monitoring gauges are used on complex aircraft to indicate the condition of various support systems. Hydraulic systems are commonly used to raise and lower landing gear, operate flight controls, apply brakes, and more. Sufficient pressure in the hydraulic system developed by the hydraulic pump(s) is required for normal operation of hydraulic devices. Remotely located indicators used by maintenance personnel are almost always direct reading, such as Bourdon tube types. Cockpit gauges usually have system pressures transmitted electrically from sensors for indication. *Figure 7-83* shows a hydraulic pressure transmitter in place in a high pressure hydraulic system of an aircraft.

Vacuum Pressure Instruments

Gyro pressure gauge, vacuum gauge, or suction gauge are all terms for the same gauge used to monitor the vacuum developed to actuate the air driven gyroscopic flight instruments. Air is pulled through the instruments causing the gyroscopes to spin. The speed at which the gyros spin needs to be within a certain range for correct operation. This speed is related to the suction pressure that is developed in the system. The suction gauge is important in aircraft relying solely on vacuum operated gyroscopic flight instruments. Vacuum is a differential pressure indication, meaning the pressure to be measured is compared to the atmospheric pressure by a sealed diaphragm or capsule. The gauge is calibrated in inches of mercury and measures the pressure in the system, which should be less than the atmospheric pressure. The interest is the lack of pressure. *Figure 7-84* shows a suction gauge calibrated in inches of mercury.

Pressure Switches

In aviation, it is often sufficient to monitor whether the pressure developed by a certain system is too high or too low so that an action can take place should one of these conditions occur. This is often accomplished using a pressure switch. A pressure switch is a simple device usually made to open or close an electric circuit when a certain pressure is reached in a system. It can be manufactured so that the electric circuit is normally open and can then close when a certain pressure is sensed, or for the circuit can be closed and then opened when the activation pressure is reached. (*Figure 7-85*)

Figure 7-82. A typical analog fuel pressure gauge.

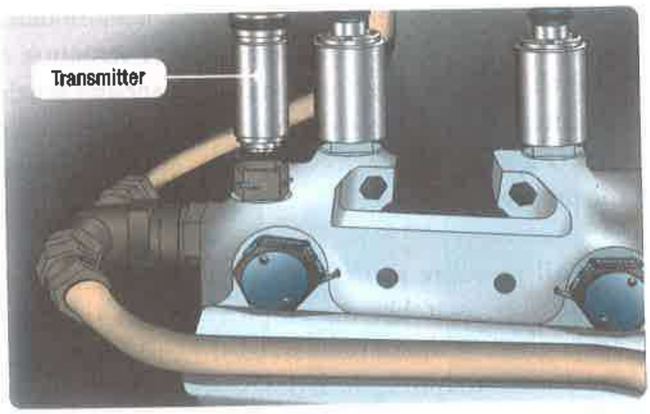


Figure 7-83. A hydraulic pressure transmitter.

INSTRUMENTS/AVIONIC SYSTEMS

Figure 7-84. Vacuum suction gauge.

Figure 7-85. A pressure switch.

Pressure switches contain a diaphragm to which the pressure being sensed is applied on one side. The opposite side of the diaphragm controls a mechanical switch of an electric circuit. Small fluctuations or buildups of pressure move the diaphragm, but not enough to throw the switch. Only when pressure meets or exceeds a preset level designed into the structure of the switch does the diaphragm move far enough for the mechanical device to close the contacts and complete the circuit. (***Figure 7-86***) Each switch must only be installed in the proper location.

A low oil pressure switch is a common example of how pressure switches are employed. It is installed in an engine so that pressurized oil can be applied to the switch diaphragm. Upon starting the engine, oil pressure increases and the pressure against the

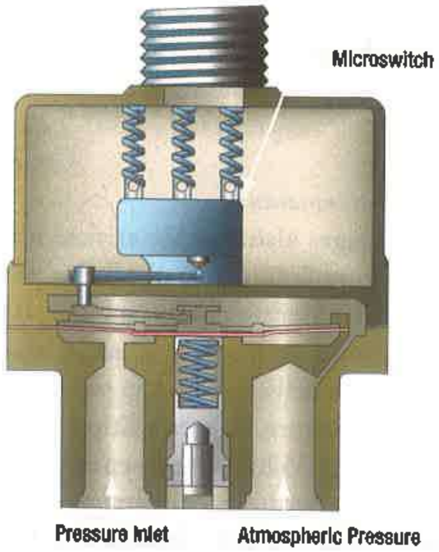


Figure 7-86. A normally open pressure switch.

diaphragm is sufficient to hold the contacts in the switch open. As such, current does not flow through the circuit and no indication of low oil pressure is given in the cockpit. Should a loss of oil pressure occur, the pressure against the diaphragm becomes insufficient to hold the contacts open. When the contacts close, they close the circuit to the indicator to warn the pilot of the situation. Pressure gauges for various components or systems work similarly. Some sort of sensing device, appropriate for the pressure being measured, is matched with an indicating display.

12.7.2 - AVIONICS SYSTEMS (ATA 22, 23, 34)

AUTOFLIGHT (ATA 22)

An automatic pilot system controls the aircraft without the pilot directly maneuvering the controls. The autopilot maintains attitude and/or direction and returns the aircraft to an original condition when it is displaced from it. Automatic pilot systems can keep aircraft stabilized laterally, vertically, and longitudinally.

The primary purpose of an autopilot system is to reduce the work strain and fatigue of controlling the aircraft and so to help the pilot to maintain exact position such as when in a hover. Autopilots have both manual and automatic modes of operation. In the manual mode, the pilot selects each maneuver and makes small inputs into an autopilot controller. The autopilot system moves flight controls of the aircraft to perform the maneuver. In automatic mode, the pilot selects the attitude and direction desired for a flight segment. The autopilot then controls the rotors to attain and maintain these parameters.

Helicopter autopilot systems provide for two, three or four axes control of an aircraft. Those that manage the aircraft around pitch and roll control the main rotor. Three axes autopilots control yaw in addition to the attitude. A helicopter with four axes can keep the vertical position and stay in a hover for a long time.

While a helicopter is not more difficult to fly than a plane, everyone agrees that rotorcraft are less stable. The three movements of a rotorcraft are defined with three axes called: pitch, roll and yaw. These flight controls are directly corrected using an old system called the Stability Augmentation System (SAS) which adjusts the attitude of the helicopter in the event of a change in the position previously defined by the pilot.

Depending on the generation, autopilots control 3 axes (pitch, roll, yaw), and can be upgraded with higher modes (heading, level, approach, navigation) up to 4 axes which also control power required for the flight and their systems management (flight planning and monitoring).

STABILITY AUGMENTATION SYSTEM

Stability Augmentation Systems mitigate short term attitude disturbances. Like all attenuation devices, they require you to keep your hands near the controls. SAS performs the basic function of stabilizing and returning the helicopter to a chosen attitude before a disturbance. The three axes of the helicopter are monitored by a gyroscope. The simplest of these systems is force compensation, which uses a magnetic brake and a series of springs to maintain cyclic control in the last chosen position. More advanced systems use electric servos which act on the flight controls. These servos receive commands from a computer connected to attitude sensors and allow the pilot to control the aircraft via the cyclic and tail rotor pedals with full stabilization in all axes. This level of assistance still requires frequent corrections on the part of the pilot.

BASIC AUTOPILOTS (3-AXES)

3-axes autopilot devices include three chains acting separately on the corresponding flight control. The pitch chain acts on the longitudinal cyclic pitch, the roll chain acts on the lateral cyclic pitch and the yaw chain acts on the tail rotor. Each is completely independent and can be engaged individually. This prevents malfunction of one channel from preventing the use of others. 3-axes autopilots allow "hands off" control which lightens the workload of the pilot. Basic stabilization is provided with information from the navigation and specific autopilot equipment. Stabilization around the pitch and roll axes is achieved from trim information provided by a vertical control unit. The stabilization of the heading is done with the information coming from a gyromagnetic compass. The altitude stabilization is done via a sensor which receives information on the altitude deviation measured by a barometric unit. (***Figure 7-87***)

This operating mode allows:
- Maneuvers around the pitch and roll axes, without loss of references.
- Maneuvers with the cyclic stick with disengagement of forces.
- Yaw control by action on the rudder pedals.
- Coordinated turns while cruising by simple action on the cyclic stick in roll.
- Slow reference changes around the pitch and roll axes using a "Beep Trim" control located on the cyclic stick.
- Maneuvers with the cyclic on the pitch or roll axes, with modification of the reference on the corresponding chain (Beep Trim and axes channels).

SUPERIOR MODES OF 3-AXES AUTOPILOTS

The installations allowing the use of higher modes associated with 3-axes autopilot systems include the 3-axes automatic piloting as described above and a coupling box which allows the maintenance of the higher modes and monitoring of the radioelectric axes. The different functions of the 3-axes autopilot permit to

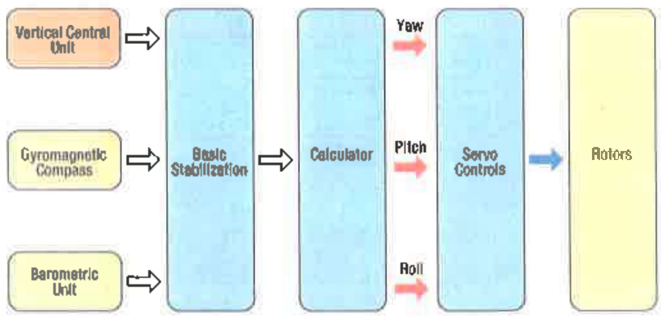


Figure 7-87. Basic functions.

control the yaw and the roll with different switches on the coupling box. (*Figure 7-88*)

Two additional superior modes link different functions through the coupling box. The first links the onboard sensors like:
-ALT - A/S - V/S - HDG

The second mode links the radio-electric sensors like:
G/S - VOR - NAV - B/C

In all cases, the functions of the autopilot must be chosen with care by the pilot. The 3-axes system is made to keep the different positions without limitations. For example, if the pilot chooses to maintain a descent position without reducing speed, it becomes possible to exceed the speed limit (Vne). The power adjustment is not a parameter that this type of autopilot can control.

SUPERIOR MODES OF 4-AXES AUTOPILOTS

Installations allowing the use of higher modes associated with 4-axes systems includes an autopilot and a coupling box that allows the activation of higher modes and the monitoring of the radioelectric axes. (*Figure 7-89*)

The difference between a 3-axes autopilot and a 4-axes autopilot is that in addition to the yaw, pitch and roll control, the collective chain can be managed automatically. It is also possible to add these options.

- Cruise Height (CRHT): capture and maintain the radio altimeter height.
- Ground Speed (GSPD): capture and maintain the ground speed.

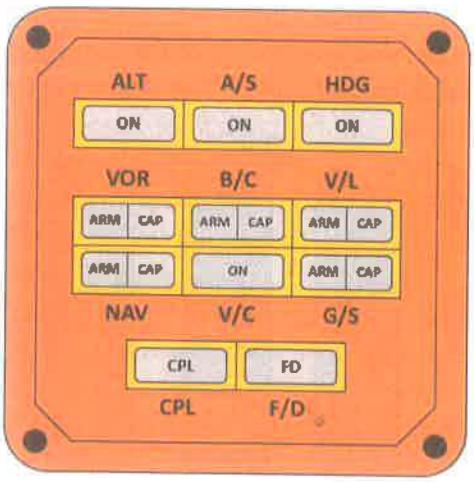


Figure 7-88. A 3-axis coupling box.

AUTOMATIC TRANSITION TO THE HOVER

There are times when the helicopter must hover over water in conditions of poor visibility and adverse weather. Even in daylight, such hovering is not ideal as the combination of wind and water with rotor downwash often masks the movement of the helicopter with respect to the water. In such cases, missing information such as from the pitot/static system have to be replaced with equally useful knowledge, such as height above water and groundspeed with respect to the surface. For example, a Doppler sensor can provide the groundspeed, and a radar altimeter can provide the height above the surface. However, it is extremely difficult for the pilot to integrate this information and maintain a safe hover in these less-than-ideal conditions. To complete this task, it is far safer to program the Automatic Flight Control System (AFCS). AFCS is described further in the flight management system section.

COMMUNICATIONS (ATA 23)

Communication systems on aircraft include radios for the flight crew to speak to air traffic control, ground operations and other aircraft while in flight. It also includes an interphone system that provides communication between the flight deck, and the cargo area while in flight and between ground crew personnel and the flight deck when the aircraft is on the ground.

VERY HIGH FREQUENCY (VHF) RADIOS

Radio voice communication between an aircraft and air traffic control is maintained with VHF radios. There are typically at least two such radios installed on an aircraft for redundancy. VHF radios are relatively short range radios in that they can operate only within line-of-sight of each other. The frequencies used range from 118.0 MHz to 136.975 MHz. 720 separate and

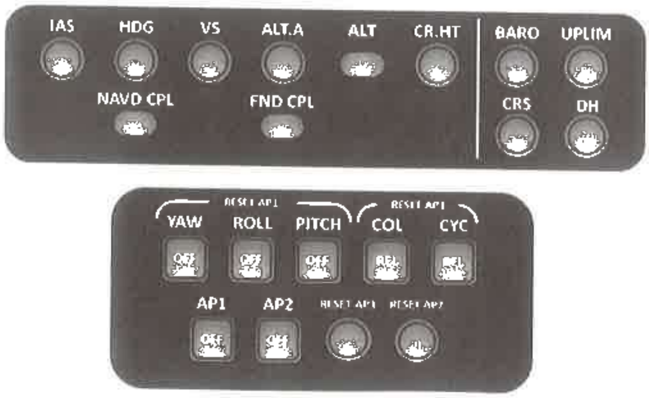


Figure 7-89. A 4-axis coupling panel.

distinct channels have been designated in this range with 25 kHz spacing between each channel. Further division of the bandwidth is possible, such as in Europe where 8.33 kHz separate each channel.

When using VHF, each party transmits and receives on the same channel. Only one party can transmit at any one time. VHF radios on aircraft are typically located in an avionics bay. They are connected to a tuning and frequency display head on the flight deck and to the flight deck audio control panel via a data bus. (***Figure 7-90***) The audio control panel illustrated in ***Figure 7-91*** allows any of three different VHF radios to be chosen.

HIGH FREQUENCY (HF) RADIOS

Some aircraft also use high frequency radios for long distance communication with an operator base station. HF radio waves bounce off the ionosphere layer of the atmosphere. This refraction extends the range of HF signals beyond line-of-sight. The frequency range is between 2 to 25 MHz.

Figure 7-91. A typical audio control panel.

Figure 7-90. Communications and other devices in an avionics bay.

SATELLITE COMMUNICATION SYSTEMS

Use of satellite communication is common on aircraft. SATCOM uses a combination of ground stations and orbiting satellites to relay transmissions to and from the aircraft. INMARSAT is one such commercially available system that includes ten orbiting satellites used to provide several different services. Airline operators typically have SATCOM radios for enroute communication with ground stations. SATCOM is also used to connect passengers to the internet.

SERVICE INTERPHONE SYSTEM

Some aircraft are equipped with a service interphone. This is an intercom type system that allows communication between the flight deck and the cargo areas as well as connecting each area with ground personnel outside the aircraft. Cabin phones (mikes and receivers) become active when picked up for use. To broadcast over the cabin speakers, a switch must be pressed.

The interphone system is accessed by maintenance personnel outside the aircraft by plugging a headset into one of the many jacks spread around the exterior of the aircraft. Exterior interphone jacks are typically located in cargo bays and equipment bays.

Modern interphone systems are controlled by an Audio Management Unit (AMU) or similar "black box" located in the equipment bay. They are DC powered and often incorporate a ground crew call system. The ground crew call system creates audible and visual alerts on the flight deck when a ground crew wishes to speak with someone on the flight deck using the interphone. It can also be used on the flight deck to notify ground crew to plug in to the interphone to communicate.

NAVIGATION SYSTEMS (ATA 34)

In the early years of aviation, a compass, a map, and dead reckoning were the only navigational tools. These were marginally reassuring if weather prevented the pilot from seeing the terrain below. Voice radio transmission from someone on the ground to the pilot indicating that the aircraft could be heard overhead was a preview of what electronic navigational aids could provide. For aviation to become a safe and reliable means of transportation, a better navigation system needed to be developed. Early flight instruments contributed greatly to safety when the ground was obscured by clouds. Navigation aids were needed to indicate where an aircraft was over the earth as it progressed towards its destination. In the 1930s and 1940s, a radio navigation system was used that was a low frequency, four-course radio range system. Airports and selected navigation waypoints broadcast two Morse code signals with finite ranges and patterns.

Pilots tuned to the frequency of the broadcasts and flew in an orientation pattern until both signals were received with increasing strength. The signals were received as a blended tone of the highest volume when the aircraft was directly over the broadcast antenna.

VOR NAVIGATION SYSTEM

One of the oldest and most useful navigational aids is the VOR system. The system was constructed after WWII and is still in use today. It consists of thousands of land based transmitter stations that communicate with radio receiving equipment onboard aircraft. Many of the VORs are located along predefined Victor airways. Ground VOR transmitters are also located at airports where they are known as TVOR (Terminal VOR). The U.S. Military has a navigational system known as TACAN that operates similarly to the VOR system. Sometimes VOR and TACAN transmitters share a location. These sites are known as VORTACs.

The position of all VORs, TVORs, and VORTACs are marked on aeronautical charts along with the name of the station, the frequency to which an airborne receiver must be tuned to use the station, and a Morse code designation for the station. Some VORs also broadcast a voice identifier on a separate frequency that is included on the chart. (***Figure 7-92***)

VOR uses VHF radio waves (108-117.95 MHz) with 50 kHz separation between each channel. This keeps atmospheric interference to a minimum but limits the VOR to line-of-sight usage. To receive VOR VHF radio waves, a V-shaped, horizontally polarized bi-pole antenna is used. A typical location for the V dipole is in the vertical fin. Follow the manufacturer's instructions for installation location. (***Figure 7-93***)

VOR transmitter signals propagate 360° from the unit and are used by aircraft to navigate to and from the station with the help of an onboard VOR receiver and display instruments. A pilot is not required to fly a pattern to intersect the signal from a VOR station since it propagates out in every direction. The radio waves are received as long as the aircraft is in range of the ground unit and regardless of the aircraft direction of travel. (***Figure 7-94***)

Figure 7-92. A VOR ground station.

Figure 7-93. V-shaped, horizontally polarized, bi-pole antennas.

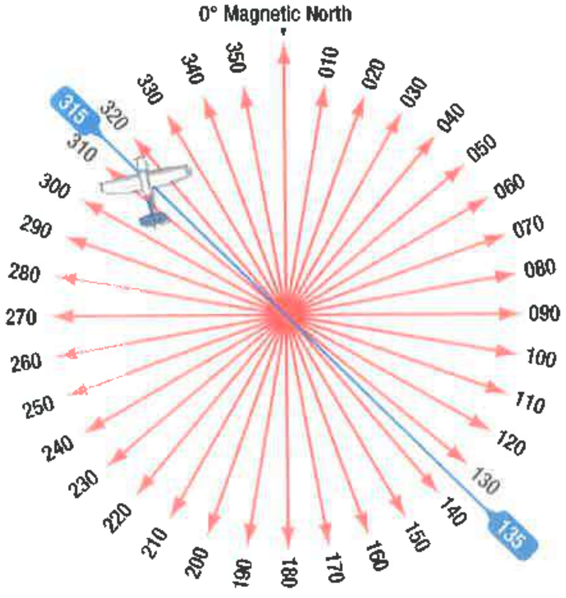


Figure 7-94. A VOR transmitter.

Most aircraft carry a dual VOR receiver. Sometimes, the VOR receivers are part of the same avionics unit as the VHF communication transceiver(s). These are known as NAV/COM radios. Internal components are shared since frequency bands for each are adjacent. (***Figure 7-95***)

Large aircraft may have dual receivers and even dual antennas. Normally, one receiver is selected for use and the second is tuned to the frequency of the next VOR station to be encountered enroute. A means for switching between NAV 1 and NAV 2 is provided as is a switch for selecting the active or standby frequency. (***Figure 7-96***)

VOR receivers can be coupled with Instrument Landing System (ILS) and glideslope receivers. Older aircraft are often equipped with a VOR gauge dedicated to displaying only VOR information. This is called an Omni Bearing Selector (OBS) or a Course Deviation Indicator (CDI). (***Figure 7-97***)

As flight instruments and displays have evolved, VOR navigation has been integrated into other instruments displays, such as the Radio Magnetic Indicator (RMI), the Horizontal Situation Indicator (HSI), an EFIS display or an Electronic Attitude Director Indicator (EADI). Flight management systems and automatic flight control systems are also made to integrate VOR information to automatically control the aircraft on its planned flight segments. Flat panel Multi Function

Figure 7-95. A NAVCOM receiver.

Figure 7-96. An VOR control head.

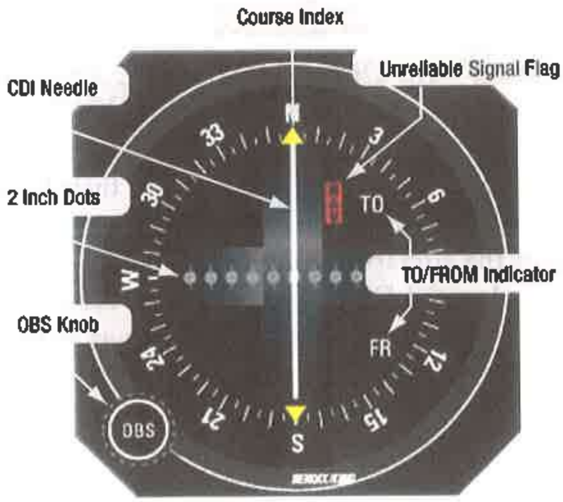


Figure 7-97. A traditional VOR gauge.

Displays (MFDs) integrate VOR information into moving map presentations and other selected displays. However, the basic information of the radial bearing in degrees, course deviation indications, and to/from information remains unchanged. (***Figure 7-98***)

At large airports, an ILS guides the aircraft to the runway while on an instrument landing approach. The aircraft VOR receiver is used to interpret the radio signals. It produces a more sensitive CDI on the same

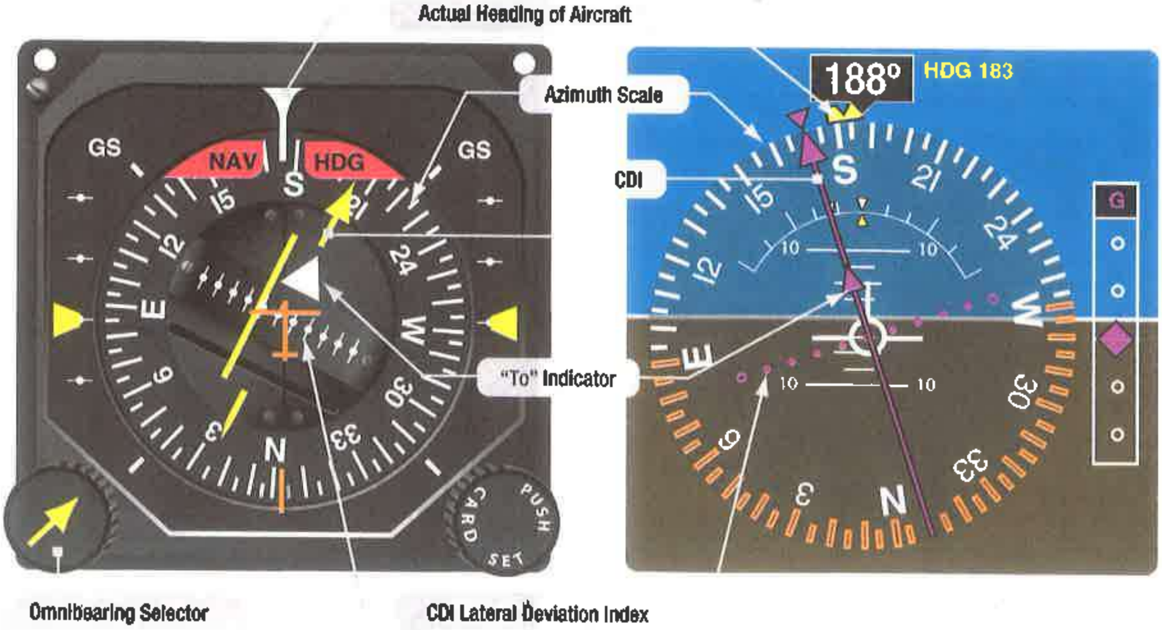


Figure 7-98. A mechanical HSI (left) and an electronic HSI (right).

instrument display as the VOR. This part of the ILS is known as the localizer. While tuned to the ILS localizer frequency, the VOR circuitry of the VOR/ILS receiver is inactive.

It is common at VOR stations to combine the VOR transmitter with Distance Measuring Equipment (DME) or a Non-Directional Beacon (NDB) such as an ADF transmitter and antenna. When used with a DME, pilots can gain an exact fix on their location using the VOR and DME together. Since the VOR indicates the aircraft bearing to the VOR transmitter and a co-located DME indicates how far away the station is, this relieves the pilot from having to fly over the station to know with certainty his/her location.

AUTOMATIC DIRECTION FINDER (ADF)

An ADF operates off of a ground signal transmitted from a Non-Directional Beacon (NDB). Early Radio Direction Finders (RDF) used the same principle.

A vertically polarized antenna is used to transmit Low Frequency radio waves in the 190 kHz to 535 kHz range. A receiver on the aircraft was tuned to the transmission frequency of the NDB. Using a loop antenna, the direction to or from the antenna could be determined by monitoring the strength of the signal received. The NDB signals were modulated with unique Morse code pulses that enabled the pilot to identify the beacon to which she or he was navigating.

The ADF improved the RDF concept. The broadcast frequency range was expanded to include medium frequencies up to about 1 800 kHz. The heading of the aircraft no longer needed to be changed to locate the broadcast transmission antenna. In early model ADFs, a rotatable antenna was used instead. The antenna rotated to seek the position in which the signal was null. The direction to the broadcast antenna was shown on an azimuth scale of an ADF indicator in the flight deck. This type of instrument is still found in use today. It has a fixed card with 0° always at the top of a non-rotating dial. A pointer indicates the relative bearing to the station. When the indication is 0°, the aircraft is on course to (or from) the station. (*Figure 7-99*)

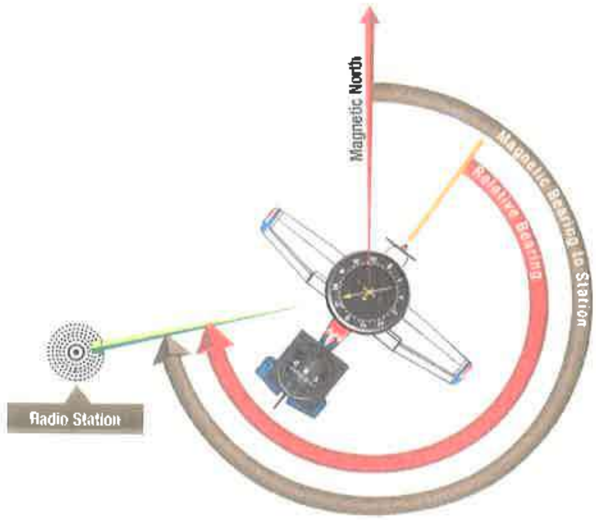


Figure 7-99. The function of an ADF indicator.

As ADF technology progressed, indicators with rotatable azimuth cards became the norm. When an ADF signal is received, the pilot rotates the card so that the present heading is at the top of the scale. This results in the pointer indicating the magnetic bearing to the ADF transmitter. This is more intuitive and consistent with other navigational practices. (***Figure 7-100***)

In modern ADF systems, an additional antenna is used to remove the ambiguity concerning whether the aircraft is heading to or from the transmitter. It is called a sense antenna. The onboard ADF receiver needs only to be tuned to the correct frequency of the broadcast transmitter for the system to work. The loop and sense antenna are normally housed in a single, low profile housing. (***Figure 7-101***)

Any ground antenna transmitting low or medium frequency radio waves in range of the aircraft receiver can be used for ADF. This includes those from AM radio stations. With an AM radio station transmission, the AM broadcast is heard instead of a station identifier code. The frequency for an NDB transmitter is given on an aeronautical chart next to a symbol for the transmitter. The identifying designator is also given. (***Figure 7-102***)

ADF receivers can be mounted in the flight deck with the controls accessible to the user. This is found on many general aviation aircraft. Alternately, the ADF receiver can be mounted in a remote avionics bay with only the control head in the flight deck. Dual ADF receivers are common. ADF information can be displayed on the ADF indicators mentioned or it can be digital. Modern MFDs usually display the ADF digitally. (***Figure 7-103***)

Continued refinements to ADF technology have brought it to its current state. The rotating receiving antenna is replaced by a fixed loop with a ferrite core. This increases sensitivity and allows a smaller antenna to be used. The most modern ADF systems have two loop antennas mounted at 90° to each other. Technicians should note that the installation of the ADF antenna is critical to a correct indication since it is a directional device. Calibration with the longitudinal axis of the fuselage or nose of the aircraft is important. Follow all manufacturer instructions.

Figure 7-100. A movable card ADF indicator.

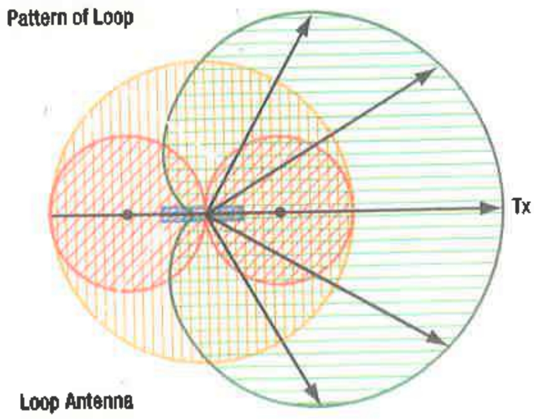


Figure 7-101. The reception fields of a loop and sense antenna.

RADIO MAGNETIC INDICATOR (RMI)

To save space in the instrument panel and to consolidate related information into one easy to use location, the RMI combines indications from a magnetic compass, VOR, and ADF into one instrument. (***Figure 7-104***)

The azimuth card of the RMI is rotated by a remotely located flux gate compass. Thus, the magnetic heading of the aircraft is always indicated. The lubber line is usually a marker or a triangle at the top of the instrument dial.

Figure 7-102. Non-directional broadcast antenna.

Figure 7-103. A cockpit mountable ADF receiver.

The VOR receiver drives the solid pointer to indicate the magnetic direction to a tuned station. When the ADF is tuned to an NDB, the double pointer, indicates the magnetic bearing to the NDB. Since the flux gate compass continuously adjusts the azimuth card so that

Figure 7-104. A radio magnetic indicator (RMI).

the aircraft heading is at the top of the instrument, pilot workload is reduced. The pointers indicate where the VOR and ADF transmission stations are in relationship to where the aircraft is currently positioned. Push buttons allow conversion of either pointer to either ADF or VOR for navigation.

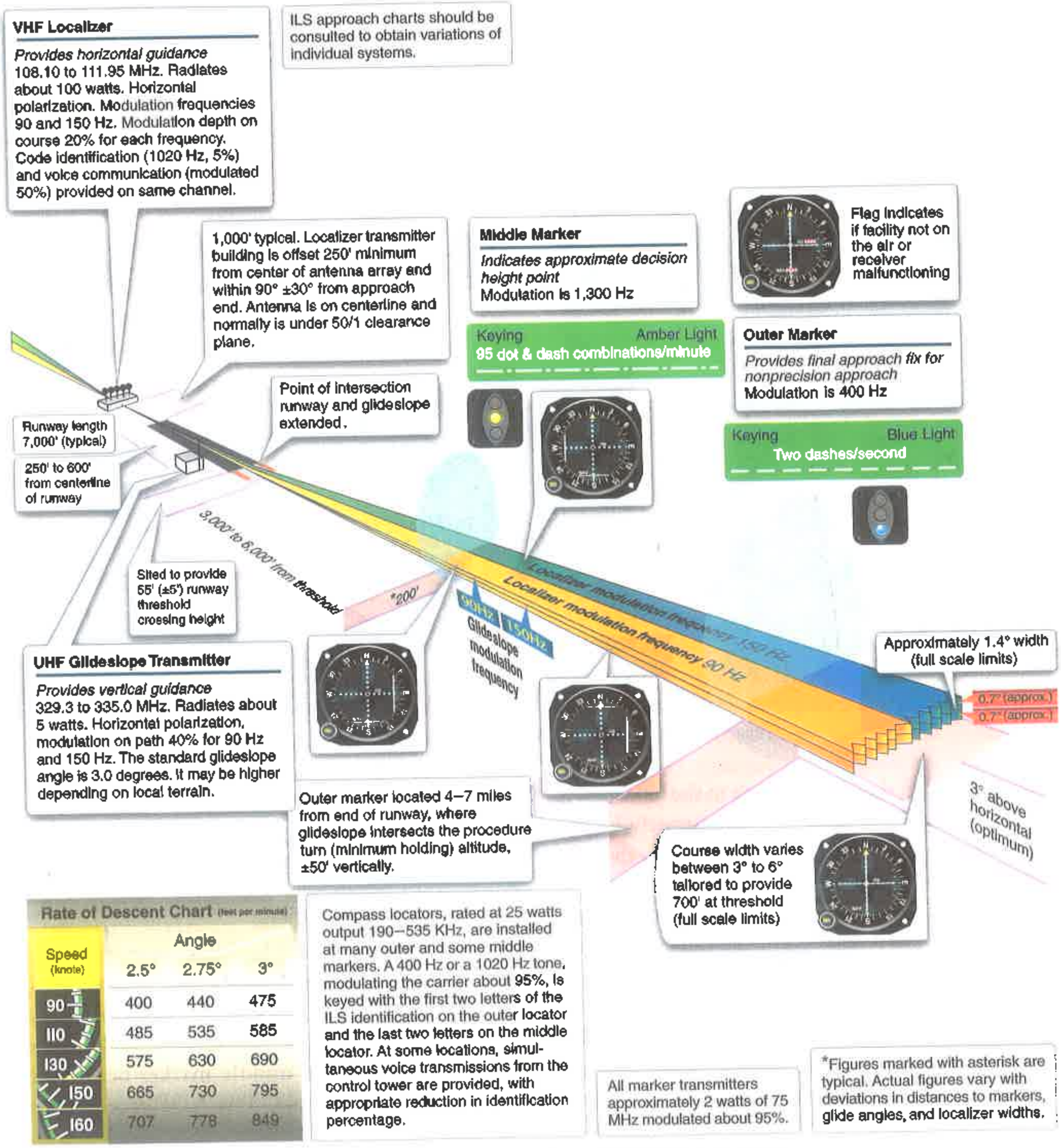
INSTRUMENT LANDING SYSTEMS (ILS)

An ILS is used to land an aircraft when visibility is poor. This radio navigation system guides the aircraft down a slope to the touch down area on the runway. Multiple radio transmissions are used that enable an exact approach to landing with an ILS. A localizer is one of the radio transmissions. It is used to provide horizontal guidance to the center line of the runway. A separate glideslope broadcast provides vertical guidance of the aircraft down the proper slope to the touch down point.

Compass locator transmissions from outer and middle approach marker beacons aid the pilot in intercepting the approach navigational aid system. Marker beacons provide distance from the runway. Together, all these radio signals make an ILS a fully accurate and reliable means for landing an aircraft. (***Figure 7-105***)

LOCALIZER

The localizer broadcast is a VHF broadcast in the lower range of the VOR frequencies (108 MHz-111.95 MHz) and on odd frequencies only. Two modulated signals are produced from a horizontally polarized antenna complex beyond the far end of the approach runway. They create an expanding field that tapers to runway width near the landing threshold. The left side of the approach area is filled with a VHF carrier wave modulated with a 90



Speed (knots)	Angle 2.5°	2.75°	3°
90	400	440	475
110	485	535	585
130	575	630	690
150	665	730	795
160	707	778	849

Figure 7-105. Components of an instrument landing system (ILS).

Hz signal. The right side of the approach contains a 150 MHz modulated signal. The aircraft VOR receiver is tuned to the localizer VHF frequency that can be found on published approach plates and aeronautical charts.

If the aircraft receives a 150 Hz signal, the CDI of the VOR/ILS display deflects to the left. This indicates that the runway is to the left. The pilot must correct course with a turn to the left. This centers the CDI on the display and centers the aircraft with the centerline of the runway. If the 90 Hz signal is received by the VOR, the CDI deflects to the right. The pilot must turn toward the right to center the aircraft with the runway center line. (*Figure 7-106*)

GLIDESLOPE

The vertical guidance required for an aircraft to descend for a landing is provided by the glideslope of the ILS.

Figure 7-106. An ILS localizer antenna.

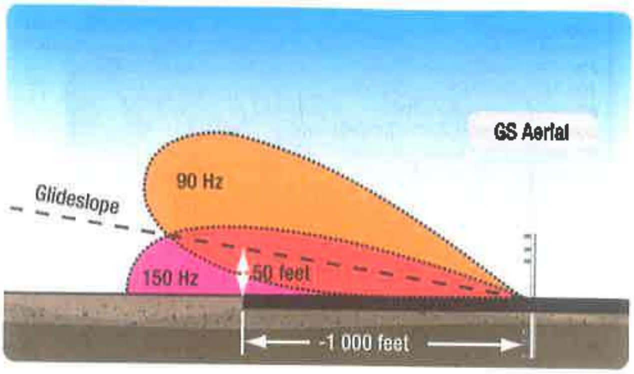


Figure 7-107. A glideslope antenna.

Figure 7-108. A traditional and digital course deviation indicator.

Radio signals funnel the aircraft down to the touchdown point on the runway at an angle of approximately 3°. The transmitting glideslope antenna is located off to the side of the approach runway approximately 300 meters from the threshold. It transmits in a wedge-like pattern with the field narrowing as it approaches the runway. (***Figure 7-107***)

Like the localizer, the glideslope transmits two signals, one modulated at 90 Hz and the other modulated at 150 Hz. The glideslope receiver deciphers the signals like the localizer receiver. It drives a vertical course deviation indicator known as the glideslope indicator. The glideslope indicator operates identically to the localizer only 90° to it. The VOR/ILS localizer CDI and the glideslope are displayed together on whichever instrumentation is in the aircraft. (***Figure 7-108***)

The UHF antenna for aircraft reception of the glideslope comes in many forms. A single dipole antenna mounted inside the nose of the aircraft is common. Antenna manufacturers have also incorporated glideslope reception into the same dipole antenna used for the VHS VOR/ILS reception. Blade type antennas are also used. (***Figure 7-109***) ***Figure 7-110*** shows a VOR and a glideslope receiver for a general aviation aircraft ILS.

COMPASS LOCATORS

It is imperative that a pilot be able to intercept the ILS to enable its use. A compass locator is a transmitter designed for this purpose. There is one located at the outer marker beacon 6-11 km from the runway threshold. Another may be located at the middle marker beacon about 1 km from the threshold. The ADF receiver is used

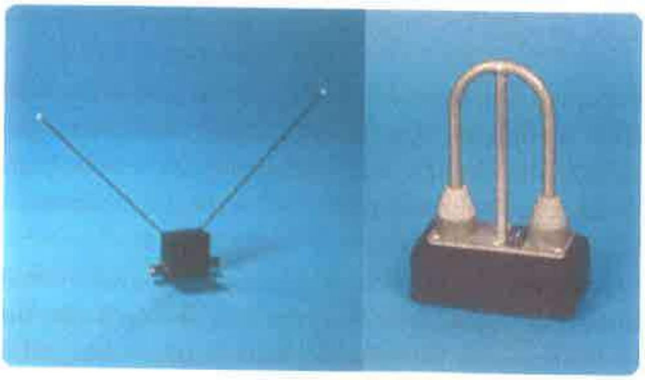


Figure 7-109. Glideslope antennas.

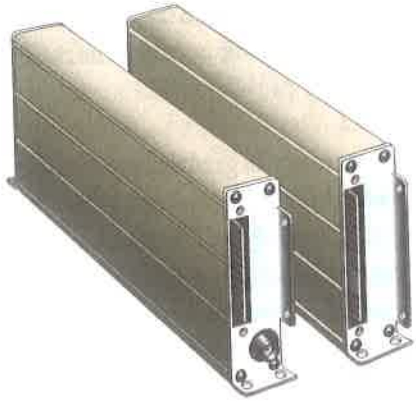


Figure 7-110. A localizer and a glideslope receiver.

Figure 7-112. Various marker beacon instrument panel display lights.

to intercept the locator, so no additional equipment is required. Once located, the pilot maneuvers the aircraft to fly down the glidepath to the runway. (***Figure 7-110***)

MARKER BEACONS

Marker beacons are the final radio transmitters used in an ILS. They transmit signals that indicate the position of the aircraft along the glidepath to the runway. The transmission is very narrow and directed straight up. A marker beacon receiver receives the signal and uses it to light a blue light on the instrument panel. This, plus the oral tone in combination with the localizer and the glideslope indicator, positively locates the aircraft on an approach. (***Figure 7-111***)

A middle marker beacon is also used. It is located on approach approximately 1 km from the runway and transmits at 75 MHz. The middle marker transmission is modulated to not be confused with the all dash tone of the outer marker. When the signal is received, it is used in the receiver to illuminate an amber colored light on the instrument panel. (***Figure 7-112***)

Some ILS approaches have an inner marker beacon that transmits a signal at 3 000 Hz in a series of dots only. It is placed at the land-or-go-around decision point of the approach close to the runway threshold. If present, the signal is used to illuminate a white light on the instrument panel. The three marker beacon lights are usually incorporated into the audio panel of a general aviation aircraft or may exist independently on a larger aircraft. Electronic display aircraft usually incorporate marker lights close to the glideslope display. ILS radio components can be tested with an ILS test unit. Localizer, glideslope, and marker beacon signals are generated to ensure proper operation of receivers and correct display on flight deck instruments. (***Figure 7-113***)

MICROWAVE LANDING SYSTEM (MLS)

A Microwave Landing System is an all-weather precision radio guidance system intended for installation at major airports to assist aircraft in landing, including in blind conditions. MLS allows an approaching aircraft to determine when it is aligned with the destination runway and on the correct glide path for a safe landing.

Figure 7-111. An outer marker transmitter antenna.

Figure 7-113. An ILS test unit.

MLS has many advantages over ILS, including a greater selection of channels to avoid interference with nearby installations, excellent all-weather performance, a small footprint at the airport and wide vertical and horizontal "capture" angles which allow approaches to wider areas around the airport.

MLS antennas are smaller and transmit a higher frequency signal. They do not have to be placed at a specific location as they can offset their signals electronically. This makes placement easier compared with the physically larger ILS systems which had to be placed at the ends of the runways and along the approach path.

A further advantage is that the MLS signals cover a very wide fan shaped area off the end of the runway, allowing controllers to direct aircraft approaching from a variety of directions or guide aircraft along a segmented approach. In comparison, ILS could only guide the aircraft down a single straight line, requiring controllers to distribute planes along only that line. MLS thus allows aircraft to approach from whatever direction they were already flying in, as opposed to flying to a parking orbit before being able to "capture" the ILS signal. This is particularly valuable at larger airports, as it allows aircraft to be separated horizontally much closer to the airport. These advantages are similar regarding elevation as the fan shaped coverage allows for variations in descent rate, making MLS useful for aircraft with steeper approach angles such as helicopters.

Unlike ILS, which required a variety of frequencies to broadcast the various signals, MLS uses a single frequency, broadcasting the azimuth and altitude information one after the other. This reduces the chance of frequency conflicts. MLS also offers two hundred separate channels, making conflicts between airports in the same area easily preventable.

Finally, MLS accuracy is greatly improved over ILS. For example, standard DME equipment used with ILS offer a range accuracy of only ±350 meters. MLS improved this to ±30 meters in what is referred to as DME/P (for precision), and offer similar improvements in azimuth and altitude. This allowed MLS to guide extremely accurate CAT III approaches, whereas this previously required an expensive ground based high precision radar.

MLS identification is a four letter designation starting with the letter M and it is transmitted in International Morse Code at least six times per minute by the approach azimuth (and back azimuth) ground equipment.

MLS Expansion Capabilities

The standard MLS configuration can be expanded by adding the following functions:

- Back azimuth: provides lateral guidance for missed approach and departure navigation.
- Auxiliary data transmissions: Provides additional data, including refined airborne positioning, meteorological information, runway status, and other supplementary information.

FLIGHT DIRECTOR SYSTEMS; DISTANCE MEASURING EQUIPMENT (DME)

Many VOR stations are co-located with the military version of the VOR station, known as TACAN. When this occurs, the navigation station is known as a VORTAC station. Civilian aircraft can then make use of one of the TACAN features not originally installed at civilian VOR stations - Distance Measuring Equipment (DME). A DME system calculates the distance from the aircraft to the DME station and displays it on the flight deck. It can also display calculated aircraft speed and estimated time of arrival when the aircraft is traveling to the station. DME ground stations have subsequently been installed at civilian VORs, as well as with ILS localizers. These are known as VOR/DME and ILS/ DME or LOC/DME. The latter aid for use in approach to the runway during landings. The DME system consists of an airborne transceiver, display, and antenna, as well as the ground based DME unit and its antenna. (***Figure 7-114***)

DME is useful because when knowing the bearing (from the VOR) and the distance to a known point (the DME antenna at the VOR), a pilot can positively identify the location of the aircraft. DME operates in the Ultra High Frequency (UHF) range from 962 MHz to 1 213 MHz. A carrier signal transmitted from the aircraft is modulated with a string of integration pulses. The ground unit receives the pulses and returns a signal to the aircraft. The time that transpires for the signal to be sent and returned is calculated and converted into nautical miles for display. Time to station and speed are also calculated and displayed. DME readout can be on a dedicated DME display or it can be part of an EHSI, EADI, EFIS, or on the primary flight display. (*Figure 7-115*)

The DME frequency is paired with the co-located VOR or VORTAC frequency. When the correct frequency is tuned for the VOR signal, the DME is tuned automatically. Tones are broadcast for the VOR station identification and then for the DME. The hold selector on a DME panel keeps it tuned in while the VOR selector is tuned to a different VOR. In most cases, the UHF signal is transmitted and received via a small blade type antenna mounted to the underside of the fuselage centerline. (*Figure 7-116*)

A traditional DME displays the distance from the DME station to the aircraft. This is called the slant distance and is very accurate. However, since the aircraft is at altitude, the distance to the DME station from a point directly beneath the aircraft is shorter. Some modern DMEs are equipped to calculate this difference and to display it. (*Figure 7-117*)

AREA NAVIGATION SYSTEMS (RNAV)

Area navigation is a general term used to describe the navigation from point A to point B without directly flying over the navigational aids such as the VORs,

Figure 7-114. A VOR with DME ground station.

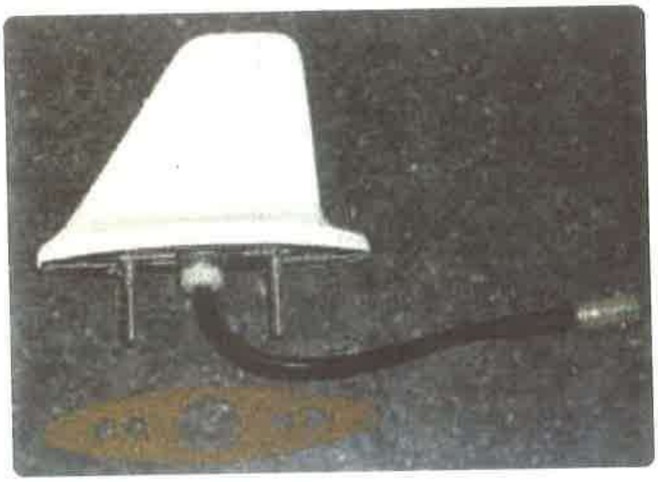


Figure 7-116. A typical aircraft mounted DME antenna.

Figure 7-115. Distance information from the DME.

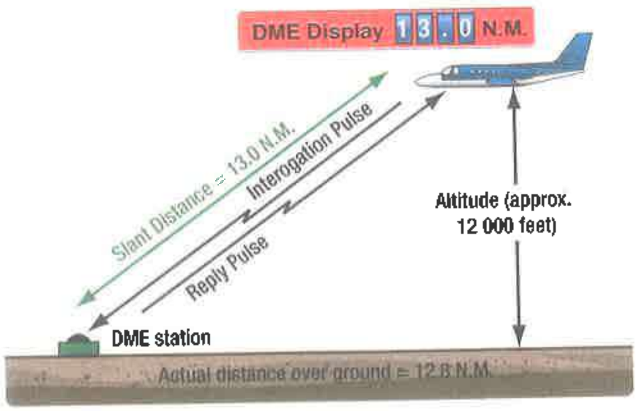


Figure 7-117. Many DMEs only display the slant distance.

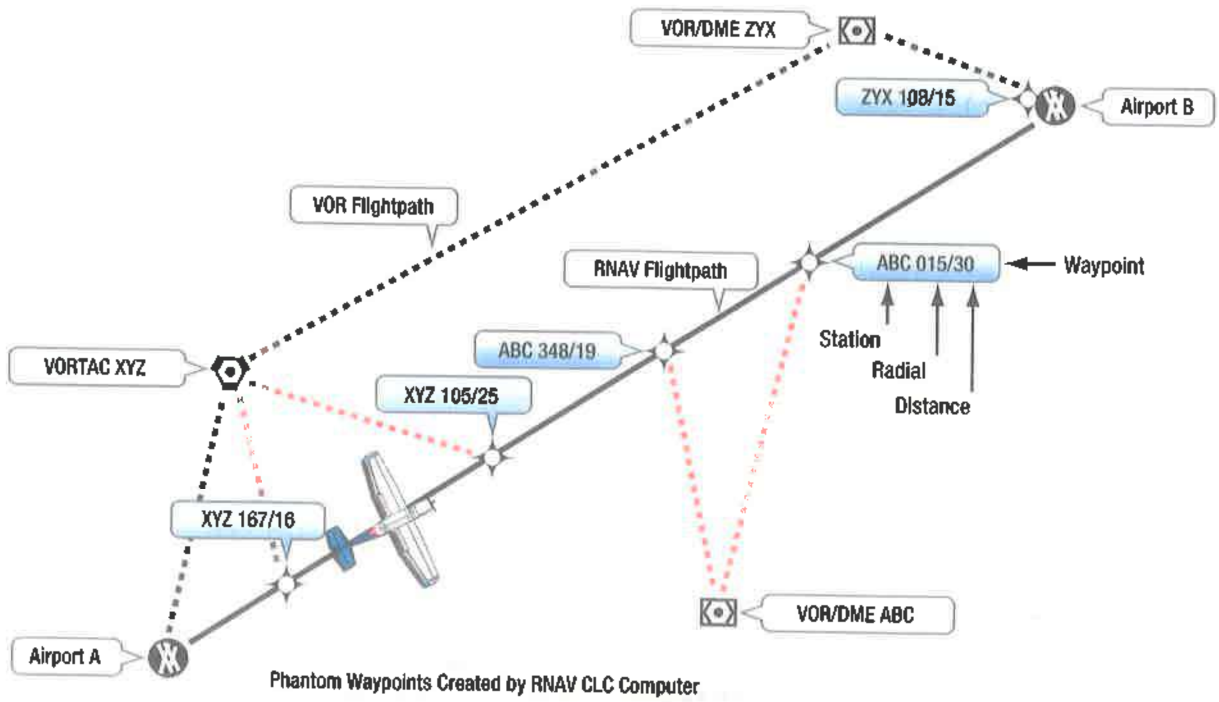


Figure 7-118. The pilot uses the aircraft's course deviation indicator.

VORTACs or those based around LORAN. All RNAV systems make use of waypoints. A waypoint is a designated geographical location used for route definition or progress reporting. It can be defined or described using latitude/longitude coordinates, or in the case of VOR based RNAV, a point on a VOR radial followed by that point distance from the VOR station (for example 200/25 means a point 25 nautical miles from the VOR station on its 200° radial).

Figure 7-118 illustrates an RNAV route of flight from airport A to airport B. The VOR/DME stations shown are used to create phantom waypoints that may be overflown rather than the actual stations. This allows a more direct route to be taken. The phantom waypoints are entered into the RNAV course-line computer as a radial and distance number pair. The computer then creates the waypoints and causes the aircraft CDI to operate as though they are actual VOR stations. A mode switch allows the choice between standard VOR navigation and RNAV. RNAV uses the VOR receiver, antenna, and VOR display equipment such as the CDI. The computer in the RNAV unit uses basic geometry and trigonometry calculations to produce heading, speed, and time readouts for each waypoint. VOR stations need to be within line of sight and operational range from the aircraft for RNAV use. (*Figure 7-119*)

Figure 7-119. A RNAV unit.

RNAV has increased in flexibility with the development of GPS. Integration of GPS data into a planned RNAV flight plan is possible as is GPS route planning without the use of a VOR station.

FLIGHT MANAGEMENT SYSTEM

A Flight Management System (FMS) is an on-board multi-purpose navigation, performance, and aircraft operations computer designed to provide virtual data and operational harmony between elements associated with a flight from preengine start to landing and engine shut-down.

An FMS comprises four main components:

- The Flight Management Computer (FMC) is a computer that uses a large data base to allow routes to be pre-programmed and fed into the system by means of a data loader. The system is constantly updating with the aircraft's position by referencing available navigational aids. The most appropriate aids are automatically selected during updates.

- The Automatic Flight Control (AFC) receives sensor information from other aircraft systems. Depending on whether the aircraft is under Autopilot or manual control, AFC mode selections made by the crew will either automatically move and control the aircraft flight control surfaces or display Flight Director commands for the pilot to follow.
- The Aircraft Navigation System is an integrated package which continuously calculates the aircraft's position. It may include Inertial Reference System (IRS) and GPS inputs in addition to receivers for ground based aids. In the case of an EFIS, the display of these navigational inputs is predicated on the Attitude and Heading Reference System.
- An EFIS or equivalent electromechanical instrumentation of the aircraft's status is provided on either EFIS or conventional instrumentation and is where the effect of FMS aircraft control is principally visible.

SATELLITE NAVIGATION SYSTEMS

Currently, information about an aircraft's position is obtained generally from specialized electronic positioning systems, in particular Satellite Navigation Systems (SNS). The last few years have given rise to many important changes in the operation and exploitation of these systems which transmit a variety of signals on multiple frequencies.

An SNS satellite transmits a signal containing its position and the exact moment of transmission. This message is superimposed with the code which contains the time reference. The synchronization of the signals is obtained by atomic clocks on board each satellite.

The receiver compares the time of arrival with respect to its own clock, with the time of transmission indicated and so measures the distance from the satellite. These measurements are repeated on all visible satellites and make it possible to continuously calculate a position.

Each distance measurement, whatever the system used (low or geostationary constellation or a local beacon) places the receiver on a sphere centered on the transmitter. Using at least three emitters, these spheres have a single point of intersection. As the local clock of the receiver is rarely as accurate, only the time differences are precise. This then requires four beacons or satellites to make a point instead of three (if we know the altitude, three points are enough). As the receivers are mobile, and the measurements are made at different points, and the radio waves have a slightly variable speed depending on the ionospheric layers crossed, the receiver must therefore integrate these various errors, using measurements from various satellites and filters to obtain the most probable point.

For applications requiring absolute point safety (blind landing, anti-collision, etc.) the navigation signals are supplemented by so-called integrity signals which make it possible to eliminate any measurement from a transmitter with a temporary or prolonged fault. This integrity signal is provided by an augmentation system which monitors the health of satellites in real time, such as the European EGNOS satellite system which was developed specifically for civil aviation, but which can, under certain conditions, also render services for maritime or land navigation. The satellite positioning systems with global coverage are:

- GPS for the United States
- GLONASS for Russia
- Galileo for Europe
- Compass or BeiDou-2 and 3 for China

Positioning systems with regional coverage are:

- IRNSS for India
- QZSS for Japan

INERTIAL NAVIGATION SYSTEM (INS)

Inertial navigation relies on knowing your initial position, velocity, and attitude and then measures your attitude rates and various accelerations. The operation of INS is based on Newton's laws of classical mechanics. It is the only form of navigational aid that does not rely on external references.

Given the ability to precisely measure the acceleration of an aircraft, it is then possible to calculate the change in acceleration by performing successive calculations with respect to time. In order to navigate by inertial reference frames, it is necessary to keep track of the direction in which the accelerometers are pointing. These accelerations may be detected using gyroscopic sensors that can determine the orientation of the accelerometers at all times.

Inertial navigation uses gyroscopes and accelerometers to maintain an estimate of the position, velocity, and attitude rates of the aircraft on which the INS is carried. An INS consists of an Inertial Measurement Unit (IMU), instrument support electronics and one or more navigation computers to calculate the position of the host vehicle.

There are many different designs of INS with different characteristics, but they fall generally into two categories which are the gimbaled or stabilized platform and the strap down.

The original applications of INS used stable platform techniques. In such systems, the inertial sensors were mounted on a stable platform and mechanically isolated from the rotational motion of the aircraft. Platform systems are still in use, particularly when requiring very accurate data. Modern systems have removed most of the mechanical complexity of platform systems by having the sensors attached rigidly, or "strapped down", to the structure of the aircraft. The potential benefits of this approach are lower cost, reduced size, and greater reliability. The major disadvantage is a substantial increase in computing complexity.

Advantages of the INS are, the autonomy of the system which does not depend on any external aid or visibility conditions. It is inherently well suited for aircraft navigation, guidance and control. Its IMU measures the variables to be controlled (for example, position, speed and attitude). Another advantage is that it is immune to jamming and inherently stealthy. It neither receives nor emits detectable radiation and does not require an external antenna that can be detectable by radar.

The disadvantages of INS are that navigation errors increase over time. This system is significantly more expensive than that of GPS receivers. The time required to initialize the INS attitude by aligning the gyrocompass is measured in minutes, while for GPS it is measured in seconds. Another downside is the cost of maintenance. Electromechanical avionics systems such as INS tend to have higher failure rates and repair costs than purely electronic avionics systems such as GPS.

The size and weight of an INS is larger than GPS and the power requirements are always higher than of GPS receivers. A synergy with GPS has not only improved the inertial navigation platform, it has made it less expensive. Sensor errors that were unacceptable for stand-alone INS became acceptable for integrated operation and the manufacturing and calibration costs to eliminate these errors are eliminated.

INS also improves GPS performance by carrying the navigation solutions during a loss of GPS signals and so allowing rapid reacquisition.

AIR TRAFFIC CONTROL TRANSPONDER, SECONDARY SURVEILLANCE RADAR

A radar beacon transponder provides positive identification and location of an aircraft on the radar screens of air traffic control. For each aircraft equipped with an altitude encoder, the transponder also provides ATC with the pressure altitude of the aircraft which is displayed with the on-screen blip that represents the aircraft. (***Figure 7-120***)

Radar capabilities at airports vary. Generally, two types are used by ATC. The primary radar transmits directional UHF or SHF radio waves sequentially in all directions. When the radio waves encounter an aircraft, a part of those waves reflects to a ground antenna.

Calculations are made in a receiver to determine the direction and distance of the aircraft from the transmitter. The azimuth direction and distance from the tower are presented giving controllers a two-dimensional fix on the aircraft. (***Figure 7-121***)

Secondary Surveillance Radar

A Secondary Surveillance Radar is also used by ATC to verify the aircraft position and to add the third dimension of altitude to its location. ATC radar transmits coded pulse trains that are received by the transponder onboard the aircraft. Mode 3/A pulses, as they are known, aid in confirming the location of the aircraft. The transponder gets the pressure altitude of the aircraft from an altitude encoder that is electrically connected to the transponder. Typical aircraft transponder antennas are illustrated in ***Figure 7-122.***

This ATC transponder system is known as Air Traffic Control Radar Beacon System (ATCRBS). To increase safety, Mode S altitude response has been developed. With Mode S, each aircraft is pre-assigned a unique identity code that displays along with its pressure

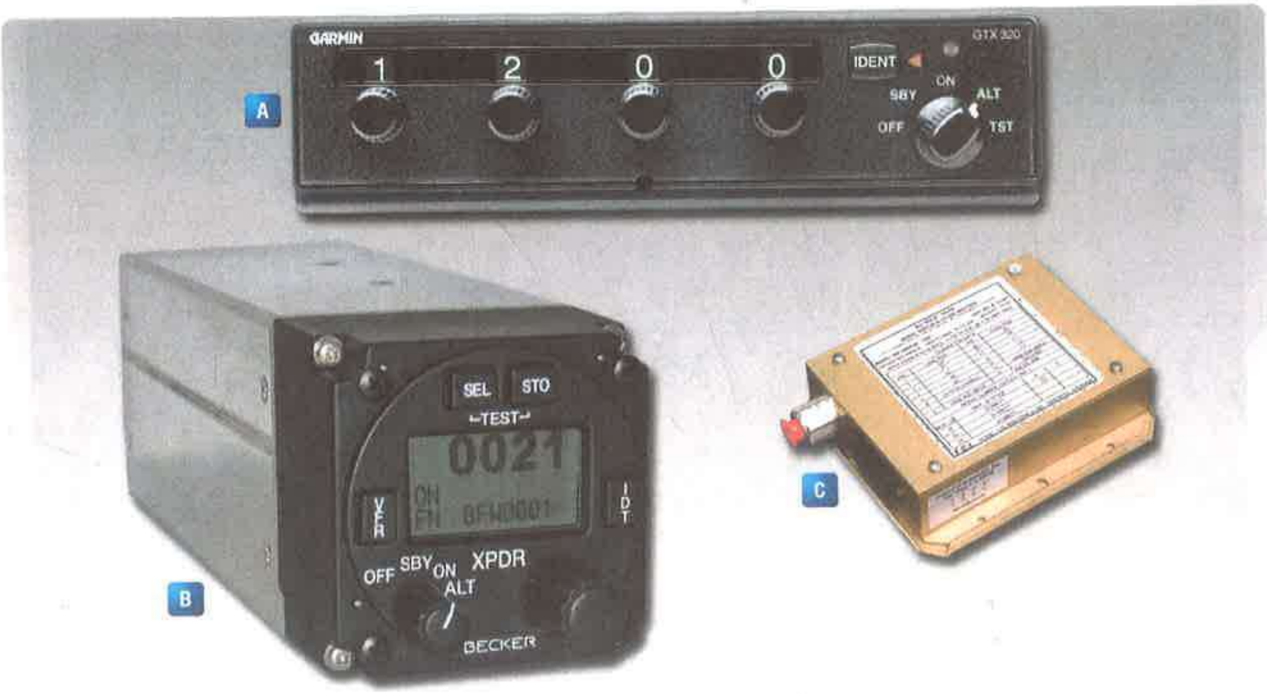


Figure 7-120. A traditional transponder control head.

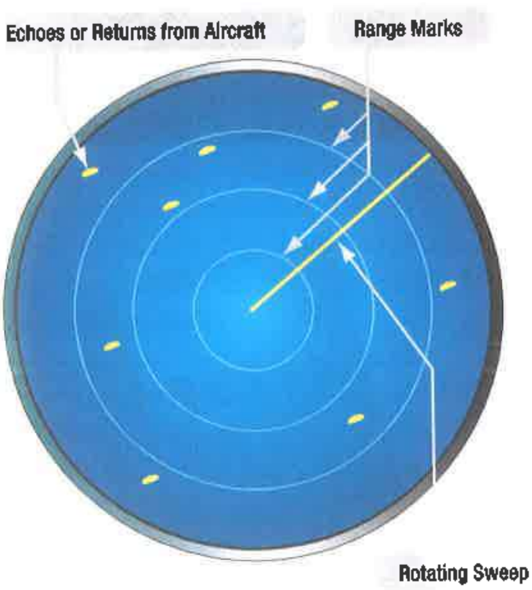


Figure 7-121. A plan position indicator (PPI).

altitude on radar when the transponder responds to SSR interrogation. Since no other aircraft respond with this code, the chance of two pilots selecting the same response code on the transponder is eliminated. A flight data processor assigns the beacon code and searches flight plan data for useful information to be displayed on screen in a data block for each aircraft. (*Figure 7-123*)

Mode S is sometimes referred to as Mode Select. It is a data packet protocol that is also used in onboard collision avoidance systems. Mode S interrogates one aircraft at a time. Transponder workload is reduced by not having to respond to all interrogations at once. Additionally, location information is more accurate with Mode S. A single reply called monopulse, in which the phase of the transponder reply is used to calculate position, is sufficient to locate the aircraft.

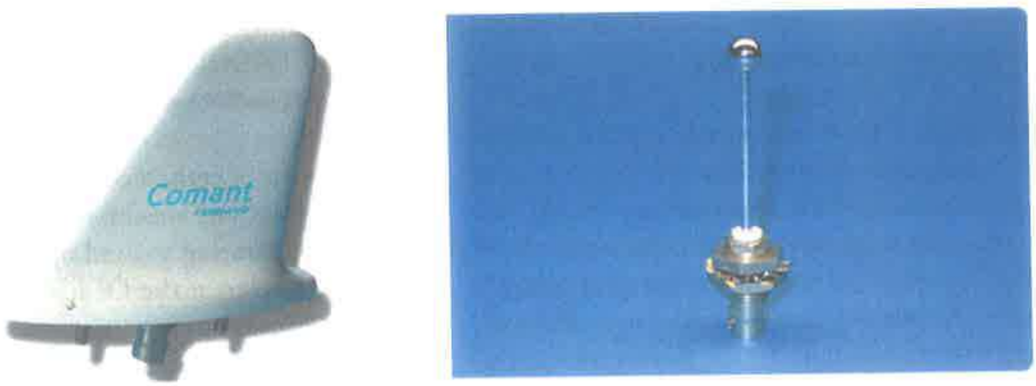


Figure 7-122. Aircraft radar beacon transponder antennas.

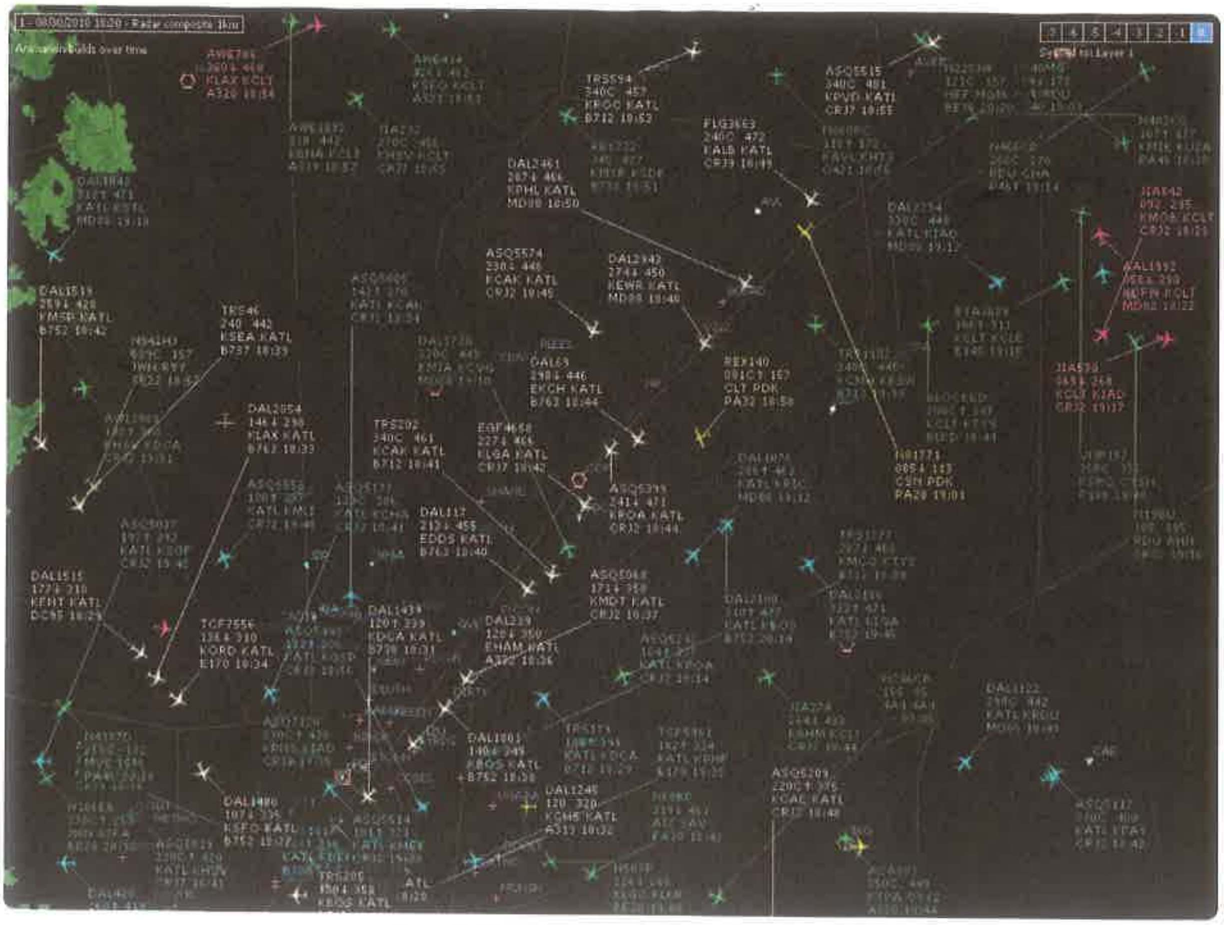


Figure 7-123. Air traffic control radar technology.

TRANSPONDER TESTS AND INSPECTIONS

Because of the danger involved should a transponder malfunction and for example report the wrong altitude information, the functional condition of all transponders is of great concern to aviators. Recent data suggests that such testing may not affect the number of transponder malfunctions. Widespread testing may be more of a problem because, if not performed with strict adherence to manufacturer testing guidelines, transponder radio signals may be transmitted into the atmosphere. Errant signals may cause aircraft to take evasive action or divert the attention of the flight crew from other critical matters. Operating a transponder in a hangar does not eliminate false interrogation and reply.

Technicians should follow the requirements for periodic testing of transponders issued by the NAA of the country of registration of the aircraft. They should also be sure to comply with any airworthiness directive of that country, EASA, and the country of aircraft manufacture. As with many radio electronic devices, specialized test equipment exists to test airworthy operation of a transponder. (*Figure 7-124*)

Figure 7-124. A handheld transponder test unit.

Transmission of certain codes reserved for emergencies or military activity must be avoided. The procedure to select a code during ground operation is to do so with the transponder in the OFF or STANDBY mode to avoid inadvertent transmission.

- Code 0000 is reserved for military use and is a transmittable code.

- Code 1200 is reserved for VFR flight not under ATC direction.
- Code 7500 is used in a hijack situation.
- Code 7600 and 7700 are also reserved for emergency use.

Even the inadvertent transmission of code 1200 could result in evasive action. All signals received from a radar beacon transponder are taken seriously by ATC.

ALTITUDE ENCODERS

Altitude encoders convert the aircraft pressure altitude into a code sent by the transponder to ATC. Increments of 100 feet are usually reported. Encoders have varied over the years. Some are built into the altimeter instrument used in the instrument panel and connected by wires to the transponder. Others are mounted out of sight on an avionics rack or similar out of the way place. These are known as blind encoders. On transport category aircraft, the altitude encoder may be a large black box with a static line connection to an internal aneroid. Modern general aviation encoders are smaller and lightweight, but still often feature an internal aneroid and static line connection. Some encoders use micro-transistors and are completely solid state including the pressure sensing device from which the altitude is derived. No static port connection is required. Data exchange with GPS and other systems is becoming common. (*Figure 7-125*)

When a transponder selector is set on ALT, the digital pulse message sent in response to the secondary surveillance radar interrogation becomes the digital representation of the pressure altitude of the aircraft.

There are 1 280 altitude codes, one for each 100 feet of altitude between 1 200 feet mean sea level (MSL) and 126 700 feet MSL. Each altitude increment is assigned a code. While these would be 1 280 of the same codes used for location and identification (IDENT), the Mode C (or S) interrogation deactivates the 4 096 location codes and causes the encoder to become active.

The correct altitude code is sent to the transponder which then replies to the interrogation. The SSR receiver recognizes this as a response to a Mode C (or S) interrogation and interprets the code as altitude code.

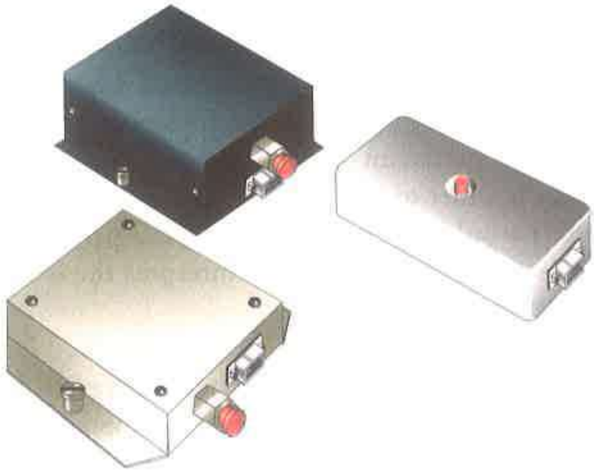


Figure 7-125. Modern altitude encoders.

TRAFFIC ALERT AND COLLISION AVOIDANCE SYSTEM (TCAS)

The ever increasing volume of air traffic has caused a corresponding increase in concern over collisions. Onboard collision avoidance equipment, long a staple in larger aircraft, is now common in general aviation aircraft. New applications of electronic technology combined with lower costs make this possible. TCAS is a transponder based air-to-air traffic monitoring and alerting system. The transponder of an aircraft with TCAS can interrogate the transponders of other aircraft nearby by using SSR technology (Mode C and Mode S). This is done with a 1 030 MHz signal. Interrogated aircraft transponders reply with an encoded 1 090 MHz signal that allows the TCAS computer to display the position and altitude of each aircraft to their respective pilots.

WEATHER AVOIDANCE RADAR

Although more and more aircraft are equipped with one or two airborne weather radars, incursions into very active cumulonimbus clouds still occur, resulting in serious injury or damage to aircraft. In flight, cumulonimbus structures can be a major source of danger due to turbulence and heavy precipitation.

Hail is a major threat because of its physical effect and because weather radars do not indicate the nature of the returns. Only knowing the structure of a cumulonimbus and observing different indices can help. When possible, it is best for pilots to avoid a storm by flying on the windward side of cumulonimbus.

The turbulence associated with a cumulonimbus is not limited to the interior of the cloud. Weather radars cannot detect turbulence in clear air, so it is necessary to take precautionary measures. A cumulonimbus must be cleared by a minimum of 5 000 ft vertically and 20 NM laterally in order to minimize the risk of encountering severe turbulence. Knowledge of the principle of radar is essential to accurately interpret the weather radar display.

Weather radar only detects precipitation droplets. How much it detects depends upon the size, composition and number of droplets. Water particles are five times more reflective than ice particles of the same size. The radar is able to detect the rainfall, wet hail and wet turbulence. However, it is not able to detect clouds, fog or wind (droplets are too small), clear air turbulence, windshear, or lightning. Radar echo returns are proportional to droplet size, and therefore, precipitation intensity. Droplets that are too small (fog droplets) will return no echo, whereas heavy droplets (thunderstorm droplets) will return the majority of radar waves.

It is important to note that reflectivity of particles is not directly proportional to the hazard that may be encountered in a cell. Air can be very humid, when close to the sea for instance. In this case, thermal convection will produce clouds that are full of water. These clouds will have a high reflectivity, but will not necessarily be a high threat. On the other hand, there are equatorial overland regions where converging winds produce large scale uplifts of dry air. The resulting weather cells have much less reflectivity than mid-latitude cells, making them much harder to detect. However turbulence in or above such clouds may have a higher intensity than indicated by the image on the weather radar display.

Because the weather radar display depends on signal returns, heavy precipitation may conceal even stronger weather: The major part of the signal is reflected by the frontal part of the precipitation. The aft part returns weak signals that are displayed as green or black areas. The flight crew may interpret these as a no or small threat area. Modern weather radars are now able to apply a correction to a signal when it is suspected to have been attenuated behind a cloud. This reduces the attenuation phenomenon. However, a black hole behind a red area on a weather radar display should always be considered as a zone that is potentially very active.

Some weather radars are fitted with a Turbulence Display Mode. This function (the TURB function) is based on the Doppler effect and is sensitive to precipitation movement. Like the weather radar, the TURB function needs a minimum amount of precipitation to be effective. To help make safe flight path decisions, and especially when the weather ahead is represented as dense, the turbulence display mode should be used.

Previous generations of weather radars use parabolic antennae and C-band frequencies (4 000-8 000 MHz). Newer weather radars are fitted with flat antennae and use X-Band (8 000 - 12 500 MHz) frequencies, that offer the following advantages:

- Higher pulse energy.
- A narrower beam, that significantly improves the target resolution.
- Higher reflectivity, and therefore a higher total energy return.
- Turbulence and windshear detection.
- Low power consumption.

Consequently, the X band radars are intended to be used as weather avoidance tools and not as tools for penetrating adverse weather.

New generation weather radars are fitted with an auto-tilt function, that will set the radar antenna tilt automatically according to the altitude of the aircraft or an auto-scanning function, that will continuously scan both vertically and horizontally along the aircraft's intended trajectory and will store and display a three-dimensional weather representation.

Care should be taken when using radar on the ground as the radar can be harmful to the human body.

RADIO ALTIMETER (RA)

The radio altimeter is used to measure the absolute height of the airplane above terrain. This is accomplished by transmitting a signal to the ground and processing the received signal as a voltage proportional to the height. This voltage is used to position a pointer over an indicator for a visual indication of altitude.

Using frequency modulated continuous wave techniques, radio altimeters provide a continuous indication of the height above the surface immediately below the aircraft, up to a maximum of 5 000 feet, with 2 500 feet as the most common range. They are particularly suitable for measuring land clearance at low altitude and providing height data to Autoland ILS equipment. The radio altimeter is instantaneous and precise, but gives no indication of high ground ahead. Since the height measurement is absolute, flying over hilly terrain will cause sympathetic variations in the aircraft's height readings on the radio altimeter display.

A radio altimeter consists of a transmitter/receiver, an integrated timing device, a transmitting antenna, a receiving antenna and a display.

The time elapsed, from the transmission of an electromagnetic signal to its reception by the aircraft after reflection on the ground, is measured. As long as the path followed by the wave is vertical, descending and ascending, then the elapsed time measures the height of the aircraft.

All radio altimeter displays have an adjustment control for a decision height, at which a warning will be given. The height can be adjusted by positioning an outer slider against the required height on the scale. The set control will normally serve as a Press-To-Test) function which, when engaged, causes the display to be shown at a predetermined value.

Question: 7-1
From top to bottom, which instruments are included on the panel in the Standard T configuration?

Question: 7-2
Name the three errors which must be corrected when navigating with a compass.

Question: 7-3
Which instrument(s) require a static pressure source, but not a pitot pressure source?

Question: 7-4
Name 2 instrument systems whose components make use of the piezoelectric principle.

Question: 7-5
In what way does a HUMS system aid in vibration control?

Question: 7-6
What two principles are the basis of operation of all synchro systems?

Question: 7-7
What are the two primary signals in an Instrument Landing System (ILS) that allow a pilot to find and approach a runway?

Question: 7-8
What is the primary advantage of an inertial navigation system?

ANSWERS

Answer: 7-1
Top left – airspeed indicator; top center – artificial horizon; top right – altimeter; bottom – gyroscopic heading indicator.

Answer: 7-2
Magnetic deviation, magnetic variation, dip error.

Answer: 7-3
Altimeter and Vertical Speed Indicator, (plus those related sensing components such as an altitude encoder).

Answer: 7-4
Ring Laser Gyros, MEMS systems, (other assorted pressure sensors).

Answer: 7-5
In sends vibration sensor outputs to cockpit displays, maintenance computers, and flight reorders.

Answer: 7-6
All magnetic fields have a defined direction. Magnetic and electromagnetic fields can interact.

Answer: 7-7
Localizer providing horizontal direction, and a Glideslope providing vertical direction.

Answer: 7-8
It does not depend on external signal receptions.

HELICOPTER AERODYNAMICS, STRUCTURES AND SYSTEMS

ELECTRICAL POWER (ATA 24)

SUB-MODULE 08

PART-66 SYLLABUS LEVELS
CERTIFICATION CATEGORY → B1.3 B1.4

Sub-Module 08
ELECTRICAL POWER (ATA 24)
Knowledge Requirements

12.8 - Electrical Power (ATA 24)
Batteries Installation and Operation; 3
DC power generation, AC power generation;
Emergency power generation;
Voltage regulation, Circuit protection.
Power distribution;
Inverters, transformers, rectifiers;
External/Ground power.

BATTERIES: INSTALLATION AND OPERATION

Aircraft batteries are used for many functions including ground power, starting, emergency power improving DC bus stability, and fault clearing. Aircraft batteries are usually identified by the material used for the plates. Most small helicopters use lead-acid batteries. Most medium and large helicopters use Nickel Cadmium (NiCd) batteries. However other lead-acid types of batteries are becoming available such as Valve Regulated Lead-Acid (VRLA) batteries. The battery best suited for a particular application depends on the relative importance of weight, cost, volume, service life, discharge rate, maintenance, and charging rate.

LEAD-ACID BATTERIES

Lead-acid batteries, also known as flooded or wet batteries, are assembled with electrodes (plates) that have been fully charged. The electrolyte is added to the battery when it is placed in service. ***Figure 8-1*** illustrates a lead-acid cell construction with each cell containing positive plates of lead dioxide (PbO_2), negative plates of spongy lead, and an electrolyte (sulfuric acid and water).

Battery life begins when the electrolyte is added. An aircraft battery consists of 6 or 12 lead-acid cells connected in series. The open circuit voltage of the 6 cell battery is approximately 12 volts, and the open circuit voltage of the 12 cell battery is approximately 24 volts.

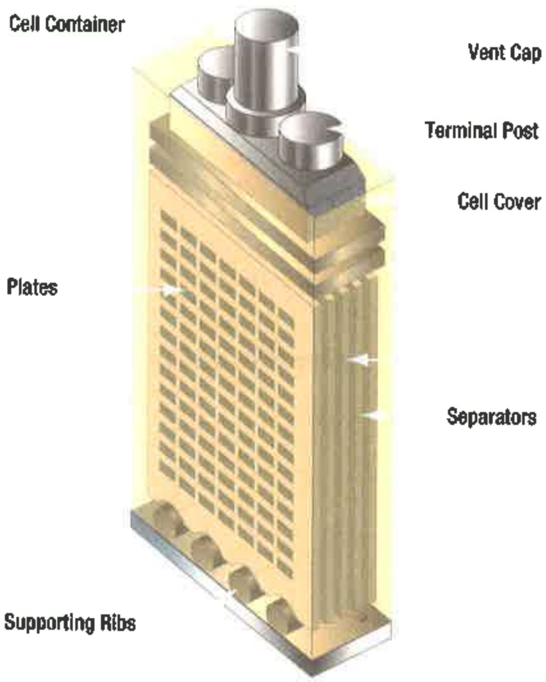


Figure 8-1. Lead-acid cell construction.

Open circuit voltage is the voltage of the battery when it is not connected to a load. When flooded (vented) batteries are charged, the oxygen generated at the positive plates escapes from the cell. Concurrently, at the negative plates, hydrogen is generated from water and escapes from the cell. The overall result is water loss. Therefore, flooded cells require periodic water replenishment. (***Figure 8-2***)

Valve Regulated (Sealed) Lead-Acid Batteries (VRLA)

VRLA batteries contain electrolyte absorbed in glass mat separators with no free electrolyte and are referred to as sealed batteries. (***Figure 8-3***) The electrochemical reactions for VRLA batteries are the same as flooded batteries, except for the gas recombination mechanism

Figure 8-2. Lead acid battery installation.

Figure 8-3. Valve-regulated lead-acid battery.

that is predominant in VRLA batteries. These types of batteries are sometimes authorized as a replacement for NiCd batteries in aircraft.

When VRLA batteries are charged, oxygen combines chemically with the lead at the negative plates in the presence of H_2SO_4 to form lead sulfate and water. This oxygen recombination suppresses the generation of hydrogen at the negative plates. Only a small quantity of water may be lost because of self discharge reactions, however this loss is so small that no provisions are made for replenishment. The battery cells have a pressure relief safety valve that may vent if the battery is overcharged.

NICKEL CADMIUM BATTERIES (NICD)

NiCd batteries consist of a metallic box of stainless steel, plastic coated steel or titanium, containing several individual cells. (***Figure 8-4***) These cells are connected in series to obtain 12 volts or 24 volts using highly conductive nickel copper links. Inside the battery box, the cells are held in place by partitions, liners, spacers, and a cover. The battery has a ventilation system to allow the gases produced during an overcharge condition to escape and to provide cooling during normal operation.

NiCd cells in an aircraft battery are typically vented, having a low pressure release valve that releases any generated oxygen and hydrogen gases when overcharged or discharged rapidly. This also means the battery is not normally damaged by excessive rates of overcharge, discharge, or even negative charge.

Aircraft fitted with NiCd batteries typically have a fault protection system when being charged that monitors for overheating, low temperature (below −40°C), cell imbalance, open, or short circuit. If a fault is found, it turns off and sends a fault signal to the Electrical Load Management System. A NiCd battery can perform at its rated capacity when the ambient temperature of the battery is in the range of approximately 15 – 32°C. An increase or decrease in temperature from this range results in reduced capacity. NiCd batteries have a ventilation system to control the temperature of the battery. A combination of high battery temperature (above 71°C) and overcharging can lead to a condition called thermal runaway. (***Figure 8-5***)

Thermal runaway can result in a NiCd chemical fire and/or explosion of the battery being overcharged by a constant voltage source due to cyclical ever increasing temperature and charging current. If one or more cells are shorted, an existing high temperature and low charge can produce the following sequence of events:

- Excessive current.
- Increase in temperature.
- Decrease in cell(s) resistance.
- Further increase in current.
- Further increase in temperature.

LITHIUM-ION BATTERIES

The most recent type of battery to be certified in aircraft is the lithium-ion battery. These batteries have greater capacity and weigh less than NiCd or lead acid types. They have no memory such as NiCd batteries have and discharge less than half as fast when not being used. The anode is a graphite layered structure capable of storing and releasing lithium ions. Cathode materials vary. Some certified batteries are made of lithium cobalt oxide ($LiCoO_2$) with an aluminum core.

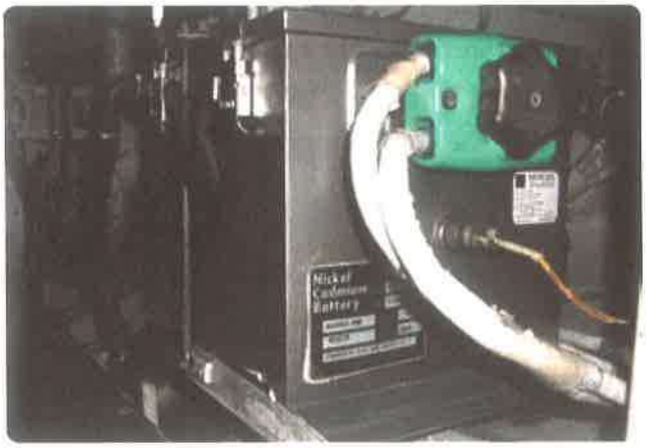


Figure 8-4. NiCd battery installation.

Figure 8-5. Thermal runaway damage.

A water free electrolyte composed of organic carbonates resides between the anode and the cathode. It functions as a transport medium for the lithium ions moving from the anode to the cathode during discharge, and from the cathode to the anode during charging. A separator porous to the Li+ ions is placed between the anode and the cathode in each cell. Typical cell output voltage is between 3 and 4.2 volts, depending primarily on the materials used to construct the cathode. Eight cells connected in series are typical as shown in ***Figure 8-6***.

Lithium-ion aircraft batteries require built-in safety devices to prevent overheating and thermal runaway. They are constructed with a wide variety of materials that result in a compromise between capacity, longevity, environmental endurance, current loading, specific energy, size, and weight, etc. Additional current monitoring and other safety and alerting devices are included to warn flight crews of battery status and malfunction. Technicians must follow all manufacturer instructions when maintaining lithium-ion batteries.

BATTERY SPECIFICATIONS

Capacity

Battery capacity is measured in ampere hours delivered at a specific discharge rate to a specified cut-off voltage at room temperature. The cut-off voltage is 1.0 volts per cell. A battery's available capacity depends upon several factors including such items as:

- Cell design: Cell geometry, plate thickness, hardware, and terminal design govern performance under specific usage conditions of temperature, discharge rate, etc.
- Discharge rate: High current rates yield less capacity than low rates.
- Temperature: Capacity and voltage levels decrease as battery temperature moves away from the 16°C to 32°C range and towards the high and low extremes.
- Charge rate: Higher charge rates generally yield greater capacity.

Battery Ratings by Specification

The one hour rate is the rate of discharge a battery can endure for 1 hour with the battery voltage above 1.67 volts per cell, 20 volts for a 24 volt lead-acid battery, or 10 volts for a 12 volt lead acid battery. The one hour capacity, measured in ampere hours (Ah) is the product of the discharge rate and time (in hours) to the specified end voltage. The emergency rate is the total essential load, measured in amperes, required to support the essential bus for 30 minutes at the above rates.

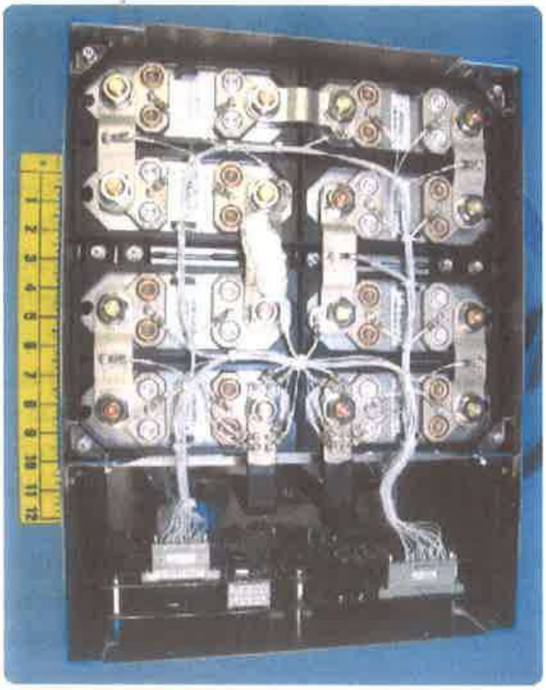


Figure 8-6. Cells and wiring in a lithium-ion aircraft battery.

BATTERY INSTALLATION

Batteries are installed in areas where adequate heat dissipation is possible and where ventilation of gases is effective. Some systems use an acid trap. These traps are bottles inserted between the battery and the fuselage which contains a neutralizing agent to prevent acid being sprayed on the fuselage. Batteries are normally clamped to a tray which is secured to the helicopter structure. A typical helicopter battery weighs weigh nearly 45 kg (100 lbs). Attached fittings on a battery facilitate the use of lifting equipment during removal and installation.

When installing batteries, exercise care to prevent inadvertent shorting of the battery terminals. Serious damage to the aircraft structure can be sustained by the resultant high discharge of electrical energy. This condition can normally be avoided by insulating the terminal posts during installation. Remove the grounding lead first for battery removal, then the positive lead. Then reconnect the grounding lead last to minimize the risk of shorting the hot terminal.

Most aircraft use 28V DC batteries but a configuration exists where two 14V batteries are connected in series to be at 28V for bus use. Often, aircraft batteries have two connectors. The larger connector is a terminal block which connects the high power output of the battery into the bus system. The smaller electrical connector is for battery control and status signals. Temperature sensors and overheat sensing are common. A cooling fan may be included in the installation. (***Figure 8-7***)

BATTERY CHARGING

Operation of aircraft batteries beyond their ambient temperature or charging voltage limits can result in excessive cell temperatures leading to electrolyte boiling, rapid deterioration of the cells and battery failure. The relationship between maximum charging voltage and the number of cells in a battery is also significant. This determines the rate at which energy is absorbed as heat within the battery. For lead-acid batteries the voltage per cell must not exceed 2.35 volts. In the case of NiCd batteries, the charging limit varies with design and construction. Values of 1.4 and 1.5 volts per cell are generally used. In all cases, follow the recommendations of the battery manufacturer.

Constant Voltage Charging

The battery charging system within a helicopter is of the constant voltage type. An engine driven generator, capable of supplying the required voltage is connected through the electrical system directly to the battery. A battery switch is incorporated in the system so the battery may be disconnected when the aircraft is not in operation.

The voltage of the generator is controlled by means of a voltage regulator connected in the field circuit of the generator. For a 120 volt system, the voltage of the generator is adjusted to approximately 14.25. On a 24 volt system, the adjustment is between 28 – 28.5 volts. When these conditions exist, the initial charging

Figure 8-7. Nickel Cadmium aircraft battery.

current through the battery is high. As the state of charge increases, the battery voltage also increases causing the current to taper down. When the battery is fully charged, its voltage is almost equal to the generator voltage and very little current flows into the battery. When the charging current is low, the battery may remain connected to the generator without damage.

When using a constant voltage system, a voltage regulator that automatically maintains a constant voltage is incorporated into the system. A higher capacity battery (example 42 Ah) has a lower resistance than a lower capacity battery (example 33 Ah). Hence a high capacity battery draws a higher charging current than a low capacity battery when both are in the same state of charge and when charging voltages are equal. The constant voltage method is the preferred charging method for lead-acid batteries.

Constant Current Charging

Constant current charging is the most convenient method when outside the aircraft because several batteries of varying voltage may be charged at once on the same system. A constant current system consists of a rectifier to change the AC supply to DC. A transformer is used to reduce the available 110 or 220 volt AC supply to the desired level before it is passed through the rectifier. With a constant current system multiple batteries may be connected in series, provided the charging current is kept at such a level that the battery does not overheat or gas excessively.

The constant current method is preferred for NiCd batteries. Typically a NiCd battery is charged at a current rate of 1CA until all the cells have reached at least 1.55 volts. Another charge cycle follows at 0.1CA, again until all the cells have reached 1.55V. The charge is finished with an overcharge, typically for not less than 4 hours at a rate of 0.1CA. The purpose of the overcharge is to expel as much of the gases collected on the electrodes, (hydrogen on the anodes and oxygen on the cathode). During the overcharge, the cell voltages go beyond 1.6V and then slowly start to drop. No cell should rise above 1.71V (dry cell) or drop below 1.55V.

Charging is done with the vent caps loosened or open. A stuck vent might increase the pressure in the cell. It also allows for refilling of water to correct levels before the end of the top-up charge while the charge current is still on. However, cells should be closed again as soon as the vents have been cleaned and checked since carbon dioxide dissolved from outside air carbonates the cells and ages the battery.

BATTERY MAINTENANCE

Battery inspection and maintenance procedures vary with the type of chemical technology and physical construction. Always follow the battery manufacturer's approved procedures. Battery performance at any time in a given application depends upon the battery's age, state of health, state of charge, and its mechanical integrity.

- To determine the life and age of a better, record its install date on the battery. During normal maintenance, battery age must also be documented in the aircraft maintenance log or the shop maintenance log.
- Lead-acid battery state of health may be determined by duration of service intervals (in the case of vented batteries), by environmental factors such as heat or cold, and by observed electrolyte leakage (as evidenced by corrosion or wiring and connectors or accumulation of powdered salts). If the battery needs to be refilled often, with no evidence of leakage, this may indicate a poor state.
- Use a hydrometer to determine the specific gravity of the lead-acid electrolyte, which is the weight of the electrolyte compared to the weight of pure water. Take care to ensure the electrolyte is returned to the cell from which it was extracted. When a specific gravity difference of 0.050 or more exists between cells, the battery is approaching the end of its useful life and replacement should be considered. Electrolyte level may be adjusted by the addition of distilled water.
- Battery state of charge is determined by the cumulative effect of charging and discharging. In a normal charging system, the aircraft generator or alternator restores a battery to full charge during a flight of 60-90 minutes.
- Mechanical integrity involves the absence of physical damage, as well as assurance that the battery hardware is correctly installed and the battery is properly connected. Battery and battery compartment venting system tubes provide a means of avoiding the buildup of explosive gases, and should be checked periodically to ensure they are connected and oriented in accordance with the maintenance manual's installation procedures.

Battery Overheating

Various factors that may lead to thermal runaway can cause overheating of a battery, such as:

- Voltage regulator incorrectly adjusted, or battery charger output too high.
- Frequent engine or APU starts.
- Ground Power units with poor voltage regulation.
- Poor ventilation of battery compartment during high ambient temperatures.
- High initial charging currents imposed on a hot battery.
- Faulty temperature sensor or wiring.

Battery Freezing

Discharged lead-acid batteries exposed to cold temperatures are subject to plate damage due to freezing of the electrolyte. To prevent freezing, maintain each cell's specific gravity at 1.275 or, for sealed batteries, check open circuit voltage. NiCd battery electrolyte is not as susceptible to freezing because no appreciable chemical change takes place between the charged and discharged states. However, electrolyte freezes at approximately -60°C.

Electrolyte Spillage

In the event of electrolyte spillage, several actions must be immediately taken including:

- Remove the battery for access if the spillage is in the battery area.
- Remove all accessible pools of electrolyte by means of clean rags.
- Remove all contaminated components and check adjacent structure for evidence of corrosion.
- Check all cables for contamination and replace if necessary.
- Dry the affected area.

Storage of Batteries

Separate facilities for storing and/or servicing flooded lead-acid and NiCd batteries must be maintained. Introduction of acid electrolyte into alkaline electrolyte causes permanent damage to vented NiCd batteries and vice versa. Batteries that are sealed can be charged and their capacity checked in the same area.

BATTERY VENTILATION SYSTEMS

Aircraft are equipped with battery ventilating systems which remove gasses and acid fumes from the battery in order to reduce fire hazards and to eliminate damage to the airframe parts. Air is carried from a scoop outside the aircraft through a vent tube to the interior of the battery case. After passing over the battery, air, gasses, and acid fumes are carried through another tube to the battery sump. This sump is a glass or plastic jar. In the jar is a felt pad about 3 cm thick and saturated with a 5% solution of bicarbonate of soda and water. The tube extends into the jar to within 1cm of the felt pad. An overboard discharge tube leads from the top of the sump jar to a point outside. The outlet for this tube is designed so there is negative pressure on the tube during flight. This helps ensure a continuous flow of air across the top of the battery through the sump and outside.

Figure 8-8 shows the construction of the vent plug. In level flight, the lead weight permits venting of gases through a small hole. In inverted flight, this hole is covered by the lead weight.

DC POWER GENERATION; AC POWER GENERATION

Most modern aircraft use AC generators for the primary source of power. However, there are still aircraft that may use DC generators. Basics on DC and AC generators have been studied in Module 3.

DC POWER GENERATION

When present, DC generators normally have an output of 14 or 28V DC. The output is regulated by controlling field current strength. However, it is more common on all modern aircraft to instead produce DC from AC generator power. Transformer rectifiers are then used to convert the AC output to the DC voltage as required to power various buses.

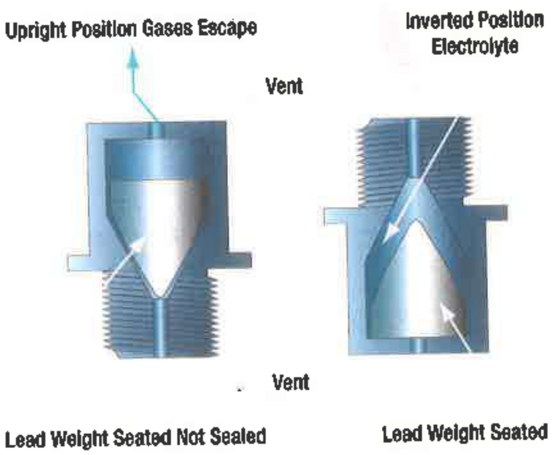


Figure 8-8. Non-spill battery vent plug.

An engine driven DC generator requires a control circuit to ensure the generator maintains the correct voltage and current for the electrical needs of the aircraft. Since engines operate at a variety of speeds and with a variety of electrical demands, generators must be regulated by some control system to keep the output within limits for all flight variables. These systems are often referred to as voltage regulators or Generator Control Units (GCU).

Generator output can be adjusted through control of its magnetic field strength which has a direct effect on generator output. Increasing field current means an increased generator output and vice versa. ***Figure 8-9*** shows a simple generator control used to adjust field current.

There are two basic types of generator controls: electromechanical and solid state (transistorized). Electromechanical controls are found on older aircraft and require regular inspection and maintenance. Solid state systems are considered to have better reliability and more accurate generator output control.

AC POWER GENERATION

An AC power system is the primary source of electrical power on most helicopters. These extremely reliable power distribution systems are computer controlled. Multiple AC generators (also known as alternators) and a variety of distribution busses are used for redundancy. Most aircraft contains two or more main AC generators driven by the engine(s), as well as a backup generator. Backup DC systems may also be employed to supply emergency power in case of multiple failures.

A variety of electrical systems are required to fly an aircraft, such as flight control systems, electronic engine controls and avionics systems. The output capacity of

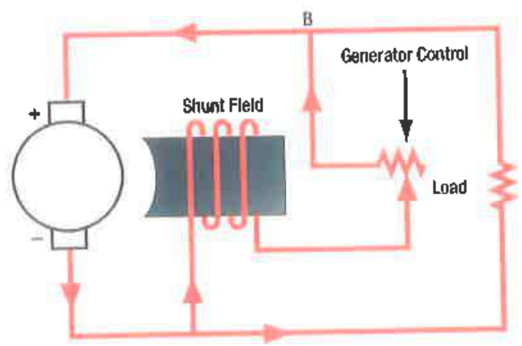


Figure 8-9. Regulation of generator voltage by field rheostat.

one engine driven AC generator can typically power all necessary systems. A second engine driven generator is often operated during flight to share the electrical loads and provide redundancy.

The complexity of multiple generators and a variety of distribution busses requires several control units to maintain a constant supply of safe power. The AC electrical system must maintain a constant output of 115 to 120 volts at a frequency of 400 Hz (±10 percent). The system must ensure power limits are not exceeded. AC generators are connected to the appropriate distribution busses at the appropriate time, and generators are in phase when needed. There is also the need to monitor and control any external power supplied to the aircraft, as well as control of all DC electrical power.

Two electronic Line Replaceable Units (LRUs) are used to control the electrical power on a large aircraft. The GCU is used for control of AC generator functions, such as voltage regulation and frequency control. The Bus Power Control Unit (BPCU) controls the distribution of power between the various distribution busses on the aircraft. The GCU and BPCU work together to control power, detect faults, take corrective actions, and report any defect to the pilots and to the aircraft maintenance computer. There is generally one GCU for each generator and at least one BPCU to control bus connections.

When the pilot calls for generator power by activating the generator control switch on the flight deck, the GCU monitors and ensures correct operation. If all systems are operating within limits, the GCU energizes the appropriate circuits and provides voltage regulation for the system. The GCU also monitors AC output to ensure a constant 400 Hz frequency. If the generator output is within limits, the GCU then connects the electrical power to the main bus through an electric contactor (solenoid). These contactors are often called generator breakers since they break (open) or make (close) the main generator circuit.

After generator power is available, the BPCU activates various contactors to distribute the power. The BPCU monitors the complete electrical system and communicates with the GCU to ensure proper operation. The BPCU employs remote sensors known as Current Transformers (CT) to monitor the system. (*Figure 8-10*)

A current transformer is an inductive unit that surrounds the main power cables of the electrical distribution system. As AC power flows through the cables, the CT receives an induced voltage related to the current flowing through the cable. The CT connects to the BPCU, which allows accurate current monitoring of the system. An aircraft employs several CTs throughout the system.

The BPCU is a dedicated computer that controls the electrical connections between the various distribution busses using Bus Tie Breakers (BTB) for connection of various circuits. These BTBs open/close the connections between the busses as needed for system operation as called for by the flight crew and the BPCU. There are three common types of distribution bus systems found on large and complex aircraft: split bus, parallel bus, and split parallel bus.

AC alternators produce a three phase AC output. For each revolution of the alternator, the unit produces three separate voltages. The sine waves for these voltages are separated by 120°. (*Figure 8-11*) This wave pattern is like those produced internally by a DC generator; however, in this case, the AC alternator does not rectify the voltage and the output is AC. The modern alternator does not utilize brushes or slip rings and is often referred to as a brushless alternator. This brushless design is extremely reliable and requires little maintenance. In

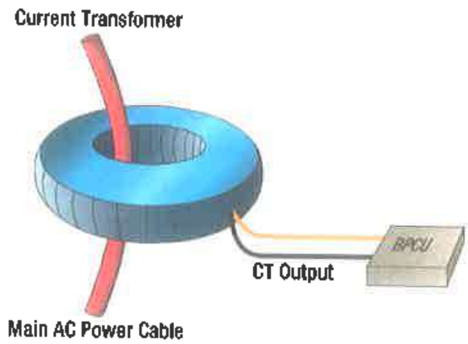


Figure 8-10. Current transformer.

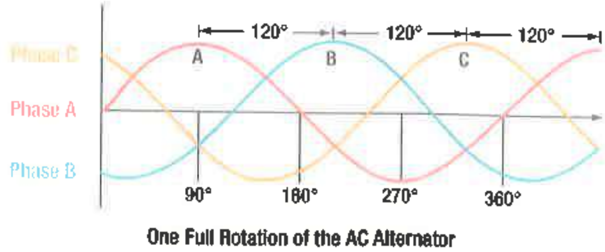


Figure 8-11. AC alternator sine waves.

a brushless alternator, energy to or from the alternator rotor is transferred using magnetic energy. In other words, energy from the stator to the rotor is transferred using magnetic flux and the process of electromagnetic induction. A large aircraft AC alternator is shown in *Figure 8-12*.

As seen in *Figure 8-13*, the brushless alternator contains three generators: the exciter generator (armature and permanent magnet field), the pilot exciter generator (armature and fields windings), and the main AC alternator (armature winding and field windings).

The need for brushes is eliminated by using a combination of these three distinct generators. The exciter is a small AC generator with a stationary field made of a permanent magnet and two electromagnets. Its armature is three phase and mounted on the rotor shaft. The exciter armature output is rectified and sent to the pilot exciter field and the main generator field.

The pilot exciter is mounted on the rotor shaft and is connected in series with the main generator field. The AC output of the pilot exciter armature is supplied to the generator control circuitry where it is rectified, regulated, and then sent to the exciter field windings. The current sent to the exciter field provides the voltage regulation for the main AC alternator. If more alternator output is needed, then more current is sent to the exciter field and vice versa.

In short, the exciter permanent magnet and armature starts the generation process, and the output of the exciter armature is rectified and sent to the pilot exciter field. The pilot exciter field creates a magnetic field and induces power in the pilot exciter armature through electromagnetic induction. The output of the pilot exciter armature is sent to the main alternator control unit and then sent back to the exciter field. As the rotor continues to turn, the main alternator field generates power into the main alternator armature, also using electromagnetic induction. The output of the main armature is three phase AC and used to power the various electrical loads.

Figure 8-12. Large aircraft AC alternator.

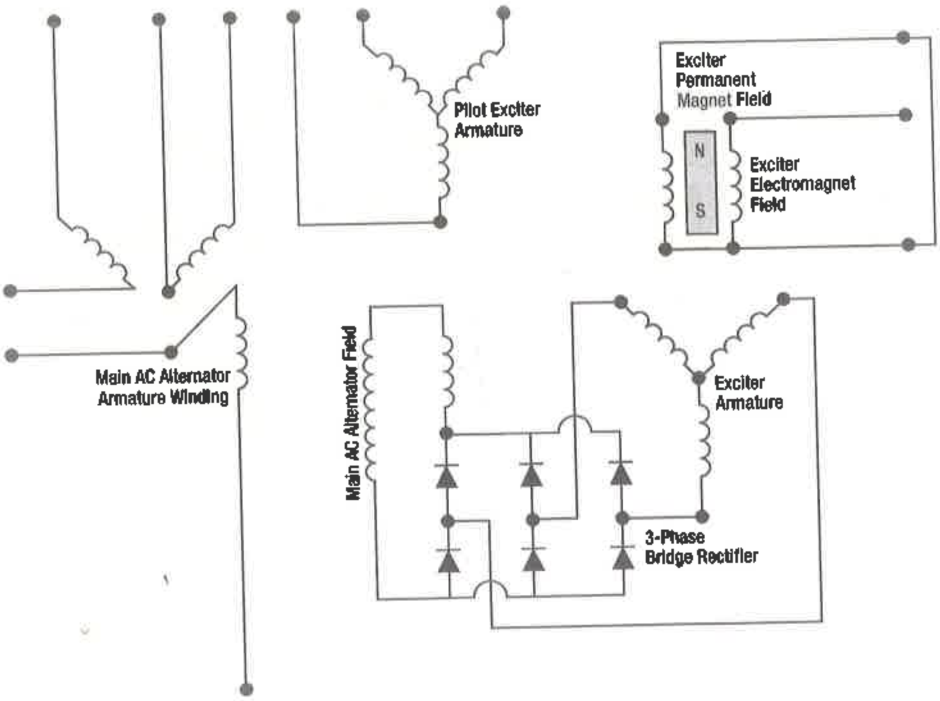


Figure 8-13. Schematic of an AC alternator.

ALTERNATOR DRIVE

The unit shown in ***Figure 8-14*** contains an alternator assembly combined with an automatic drive mechanism. This Constant Speed Drive (CSD) controls the alternator rotational speed which allows the alternator to maintain a constant 400 Hz AC output. If the frequency strays more than 10 percent from this value, the electrical systems do not operate correctly. Thus, the CSD ensures that the alternator rotates at the correct speed to ensure a 400 Hz frequency. The CSD can be an independent unit or mounted within the alternator housing. When the CSD and the alternator are contained within one unit, the assembly is known as an Integrated Drive Generator (IDG).

The CSD is a hydraulic unit like a transmission found in an automobile allowing the engine to change RPM while the speed of the car remains constant. This is the

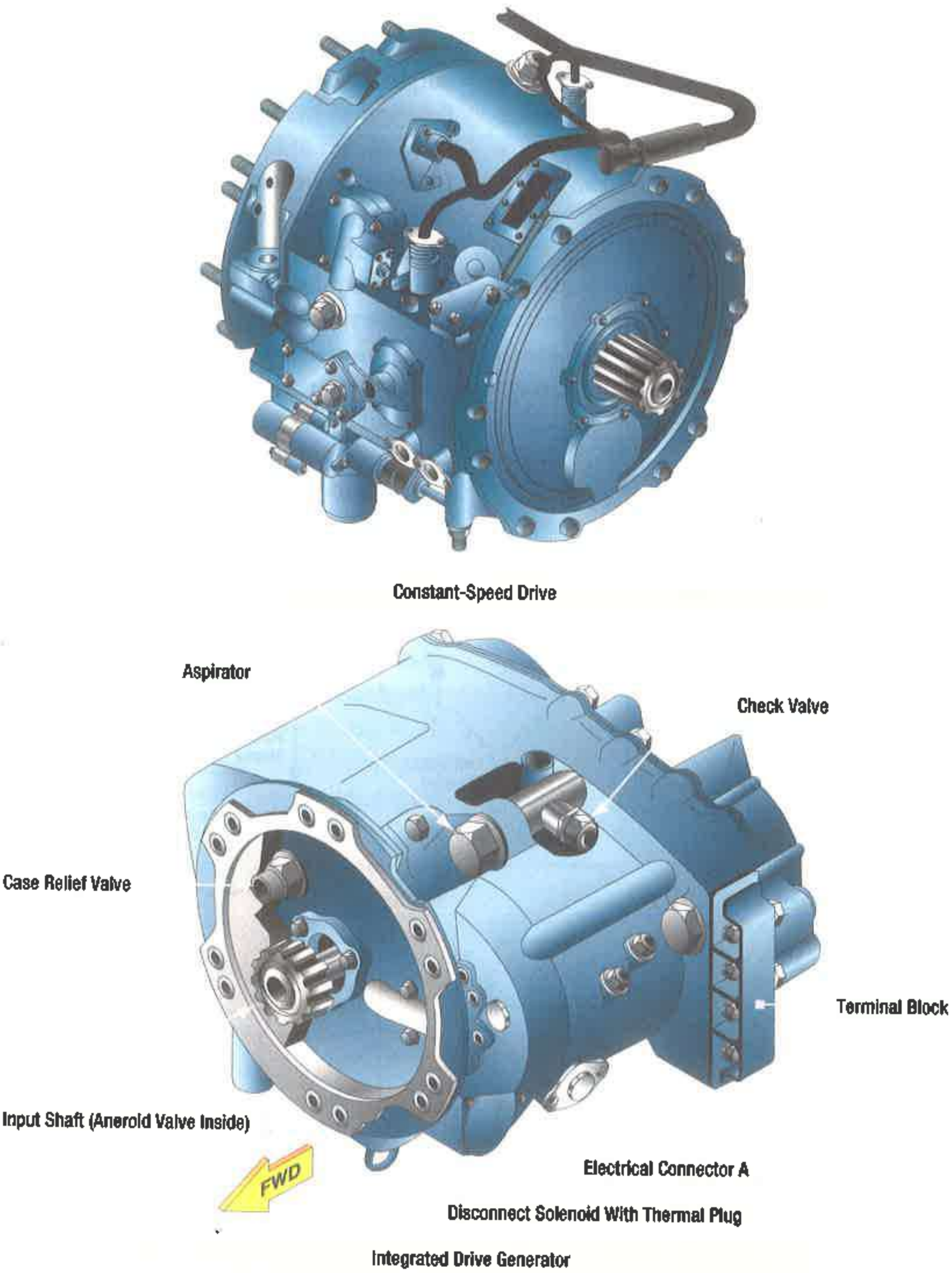


Figure 8-14. Constant-speed drive (top) and integrated drive generator (bottom).

same process that occurs for an aircraft alternator. If the aircraft engine changes speed, the alternator speed and thus the frequency of its output remains constant. A typical hydraulic type of drive is shown in ***Figure 8-15***. This unit can be controlled either electrically or mechanically. In most aircraft, it is electronic.

The hydraulic transmission is mounted between the alternator and the engine. Hydraulic or engine oil is used to operate the hydraulic transmission which creates a constant output speed to drive the alternator. The input drive shaft is powered by the engine gear case. The output drive shaft engages the drive shaft of the alternator. The speed control unit is made up of a wobble plate that adjusts hydraulic pressure to control its speed.

Figure 8-16 shows a typical electrical circuit used to control alternator speed through the circuit controls of a typical CSD. As shown, the alternator input speed is monitored by a tachometer generator. The tach generator signal is rectified and sent to the valve assembly. The valve assembly contains three electromagnetic coils

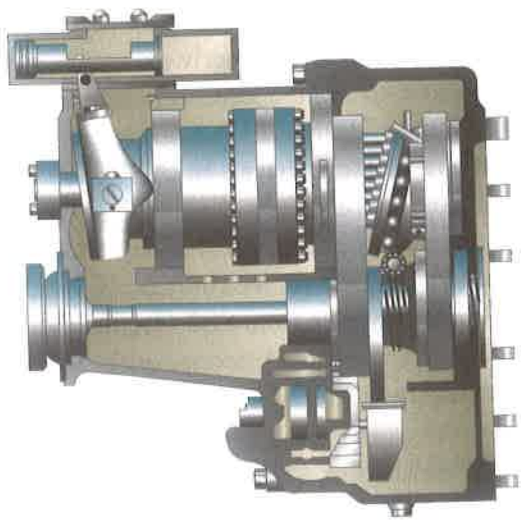


Figure 8-15. A hydraulic constant speed drive for an AC alternator.

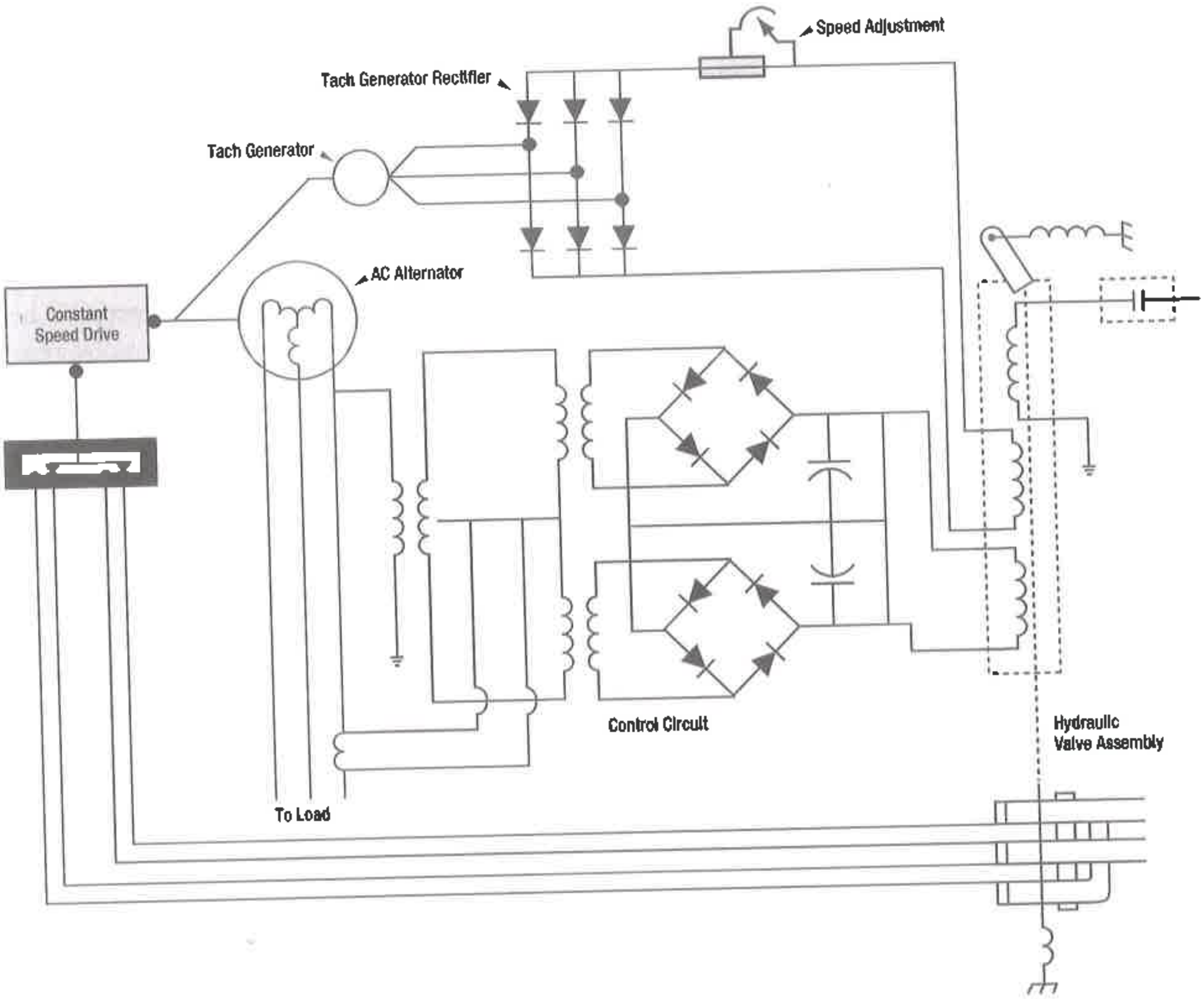


Figure 8-16. Speed control circuit.

ELECTRICAL POWER

that operate the valve. The AC alternator output is sent through a control circuit that also feeds the hydraulic valve assembly.

It should be noted that an AC alternator also produces a constant 400 Hz when directly driven by an engine that rotates at a constant speed. On many aircraft, the Auxiliary Power Unit (APU) operates at a constant RPM. AC alternators driven by APUs are typically driven directly by the engine, and no CSD is required. For these units, the alternator output frequency is monitored by the APU engine controls. If the alternator output frequency varies from 400 Hz, the APU speed control adjusts the RPM accordingly to keep the alternator output within limits.

AC ALTERNATOR CONTROL SYSTEMS

Aircraft that employ AC alternators use several computerized control units, located in the aircraft equipment bay for the regulation of AC power throughout the aircraft. (*Figure 8-17*)

Figure 8-17. Line replaceable units in an equipment rack.

If the frequency of an alternator varies from 400 Hz, or if two or more alternators connected to the same bus are out of phase, damage occurs to the system. Alternator control units contain circuitry to regulates both voltage and frequency. These units monitor a variety of factors to detect any system failures and take protective measures to ensure the integrity of the system.

A GCU ensures the AC alternator maintains a constant voltage between 115 to 120 volts and to ensure the maximum output of the alternator is never exceeded. It also monitors AC frequency and ensures the output if the alternator remains at 400 Hz. The basic method of voltage regulation is similar to that found in all alternator systems; the output of the alternator is controlled by changing the strength of a magnetic

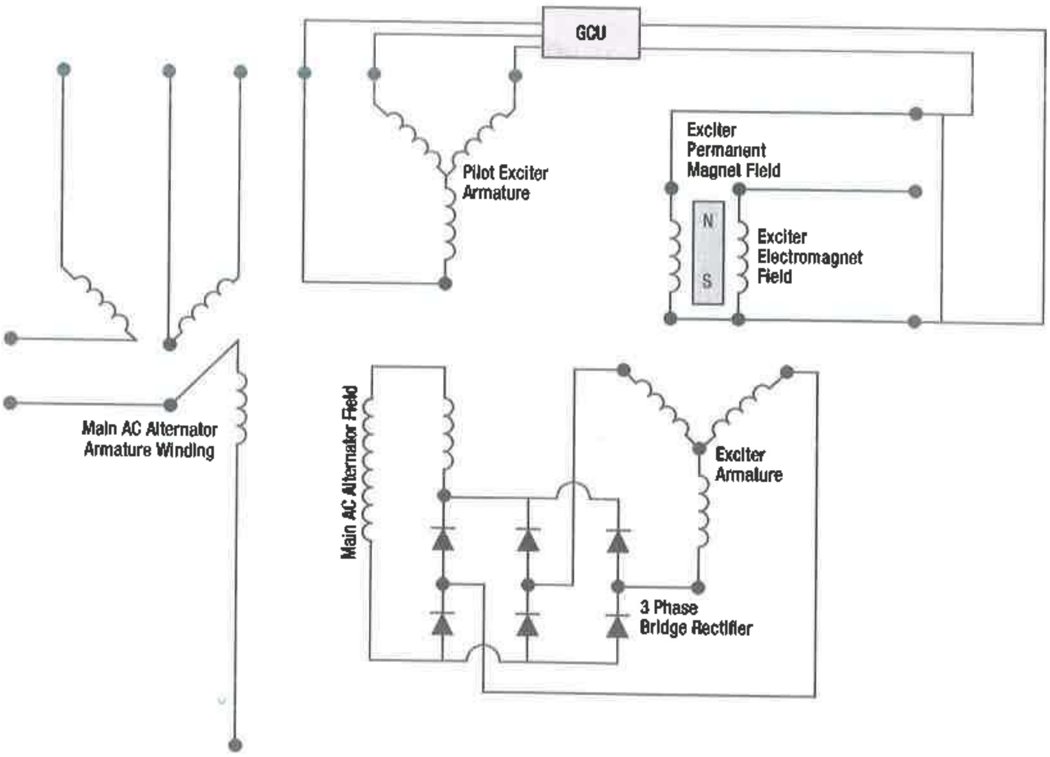


Figure 8-18. Schematic GCU control of the exciter field magnetism.

field. As shown in ***Figure 8-18***, the GCU controls the exciter field magnetism within the brushless alternator to control alternator output voltage.

When the pilot selects the operation of an alternator, the GCU monitors output to ensure voltage and frequency are within limits. If the GCU is satisfied with the alternator output, it sends a signal to an electric contactor that connects the alternator to the appropriate AC distribution bus. The contactor, often call the generator breaker, is an electromagnetic solenoid that controls a set of large contact points. The large points are necessary to handle the large amounts of current produced by most AC alternators. This same contactor is activated in the event the GCU detects a fault in the alternator output; however, in this case the contactor would disconnect the alternator from the bus.

EMERGENCY POWER GENERATION

Modern aircraft have buses and loads divided so that the failure of a single generator is controlled by isolating the failed unit and allows continued use of the other main generator or back-up generators to power the buses of the failed unit. Automatic monitoring and switching of power sources are normal.

When both the main AC generators fail simultaneously, backup or standby generators may still be used to power AC and DC buses. The failure of the main and backup generators is rare. It presents the crew with a situation in which emergency power must be used. On most large turbine powered helicopters, a standby power bus is used for emergency power when the main sources of power fail. The standby bus is usually a hot bus directly connected to the main aircraft battery. It is powered by the APU generator, a ram air turbine generator or the aircraft battery.

VOLTAGE REGULATION, CIRCUIT PROTECTION

Voltage regulators are used in helicopter primary power systems to maintain the system voltage within the limits necessary for the correct operation of the associated electrical devices. In addition, they are in some cases used to control the sharing of loads between generators operating in parallel. Voltage regulation is accomplished by varying the field strength of the AC generator exciter field in order to keep the output voltage constant under varying speed and load conditions. Most generator control systems perform several functions related to the regulation, sensing, and protection of the generation system.

VOLTAGE REGULATION

The most basic of the GCU functions is voltage regulation. Regulation of any kind requires the regulation unit to take a sample of a generator output and compare that sample to a known reference. If the generator output voltage falls outside the set limits, the regulation unit must provide an adjustment to its field current. Adjusting field current controls generator output.

PARALLEL GENERATOR OPERATIONS

On multi engine aircraft, a paralleling feature must be employed to ensure all generators operate within limits. In general, paralleling systems compare the voltages between two or more generators and adjust the voltage regulation circuit accordingly.

DIFFERENTIAL VOLTAGE

This function of a control system is designed to ensure all generator voltage values are within a close tolerance before being connected to the load bus. If the output is not within the specified tolerance, then the generator contactor is not allowed to connect the generator to the load bus.

GENERATOR CONTROLS FOR HIGH OUTPUT GENERATORS

This unit is referred to as a starter-generator. A starter-generator has the advantage of combining two units into one housing, saving space and weight. Since the starter-generator performs two tasks, engine starting and generation of electrical power, the control system is relatively complex.

A simple explanation of a starter-generator shows that the unit contains two sets of field windings. One field is used to start the engine and the other used for the generation of electrical power. (***Figure 8-19***) During the start function, the GCU must energize the series field so the armature causes the unit to act like a motor.

During the generating mode, the GCU must disconnect the series field, energize the parallel field, and control the current produced by the armature. At that time, the starter-generator acts like a generator. Of course, the

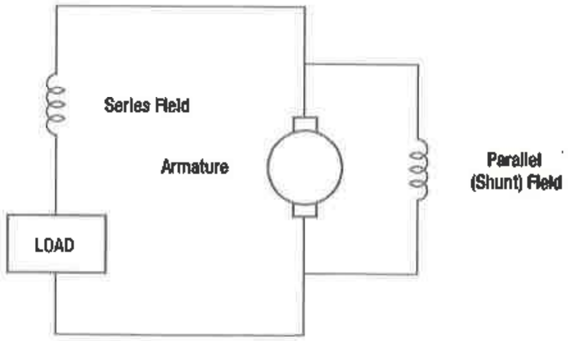


Figure 8-19. Starter-generator.

GCU must perform all the functions described earlier to control voltage and protect the system. These functions include voltage regulation, reverse current sensing, differential voltage, over excitation protection, over voltage protection, and parallel generator operations. A typical GCU is shown in ***Figure 8-20.***

In general, modern GCUs for high output generators employ solid state electronic circuits to sense the operations of the generator or starter generator. The circuitry then controls a series of relays and/or solenoids to connect and disconnect the unit to various distribution busses. One unit found in almost all voltage regulation circuitry is the Zener diode. The Zener diode is a voltage sensitive device that is used to monitor system voltage. The Zener diode, connected with the GCU circuitry controls the field current which in turn controls the generator output.

Figure 8-20. Generator Control Unit (GCU).

OTHER VOLTAGE REGULATION

Small and older aircraft that use low output DC generators must also have a means of voltage regulation. The typical device for this is a control unit or voltage regulator that modifies current to the generator field to control its output power. As flight variables and electrical loads change, the voltage regulator monitors the electrical system and makes the appropriate adjustments to ensure proper system voltage and current. They are typically electromechanical devices. The two most common are the carbon pile regulator and the three unit regulator. Each of these control field current using a type of variable resistor. A simplified generator control circuit is shown in ***Figure 8-21.***

Carbon Pile Regulators

Carbon pile regulators control DC output by sending the field current through a stack of carbon disks (the carbon pile) which are in series with the generator field. If the resistance of the disks increases, the field current decreases and the generator output goes down. If the resistance decreases, the field current increases and output goes up.

As seen in ***Figure 8-22***, a voltage coil is installed in parallel with the generator output leads. The voltage coil acts like an electromagnet that increases or decreases

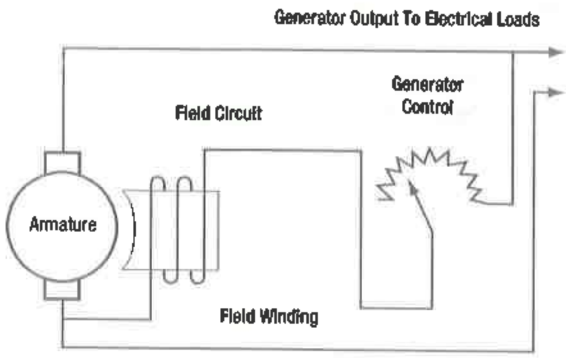


Figure 8-21. Voltage regulator for low-output generator.

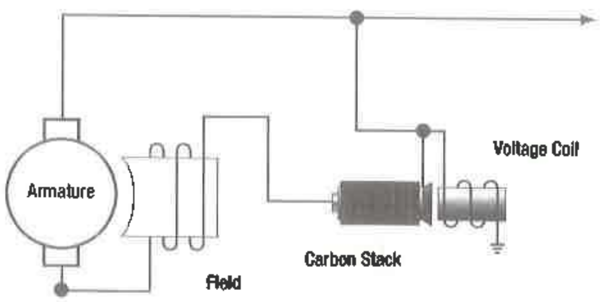


Figure 8-22. Carbon pile regulator.

strength as output voltage changes. The magnetism of the voltage coil controls the pressure on the carbon stack. The pressure on the carbon stack controls the resistance of the carbon which controls field current.

Carbon pile regulators require regular maintenance to ensure accurate voltage regulation and so most have been replaced on aircraft with more modern systems.

Three Unit Regulators

Three unit regulators used with DC generator systems are made of three distinct units. Each performs a specific function vital to correct electrical system operation. A typical three unit regulator consists of three relays mounted in a single housing. Each monitors generator outputs and opens or closes relay contact points according to system needs. A three unit regulator is shown in ***Figure 8-23***.

The voltage regulator section monitors generator output voltage and controls the generator field current as needed. As seen in ***Figure 8-24***, the voltage coil is connected in parallel with the generator output, and so measures the voltage of the system. If voltage exceeds a predetermined limit, the voltage coil becomes a magnet and opens the contact points. With the points open, field current must travel through a resistor and so field current goes down. The dotted arrow shows the current flowing through the voltage regulator when the relay points are open.

Since this voltage regulator has only two positions (open and closed), the unit must constantly adjust to maintain accurate voltage control. During normal operation, the points are opening and closing at regular intervals. They are in effect vibrating. As the points vibrate, the field current raises and lowers and the field magnetism averages to a level that maintains the correct output voltage. If the system requires more output, the points remain closed longer and vice versa.

The current limiter section contains a relay with a coil wired in series with the generator output. As seen in ***Figure 8-25***, all the output current must travel through the current coil. This creates a relay that is sensitive to the current output. That is, if generator output current increases, the relay points open and vice versa. The dotted line shows the current flowing to the generator field when the current limiter points are open. It should be noted that, unlike the voltage regulator relay, the current limiter is closed during normal flight. Only during extreme current loads must the limiter points open.

Figure 8-23. The three relays found on this regulator are used to regulate voltage.

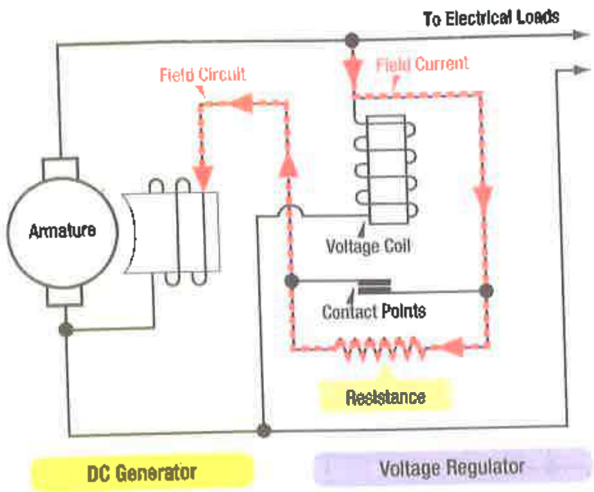


Figure 8-24. Voltage regulator.

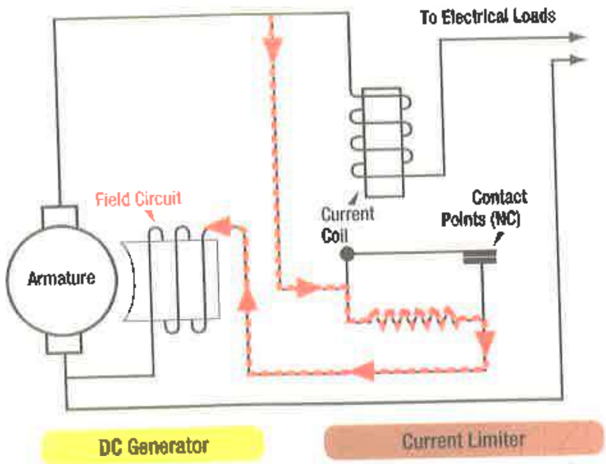


Figure 8-25. Current limiter.

The third unit of a three unit regulator prevents current from leaving the battery and feeding the generator. This direction of flow would discharge the battery and is opposite of normal operation. The simple reverse current relay shown in ***Figure 8-26*** contains both a voltage coil and a current coil. The voltage coil is wired in parallel to the generator output and is energized any time the generator output reaches its operational voltage. As the voltage coil is energized, the contact points close and the current flows to the electrical loads shown by the dotted lines.

The diagram shows the reverse current relay in its normal operating position. As current flows to the loads the coil is energized and the points remain closed. If there is no generator output due to a system failure, the contact points open because magnetism in the relay is lost. With the points open, the generator automatically disconnects from the electrical system, which prevents reverse flow from the load bus to the generator. A typical three unit regulator is shown in ***Figure 8-27***.

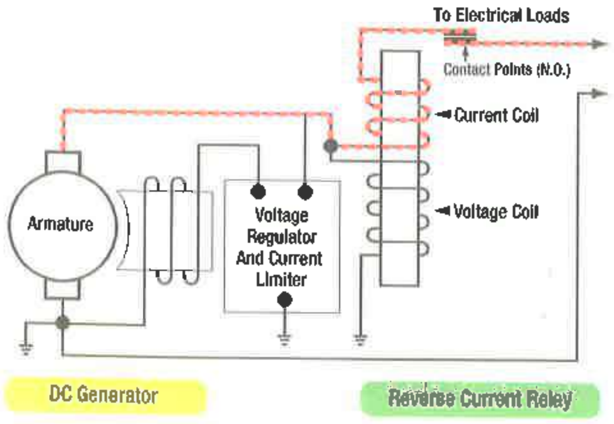


Figure 8-26. Reverse-current relay.

CIRCUIT PROTECTION

Reverse Current Sensing

If the generator were to fail and/or can not maintain the required voltage level, it eventually begins to draw current instead of providing it. It then becomes a load to the other operating generators or the battery. It must then be removed from the bus. The reverse current sensor monitors this condition where current is flowing to the generator not from it. If this occurs, the system opens the generator relay and disconnects the generator from the bus.

Over Voltage Protection

The over voltage protection system compares the sampled voltage to a reference voltage. This protection circuit is used to open the relay that controls the field excitation current. It is generally found on more complex generator control systems.

Over Excitation Protection

When one generator in a paralleled system fails, the other generators can become overexcited and tend to carry more than their share of the load. Basically, this

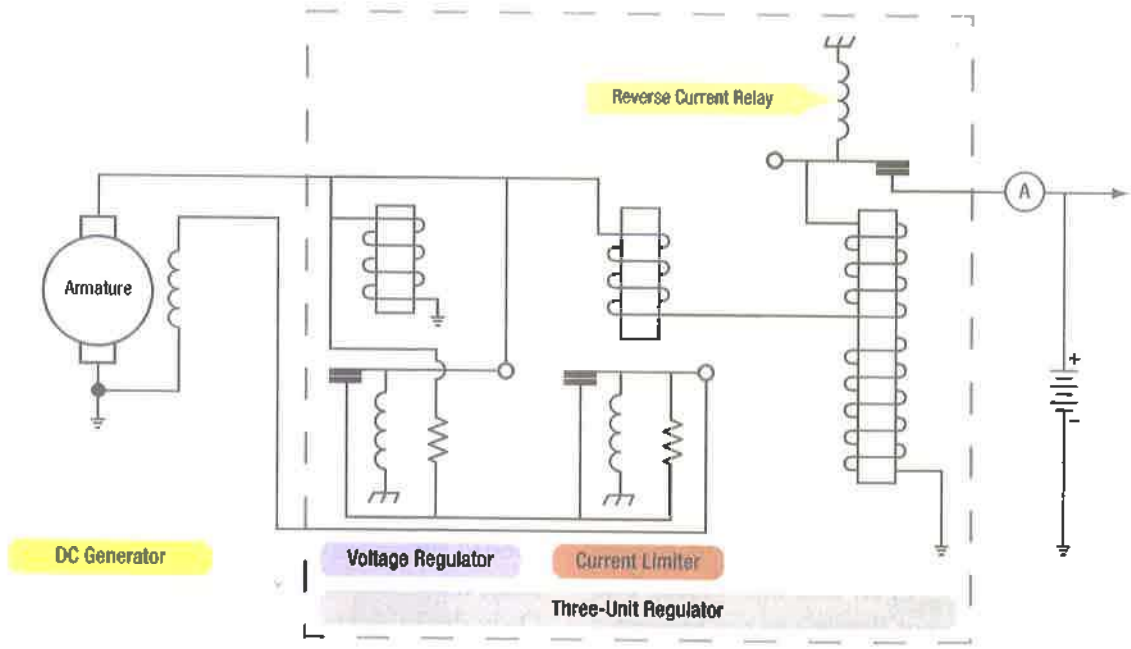


Figure 8-27. Three-unit regulator for variable speed generators.

condition causes the generator to produce too much current. If this condition is sensed, the over excited generator must be brought back within limits, otherwise damage occurs. The over excitation circuit works in conjunction with the over voltage circuit to control the generator.

Current Limiting Devices

In addition to the protection proved by the GCUs and the BPCU, individual conductive circuits on an aircraft are protected with current limiting devices. Conductors should be protected with circuit breakers or fuses located as close as possible to the power source bus. Normally, the manufacturer of the electrical equipment specifies the fuse or circuit breaker to be used when installing equipment.

The circuit breaker or fuse should open the circuit before the conductor emits smoke. To accomplish this, the time/current characteristic of the protection device must fall below that of the associated conductor.

Figure 8-28 is a chart used in selecting the circuit breaker and fuse protection for copper conductors. This limited chart is applicable to a specific set of ambient temperatures and wire bundle sizes and is presented as a sample only. It is important to consult such data before selecting a conductor for a specific purpose.

Fuses

A fuse is placed in series with the voltage source and all current must flow through it. (*Figure 8-29*) Fuses consists of a strip of metal that is enclosed in a glass or plastic housing. The metal strip has a low melting point and is usually made of lead, tin, or copper. When the current exceeds the capacity of the fuse the metal strip heats up and breaks. As a result of this, the flow of current in the circuit stops.

There are two basic types of fuses; fast acting and slow blow. The fast acting type opens immediately when their particular current rating is exceeded. This is important for electric devices that can quickly be destroyed when too much current flows through them for even a small amount of time. Slow blow fuses have a coiled construction inside. They are designed to open only on a continued overload, such as a short circuit.

Circuit Breakers

A circuit breaker is an electrical switch designed to protect a circuit from damage caused by an overload or short circuit. Its basic function is to detect a fault condition and immediately discontinue electrical flow. Unlike a fuse that operates once and then must be replaced, a circuit breaker can be reset to resume normal operation. All resettable circuit breakers should open the circuit in which they are installed regardless of the position of the operating control when an overload or circuit fault exists. Such circuit breakers are referred to as trip-free. (*Figure 8-30*)

Wire AN Gauge Copper	Circuit Breaker Amperage	Fuse Amperage
22	5	5
20	7.5	5
18	10	10
16	15	10
14	20	15
12	30	20
10	40	30
8	50	50
6	80	70
4	100	70
2	125	100
1		150
0		150

Figure 8-28. Wire and circuit protection chart.

Figure 8-29. Aircraft buss fuse.

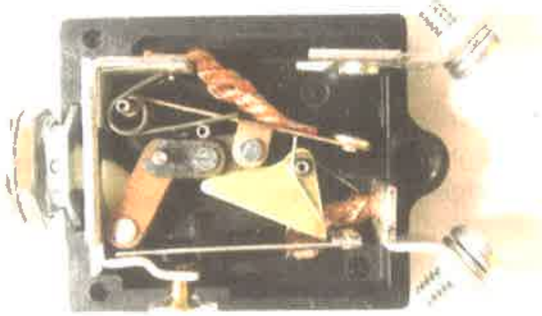


Figure 8-30. Circuit breaker panel.

Because automatic reset breakers automatically reset themselves they should not be used in aircraft. When a circuit breaker trips, the electrical circuit should be checked and the fault removed before the circuit breaker is reset. Sometimes circuit breakers trip for no apparent reason and the circuit breaker can be reset one time. If the circuit breaker trips again, a circuit fault exists, and the technician must troubleshoot the circuit before resetting it. (***Figure 8-31***)

New aircraft use a digital circuit protection architecture. This system monitors the amperage through a particular circuit. When that maximum amperage is reached, the power is rerouted away from the circuit. This system reduces the use of mechanical circuit breakers resulting in weight savings and the reduction of mechanical parts.

POWER DISTRIBUTION

POWER DISTRIBUTION ON SMALL MULTI-ENGINE AIRCRAFT

The power distribution systems found on modern multi engine aircraft contain several distribution points (busses) and a variety of control and protection components to ensure the reliability of electrical power. As aircraft employ more electronics to perform various tasks, the power systems become more complex and more reliable. One means to increase reliability is to ensure more than one power source can be used to power any given load. Another important design concept is to supply critical electrical loads from more than one bus. Multi engine aircraft typically have two generators and have multiple distribution busses fed from each generator.

Figure 8-32 shows a simplified diagram of the power distribution system for a multi engine aircraft. It contains two starter generators to start the engines and generate DC power. The system is defined as a split bus distribution system since there are two generator busses that share the electrical loads by connecting to each sub-bus through a diode and current limiter. The generators are operated in parallel and equally carry the loads. The primary power supplied for this aircraft is DC, although small amounts of AC are supplied by two inverters. The diagram shows the AC power distribution at the top and mid left side of the diagram. One inverter is used for main AC power and the second operated in standby and ready as a backup. Both inverters produce

Figure 8-31. Circuit breaker panel.

26 volt AC and 115 volt AC. There is an inverter select relay operated by a pilot controlled switch used to choose which inverter is active.

The hot battery bus (right side of ***Figure 8-32***) shows a direct connection to the battery. This bus is always hot if there is a charged battery in the aircraft. Items powered by this bus may include some basics like the entry door lighting and the aircraft clock, which should always have power available. Other items on this bus are critical to flight safety, such as fire extinguishers, fuel shut offs, and fuel pumps. During a massive system failure, the hot battery bus is the last bus on the aircraft that should fail. If the battery switch is closed and its relay activated, battery power is connected to the main battery bus and to the isolation bus. The main battery bus carries current for engine starts and external power. Hence, the main bus must be large enough for the heaviest current loads of the aircraft. It is logical to place this bus as close as practical to the battery and starters and to ensure it is well protected from shorts to ground. The isolation bus connects to the left and right busses and receives power whenever the main battery bus is energized and connects output of the left and right generators in parallel. The output of the two generators is then sent to the loads through additional busses. The generator busses are connected to the isolation bus through a current limiter. There are several current limiters used in this system for protection between busses.

As can be seen in ***Figure 8-32***, a current limiter is depicted as two triangles pointed toward each other. The current limiter between the isolation bus and the main generator busses are rated at 325 amps and can only be replaced on the ground, after the malfunction that caused the excess current draw is repaired.

Each DC generator is connected to their respective main busses. Since the busses are connected under normal circumstances, the generators operate in parallel. Both generators feed all loads together. If one generator fails or a current limiter opens, each generator can operate independently. This design allows for redundancy in the event of failure and provides battery backup in the event of a dual generator failure. In the center of ***Figure 8-32*** are four dual fed electrical busses. They are considered dual feed since they receive power from both generator busses. If a fault occurs, either bus can power any or all loads. During the design of the aircraft, the electrical loads are evenly distributed between each of the busses. It is also important to power redundant systems from

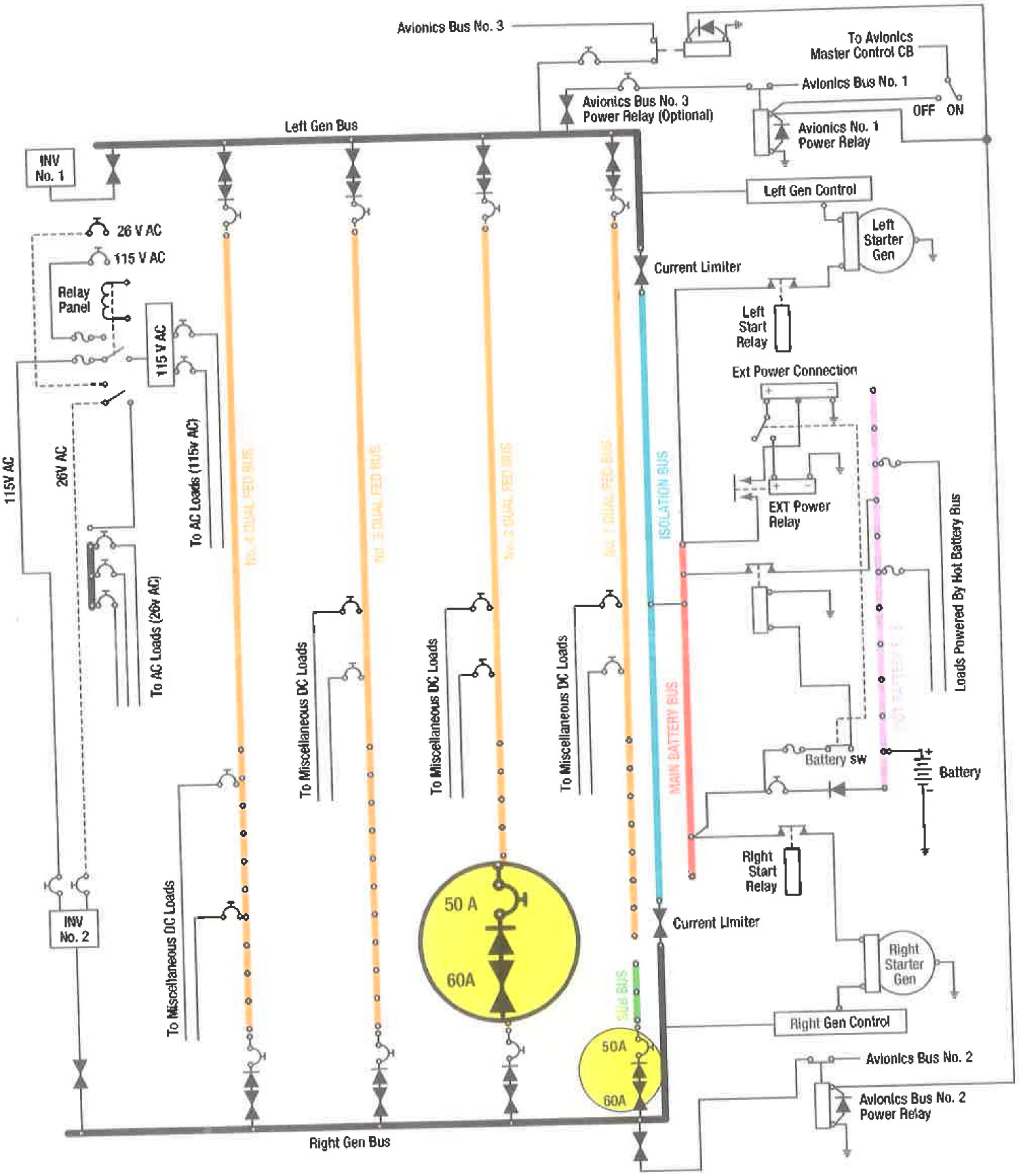


Figure 8-32. Diagram of the power distribution system for a twin-engine aircraft.

different busses. For example, the right windshield heat would be powered by a different bus from the one that powers the left windshield heat. If one bus fails, at least one windshield heater continues to work properly, and the aircraft can be landed safely.

Notice that the dual feed busses are connected to the main generator busses through both a current limiter and a diode. (***Figure 8-33***) A diode allows current flow in only one direction. The current can flow from the generator bus to the dual feed bus, but cannot flow from the dual fed bus to the main generator bus. The diode is placed in the circuit so the main bus must be more positive than the sub-bus for current flow. This circuit also contains a current limiter and a circuit breaker which can be reset by the pilot. The circuit breaker is rated at a slightly lower current value than the current limiter; therefore the circuit breaker should open if an overload exists. If the circuit breaker fails to open, the current limiter provides backup protection and disconnects the circuit.

POWER DISTRIBUTION ON LARGE AIRCRAFT

Split Bus Systems

Modern larger aircraft employ a split bus power distribution system. During normal conditions, each engine driven AC generator powers only one main AC bus. The busses are kept apart from each other, and two generators can never power the same bus simultaneously. This is especially important since the generator output current is not phase regulated. If two out-of-phase generators were connected to the same bus, damage would occur. The split bus system allows each generator to power either bus, but not at the same time.

On all modern split bus systems, the APU can be started and operated during flight. This allows the APU generator to provide back up power in the event of a main generator failure. The three AC generators are shown at the bottom of ***Figure 8-34***. These generators are connected to their respective busses through the Generator Breakers (GBs). For example, generator 1 sends current through GB1 to AC bus 1. AC bus 1 feeds a variety of primary loads, and feeds sub busses as well which in turn power additional loads. With both generators operating and all systems normal, AC bus 1 and AC bus 2 are kept isolated. During flight, the Auxiliary Power Breaker (APB) (bottom center of ***Figure 8-34***) would be open with the APU generator off. If generator 1 should fail, the following happens:

- The GB 1 is opened by the GCU to disconnect the failed generator.
- The BPCU closes BTB 1 and BTB 2 supplying AC power to AC bus 1 from generator 2.
- The pilots start the APU and connect the APU generator. At that time, the BPCU and GCUs move the appropriate BTBs to correctly configure the system so the APU powers bus 1 and generator 2 powers bus 2.

If all generators fail, AC is also available through the static inverter. (center of ***Figure 8-34***) The inverter is powered from the hot battery bus and used for essential AC loads. Of course, the GCUs and BPCU take the appropriate actions to disconnect defective units and continue to feed essential AC loads using inverter power. To produce DC power, AC bus 1 sends current to its Transformer Rectifier (TR). (***Figure 8-34***)

The TR unit is used to change AC to DC. Its output is therefore compatible with the aircraft battery at 26 volt DC. Since DC power is not phase sensitive, the DC busses are connected during normal operation. In the event of a bus problem, the BPCU may isolate one or more DC busses to ensure correct distribution of DC power.

Parallel Bus Systems

Some multi engine aircraft, employ a parallel power distribution system. During normal conditions, all engine driven generators connect together and power the AC loads. In this configuration, the generators operate

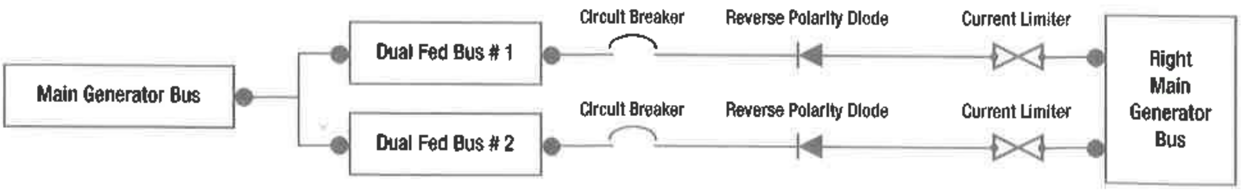


Figure 8-33. Dual-feed bus system.

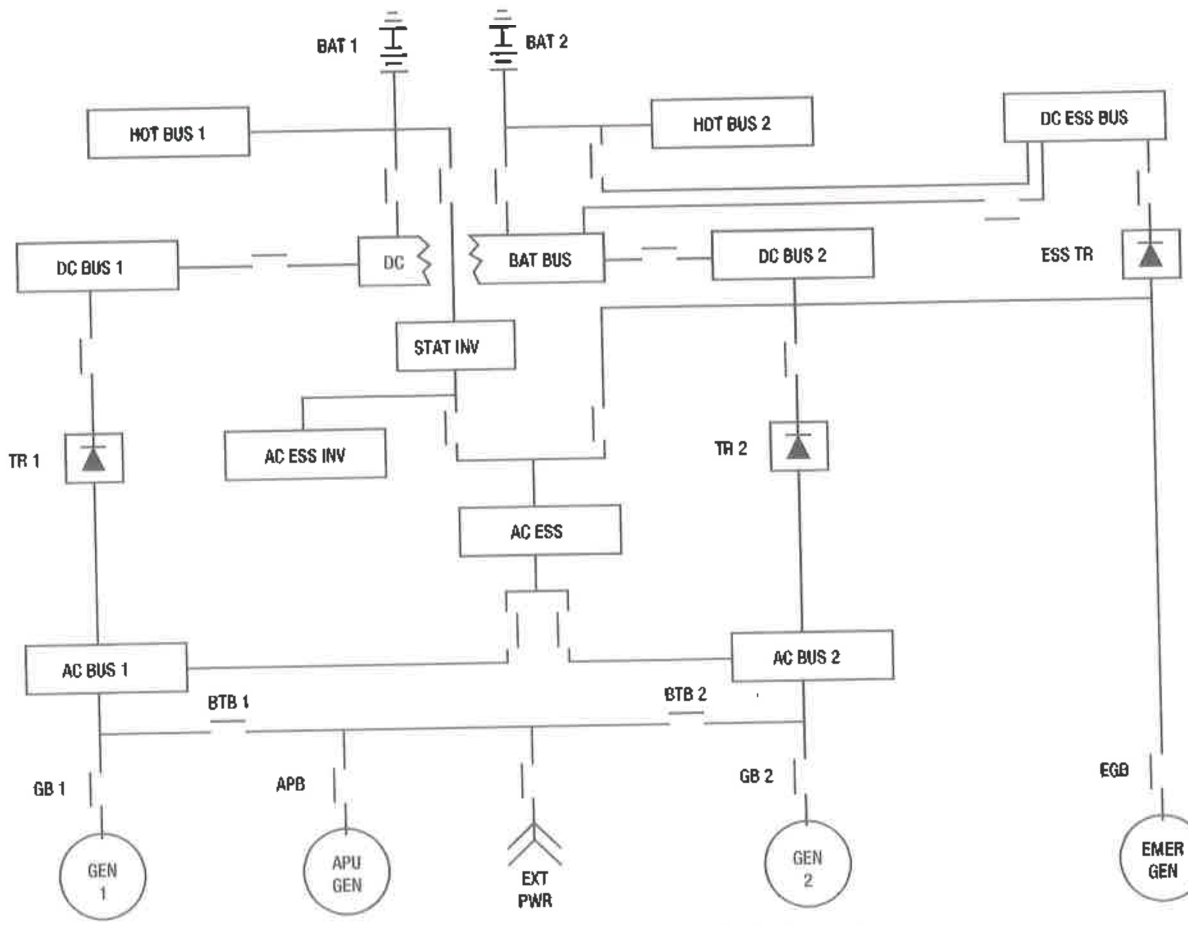


Figure 8-34. Schematic of split-bus power distribution system.

in parallel and must be phase regulated so that their output frequency reaches their positive and negative peaks simultaneously. One advantage of parallel systems is that in the event of a generator failure, the busses are already connected, and the defective generator needs only be isolated from the system. A paralleling or synchronizing bus is used to connect the generators. Most of these systems are less automated and require the flight crew to monitor the systems and manually to control the bus contactors. GBs are used to connect and disconnect the generators. ***Figure 8-35*** shows a simplified parallel power distribution system.

The APU (bottom right) is not operational in flight and cannot provide backup power. It is for ground operations only. The three main generators (bottom of ***Figure 8-35***) are connected to their respective AC bus through GB 1, 2, and 3. The AC busses are connected to the sync bus through three BTBs. In this manner, all three generators share the entire AC loads. Again, all generators connected to the sync bus must be in phase. If one fails, the flight crew would isolate the defective generator and the flight would continue without interruption.

The number 1 and 2 DC busses (top left of ***Figure 8-35***) are used to feed the DC electrical loads of the aircraft. DC bus 1 receives power from AC bus 1 though TR1. DC bus 2 is fed from AC bus 2. The DC busses also connect to the battery bus and eventually to the battery. A diode prevents the essential DC bus from powering DC bus 1, receiving power from the essential TR, which receives power from the essential AC bus. This provides an extra layer of redundancy since the essential AC bus can be isolated and fed from any main generator. ***Figure 8-35*** shows generator 3 powering the essential AC bus.

Split Parallel Systems

A split parallel bus employs the best of both split bus and parallel bus systems. It can operate with all generators in parallel or independently as in a split bus system. During normal flight, all engine driven generators are operated in parallel. The system is operated in split bus mode only under certain failure conditions or when using external power. Split parallel systems are computer controlled using GCU and BPCU.

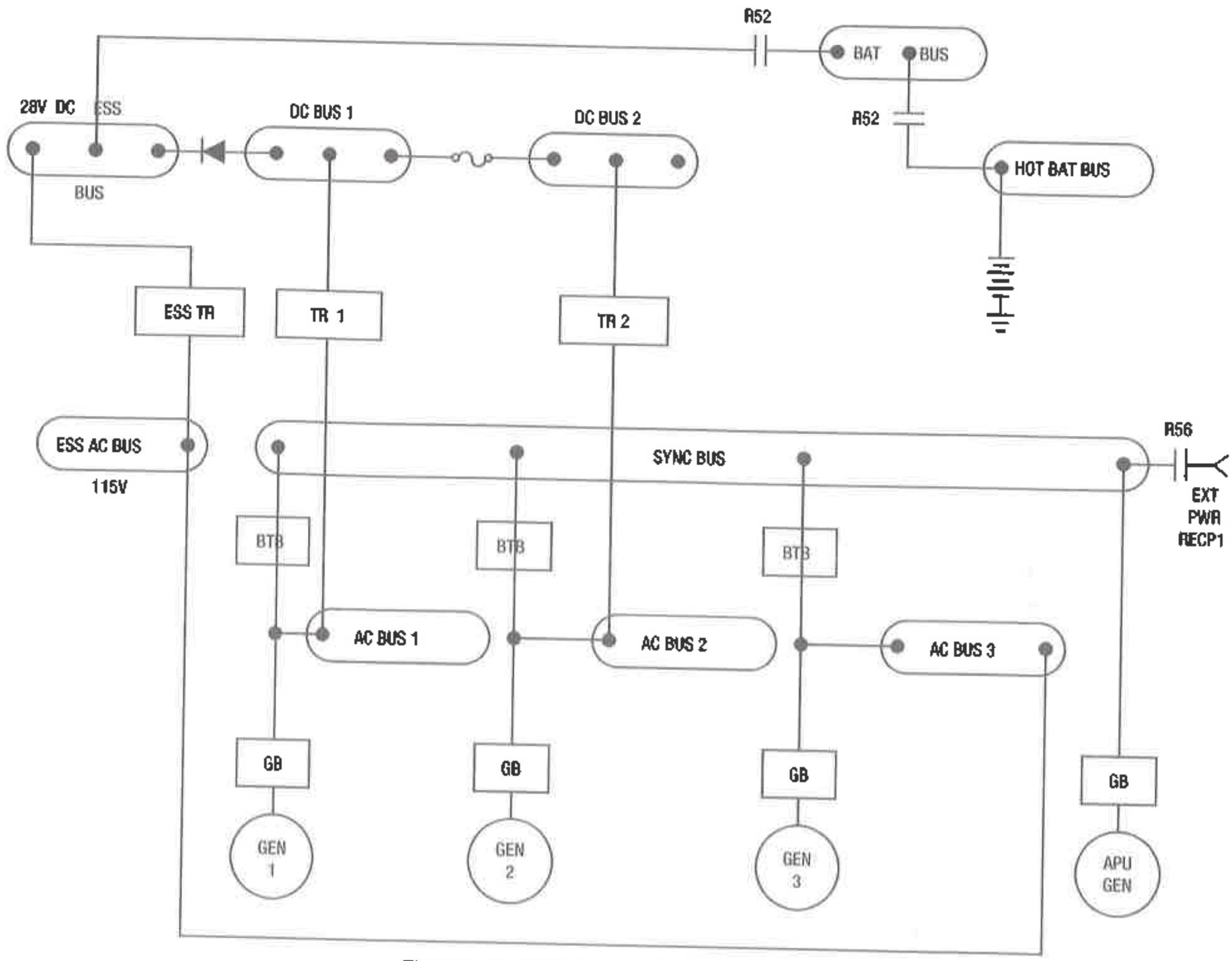


Figure 8-35. Parallel power distribution system.

Figure 8-36 shows a simplified split parallel power system. The main generators (top of *Figure 8-36*) are driven by the main engines. Each generator is connected to its load bus through a Generator Control Breaker (GCB). The generator control unit closes the GCB when the pilot calls for generator power and all systems are operating normally. Each load bus is connected to various electrical systems and additional sub-busses.

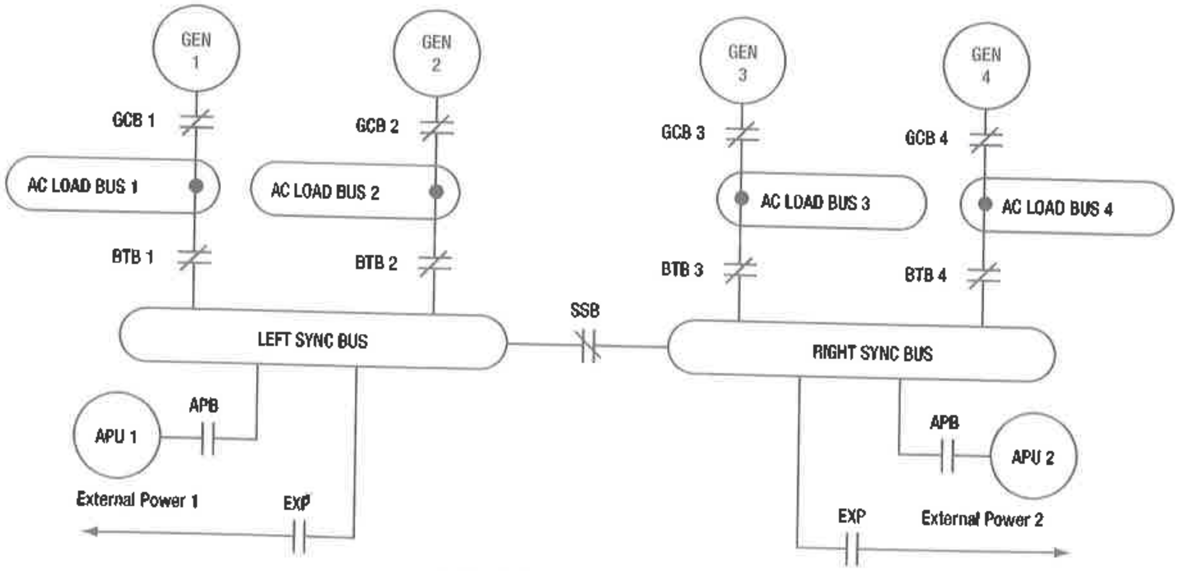


Figure 8-36. Split-parallel distribution system.

The BTBs are controlled by the BPCU and connect each load bus to the left and right sync bus. A Split Systems Breaker (SSB) is used to connect the left and right sync busses and is closed during normal flight. With the SSB, GCBs, and BTBs, in the closed position the generators operate in parallel.

If the aircraft electrical system experiences a malfunction, the control units make the appropriate adjustments to ensure all necessary loads receive power. For example, if generator 1 fails, GCU 1 detects the fault and command GCB 1 to open. With GCB 1 open, load bus 1 now feeds from the sync bus and the three operating generators. In another example, if load bus 4 should short to ground, BPCU 4 opens the GCB 4 and BTB 4. This isolates the shorted bus (load bus 4). All loads on the shorted bus are no longer powered, and generator 4 is no longer available. However, with three remaining generators operational, the flight continues safely.

All large aircraft contain a DC power distribution system. The DC system is used for battery and emergency operations. The DC system is like those previously discussed, powered by TR units. The TRs are connected to the AC busses and convert AC into 26 volt DC. The DC power systems are the final backups in the event of a catastrophic electrical failure. The systems most critical to fly the aircraft can receive power from the battery. This aircraft also contains a static inverter to provide emergency AC when needed.

INVERTERS, TRANSFORMERS, AND RECTIFIERS

INVERTERS

A modern inverter is a solid state device that converts DC power to AC power. The circuitry within an inverter is quite complex; however, for our purposes, it is simply a device that uses DC power to feed an AC distribution bus. Many inverters supply both 26 volt AC, as well as 115 volt AC. The aircraft can be designed to use either voltage or both simultaneously. If both are used, the power must be distributed on separate 26 volt and 115 volt AC busses. The AC inverter output frequency is 400 Cycles Per Second (CPS).

There are two basic types of inverters: rotary and static. Either type can be single phase or multiphase. The multiphase type is lighter for the same power rating, however there are complications in distributing multiphase power and in keeping the loads balanced. Most modern aircraft use solid state static inverters rather than the rotary type.

Rotary Inverters

There are many sizes, types, and configurations of rotary inverters. Such inverters are essentially AC generators and DC motors in one housing. The generator armature, and the motor armature are mounted on a common shaft that will rotate within the housing.

Permanent Magnet Rotary Inverters

A permanent magnet inverter is composed of a DC motor and a permanent magnet AC generator. Each has a separate stator mounted within a common housing. The motor armature is mounted on a rotor and connected to the DC supply through a commutator and brush assembly. The motor field windings are mounted on the housing and connect directly to the DC supply. A permanent magnet rotor is mounted at the opposite end of the same shaft as the armature, and the stator windings are mounted on the housing. This allows AC to be taken from the inverter without the use of brushes. ***Figure 8-37*** shows the internal wiring for this type of inverter. The generator rotor has six poles, magnetized to provide alternate north and south poles on its circumference.

When the motor field and armature are excited, the rotor will begin to turn. The permanent magnet will rotate within the AC stator coils and the magnetic flux developed by the permanent magnets will be cut by the conductors in the AC stator coils. An AC voltage will be produced in the windings whose polarity will change as each pole passes the windings. This type of inverter may be made multiphase by placing more AC stator coils in the housing in order to shift the phase the proper amount in each coil. As the name, rotary inverter implies, it has a revolving armature in the AC generator section. (***Figure 8-38***)

The DC motor in this inverter is a four pole, compound wound motor. The four field coils consist of many turns of fine wire, with a few turns of heavy wire placed on top. The fine wire is the shunt field, connected to the DC source through a filter and to ground through a centrifugal governor. The heavy wire is the series field, which is connected in series with the motor armature.

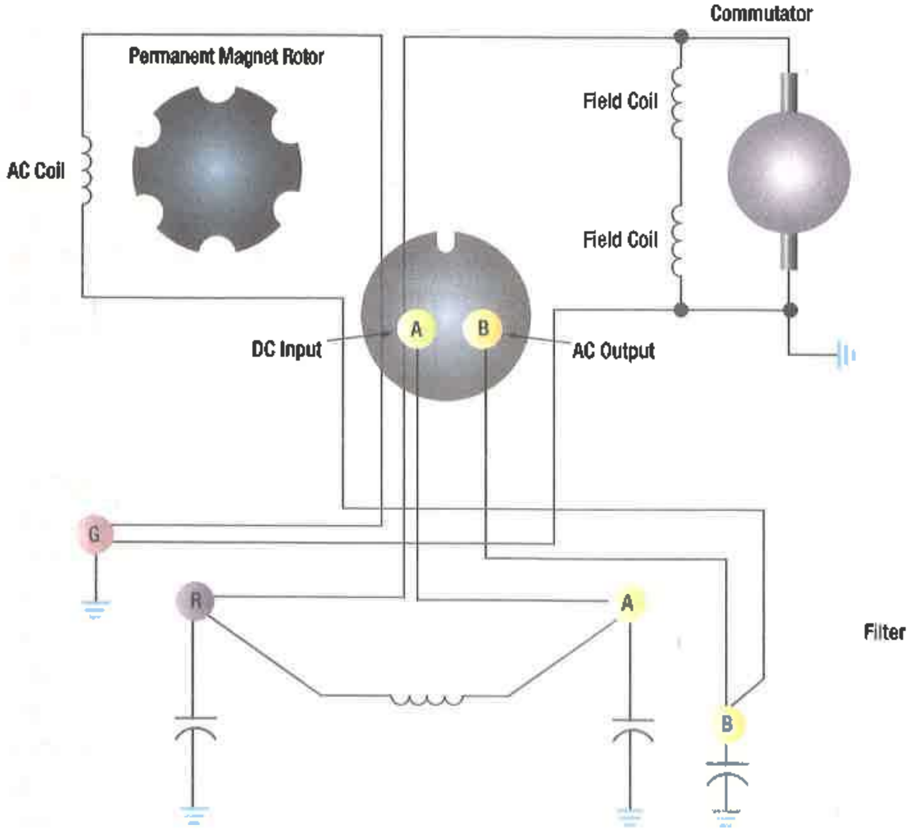


Figure 8-37. Internal wiring diagram of single-phase permanent magnet rotary inverter.

The centrifugal governor controls the speed by shunting a resistor that is in series with the shunt field when the motor reaches a certain speed.

The alternator is a three phase, four pole, star connected AC generator. The DC input is supplied to the generator field coils and connected to ground through a voltage regulator. The output is taken off the armature through three slip rings to provide three phase power. The inverter would be single phase if it had a single armature winding and one slip ring. The frequency of this type of unit is determined by the speed of the motor and the number of generator poles.

Inductor Type Rotary Inverters

Inductor inverters use a rotor made of soft iron laminations with grooves cut laterally across the surface to provide poles that correspond to the number of stator poles. As shown in ***Figure 8-39*** The field coils are wound on one set of stationary poles and the AC armature coils on the other set. When DC is applied to the field coils, a magnetic field is produced.

The rotor turns within the field coils and as the poles on the rotor align with the stationary poles a low reluctance path for flux is established from the field pole through the rotor poles to the AC armature pole and through the housing back to the field pole. In this circumstance, there will be a large amount of magnetic flux linking the AC coils.

When the rotor poles are between the stationary poles, there is a high reluctance path for flux, consisting mainly of air; and so there will be a small amount of magnetic flux linking the AC coils. This increase and decrease in flux density in the stator induces an alternating current in the AC coils.

The number of poles and the motor speed determine the frequency of this type of inverter. The DC stator field current controls the voltage. A cutaway view of an inductor type rotary inverter is shown in ***Figure 8-40***. ***Figure 8-41*** is a simplified diagram of an AC distribution system, utilizing a main and a standby rotary inverter.

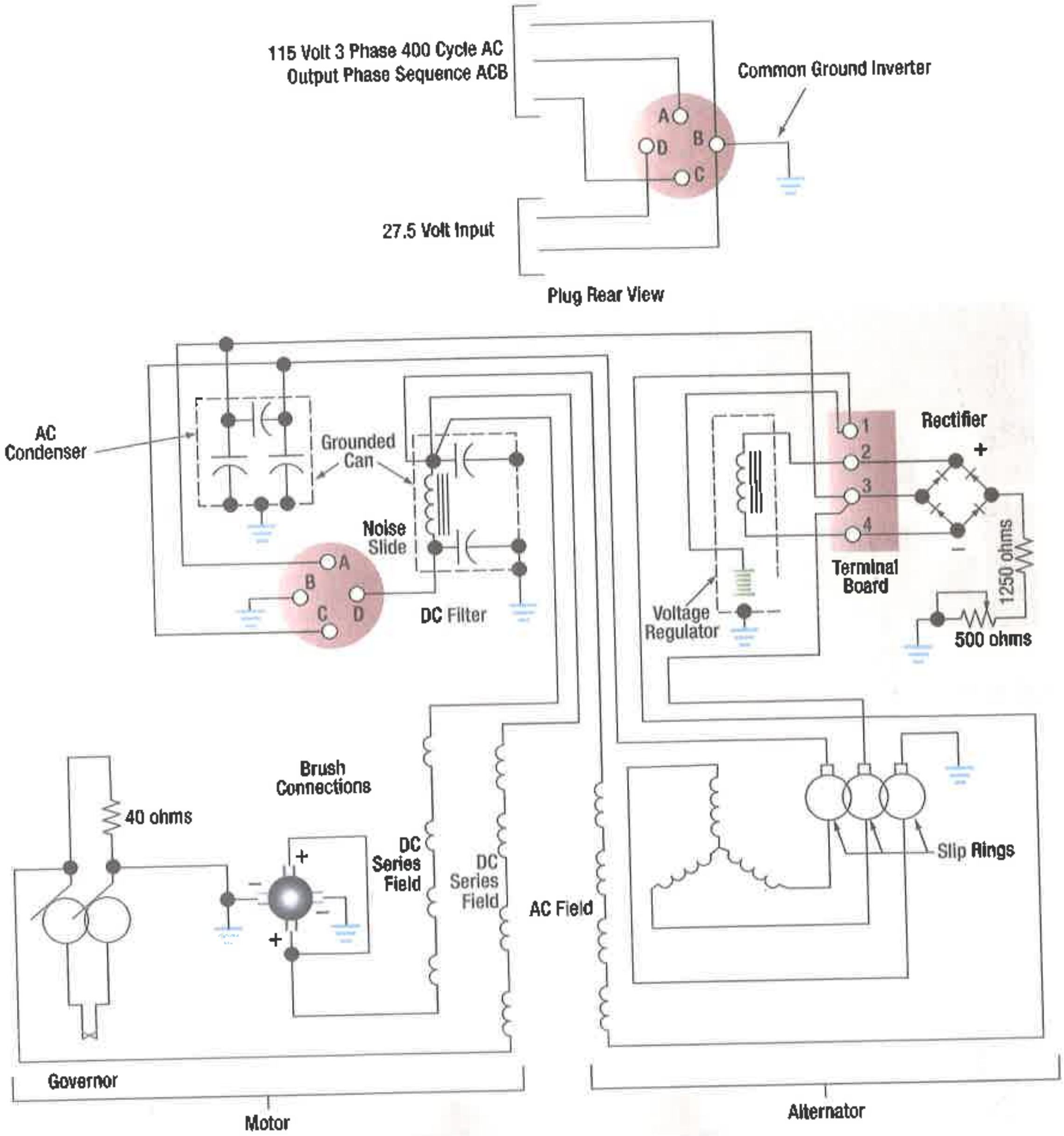


Figure 8-38. Internal wiring diagram of three-phase, revolving armature.

Static Inverters

In many applications where continuous DC voltage must be converted to AC voltage, static inverters are used in place of rotary inverters or generator sets. The rapid progress in semiconductors is extending the range of applications of such equipment into voltage and power ranges that were once impractical. Some applications are power supplies for frequency sensitive equipment, and conversion of wide frequency ranges to precise frequency power.

The use of static inverters in small aircraft also has increased rapidly. The technology has advanced to the point that static inverters are available for any requirement filled by rotary inverters. For example, 250 VA emergency AC supplies operated from batteries are in production, as are 2 500 VA main supplies. This type of equipment has advantages for aircraft applications, particularly the absence of moving parts and the adaptability to conduction cooling.

Static inverters, (referred to as solid-state inverters), are manufactured in a wide range of types which can be classified by the shape of the AC output waveform and power output capabilities. One of the most used static inverters produces a regulated sine wave output. (***Figure 8-42***). This inverter converts a low DC voltage into higher AC voltage. The AC output voltage is held to

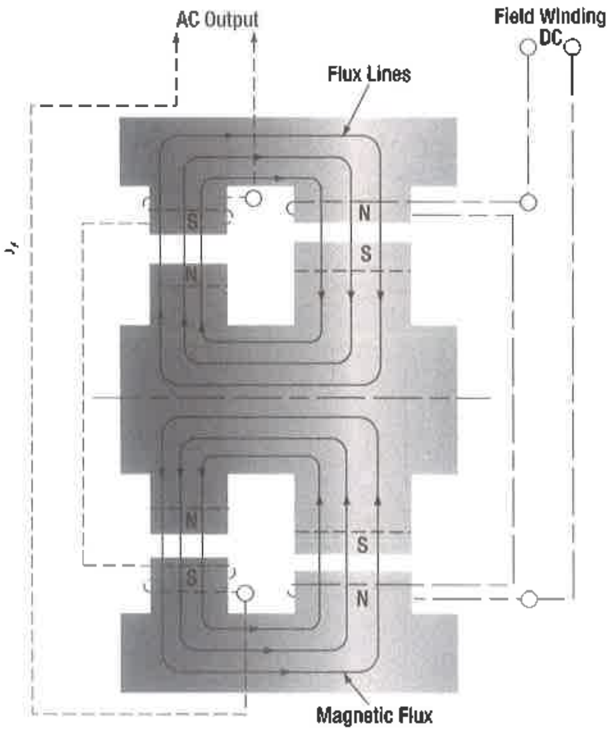


Figure 8-39. Diagram of basic inductor-type inverter.

a small tolerance, typically less than 1 percent with a full input load change. Output taps are normally provided to permit selection of various voltages. For example, taps may be provided for 105, 115, and 125 volt AC outputs. Frequency regulation is within a range of one cycle for a 0 – 100 percent load change.

Since static inverters use solid state components, they are considerably smaller, more compact, and much lighter in weight than rotary inverters. Features of static inverters include:

- High efficiency.
- Low maintenance, long life.
- No warm up period required.
- Capable of starting under load.
- Extremely quiet operation.
- Fast response to load changes.

Static inverters are commonly used to provide power for such frequency sensitive instruments as attitude and directional gyros. They can also provide power for

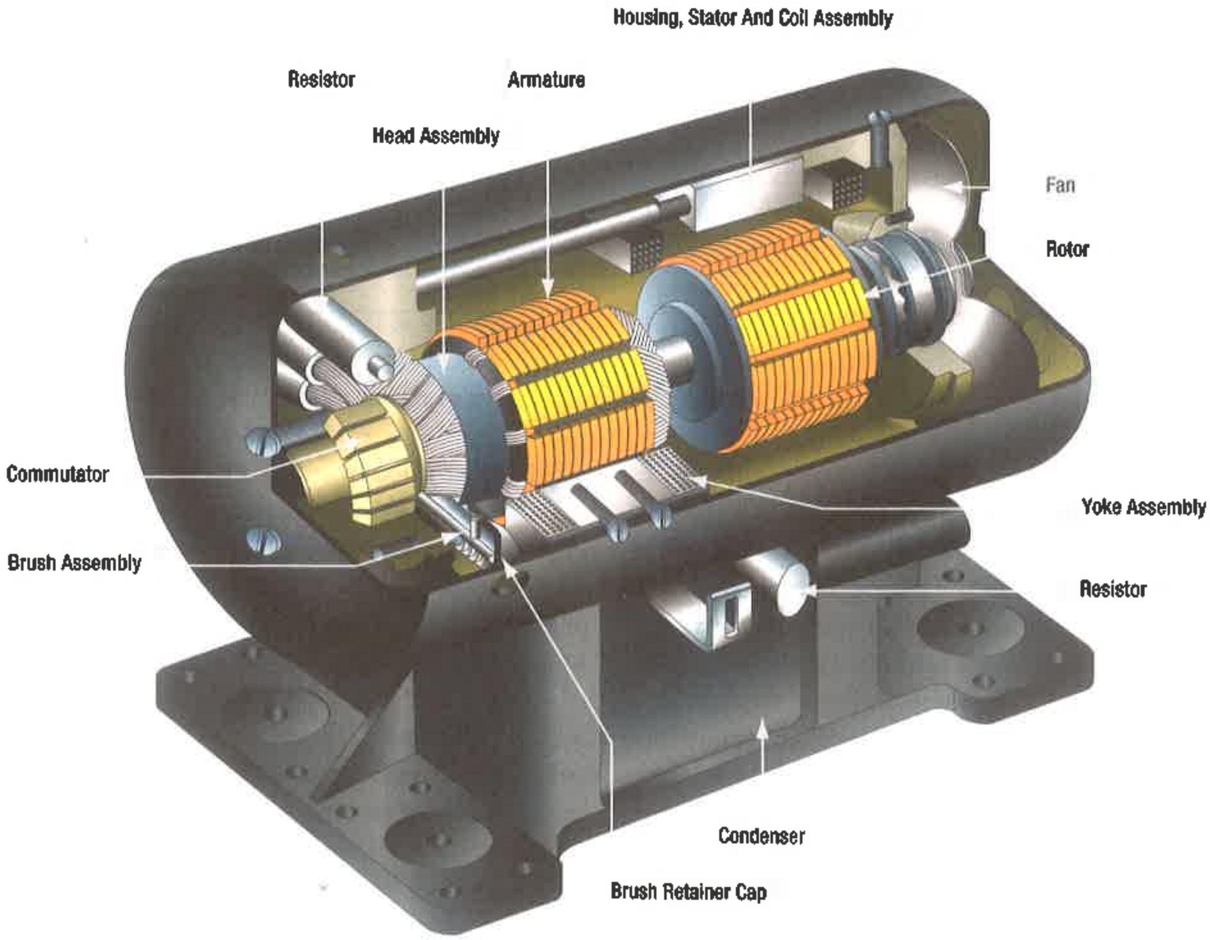


Figure 8-40. Cutaway view of inductor-type rotary inverter.

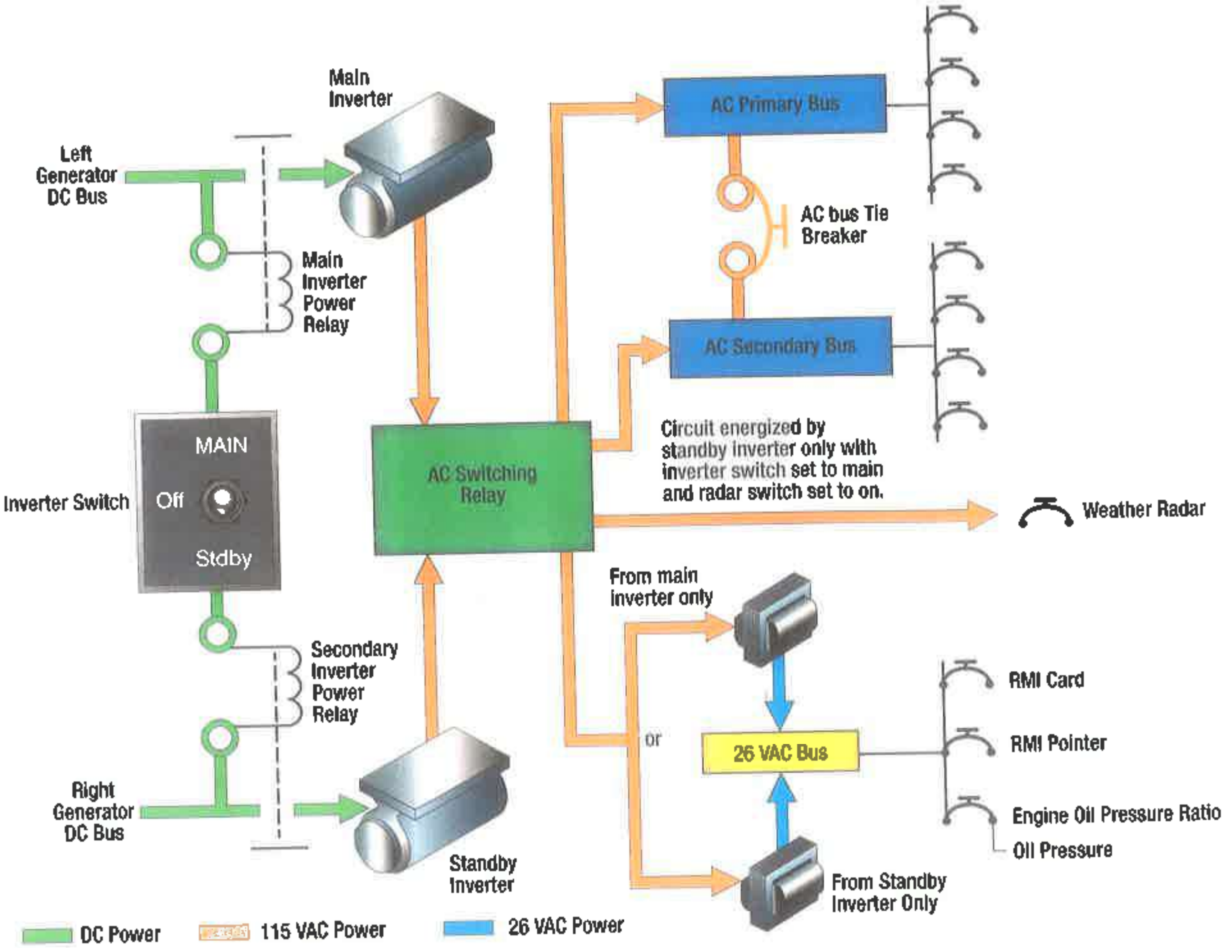


Figure 8-41. A typical aircraft AC power distribution system using main and standby rotary inverters.

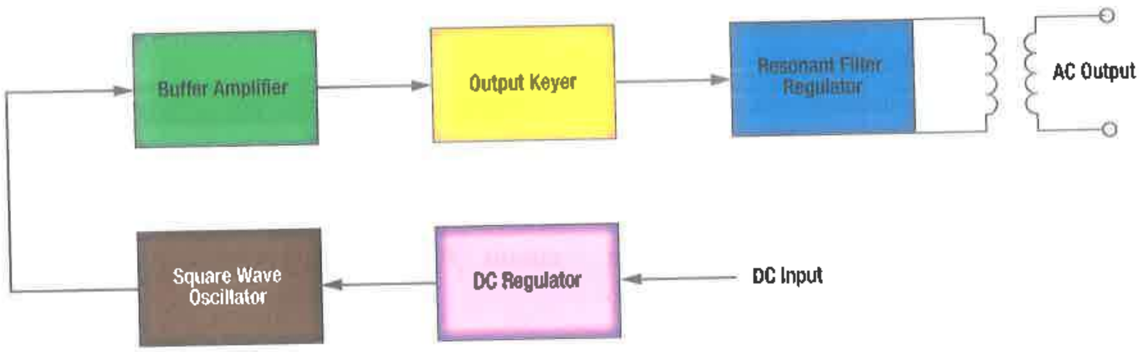


Figure 8-42. Regulated sine wave static inverter.

Autosyn and Magnesyn indicators and transmitters, radar, and other airborne applications. *Figure 8-43* is a schematic of a small aircraft auxiliary battery system with the battery as input to the inverter, and the output inverter feeding various subsystems.

TRANSFORMERS

A transformer changes electrical energy of a given voltage into electrical energy at a different voltage level. It consists of two coils that are not physically connected but are arranged so that the magnetic field surrounding one coil cuts through the other coil. When an alternating voltage is applied to (across) one coil, the varying magnetic field around that coil creates an alternating voltage in the other coil by induction. A transformer can also be used with pulsating DC, but a pure DC voltage cannot be used since only a varying voltage creates the varying magnetic field that is the basis of the induction process.

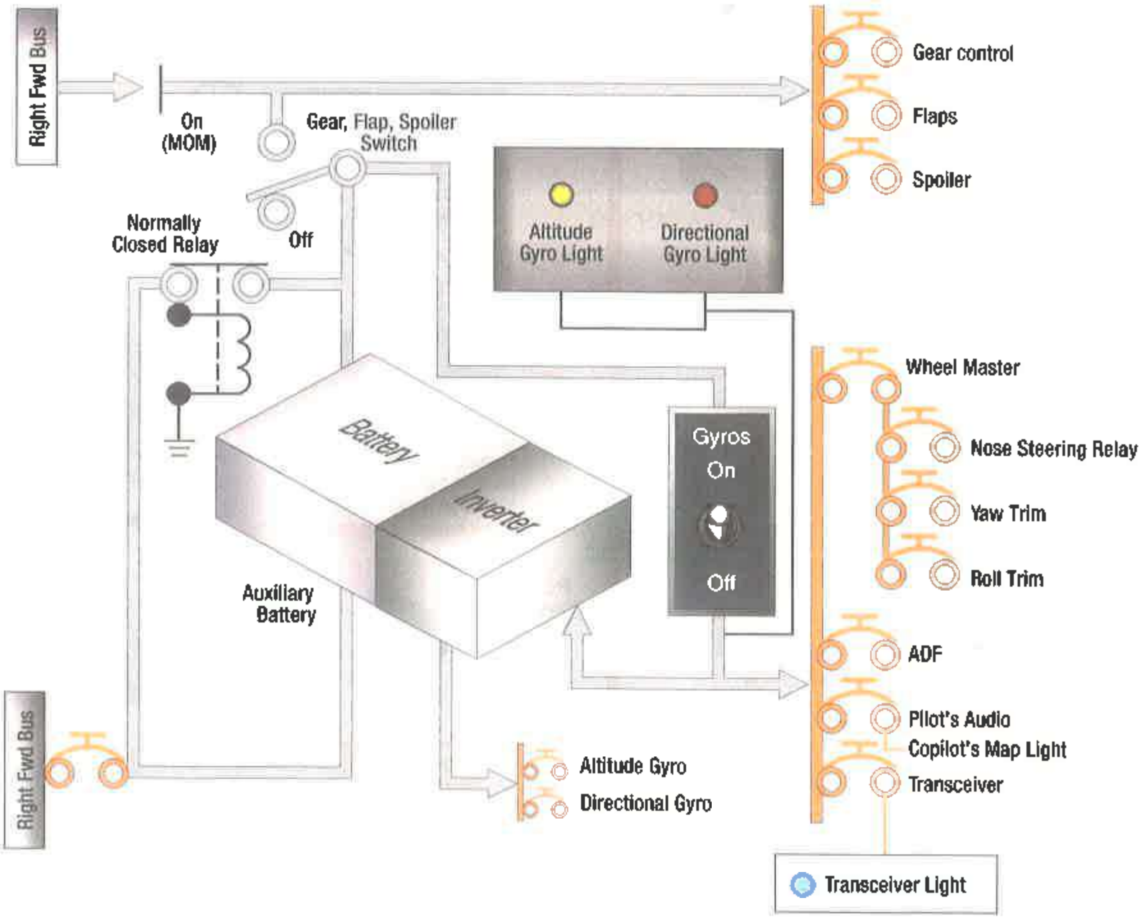


Figure 8-43. Auxiliary battery system using static inverter.

A transformer consists of three basic parts. (***Figure 8-44***) An iron core provides a circuit of low reluctance for magnetic lines of force, a primary winding which receives the electrical energy from the source of applied voltage, and a secondary winding which receives electrical energy by induction from the primary coil. The primary and secondary of this closed core transformer are wound on a closed core to obtain a maximum inductive effect between the two coils. There are two classes of transformers: voltage transformers used for stepping up or stepping down voltages, and current transformers used in instrument circuits.

In voltage transformers, the primary coils are connected in parallel across the supply voltage as shown in ***Figure 8-45A***. The primary windings of current transformers are connected in series in the primary circuit. (***Figure 8-45B***) Of these types, the voltage

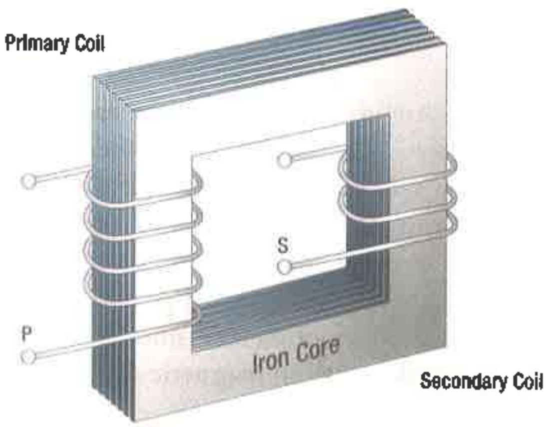


Figure 8-44. An iron-core transformer.

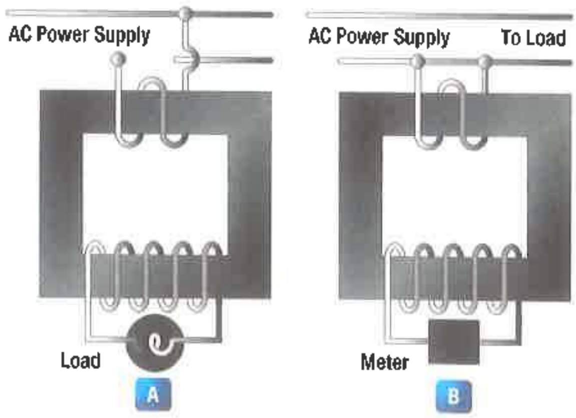


Figure 8-45. Voltage and current transformers.

There are many types of voltage transformers. Most are either step-up or step-down transformers. What determines whether a transformer is a step-up, or step-down is the "turns" ratio with similar wire diameter. The turns ratio is the ratio of the number of turns in the primary winding to the number of turns in the secondary winding. For example, the turns ratio of the step-down transformer in *Figure 8-46A* is 5 to 1, since there are five times as many turns in the primary as in the secondary. The step-up transformer shown in *Figure 8-46B* has a 1 to 4 turns ratio.

The ratio of the transformer's input voltage to the output voltage is the same as the turns ratio, if the transformer is 100 percent efficient. Thus, when 10 volts are applied to the primary of the transformer shown in *Figure 8-46A*, two volts are induced in the secondary. If 10 volts are applied to the primary of the transformer in *Figure 8-46B*, the output voltage across the secondary will be 40 volts.

No transformer can be constructed to be 100 percent efficient, although iron core transformers can approach this. This is because all the magnetic lines of force set up in the primary do not fully cut across the turns of the secondary coil. A certain amount of flux, called leakage flux, leaks out of the circuit. The measure of how well the flux of the primary is coupled into the secondary is called the "coefficient of coupling." For example, if the primary develops 10 000 lines of force and only 9 000 cuts across the secondary, the coefficient of coupling would be 0.9 or, stated another way, the transformer would be 90 percent efficient.

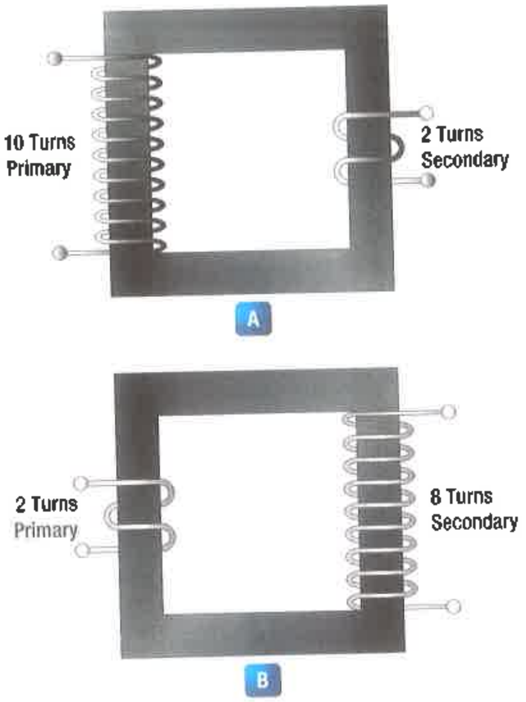


Figure 8-46. A step-down and a step-up transformer.

When an AC voltage is connected across the primary terminals, an alternating current will flow and self induce a voltage in the primary coil that is opposite and nearly equal to the applied voltage. The difference between these two voltages allows just enough current in the primary to magnetize its core. This is called the exciting, or magnetizing current. The magnetic field caused by this exciting current cuts across the secondary coil and induces a voltage by induction. If a load is connected across the secondary coil, the load current flowing through the secondary coil will produce a magnetic field which will tend to neutralize the field produced by the primary current. This will reduce the self induced (opposition) voltage in the primary coil and allow more primary current to flow. The primary current increases as the secondary load current increases, and decreases as the secondary load current decreases.

When the secondary load is removed, the primary current is again reduced to the small exciting current sufficient only to magnetize the iron core of the transformer. If a transformer steps up the voltage, it will step down the current by the same ratio. This should be evident if the power formula is considered. The power (I × E) of the output (secondary) electrical energy is the same as the input (primary) minus that loss in the transforming process. Thus, if 10 volts and 4 amps (40 watts of power) are used in the primary to produce a magnetic field, there will be 40 watts of power developed in the secondary (disregarding the loss).

When the turns ratio and the input voltage are known, the output voltage can be determined as follows: $E_2/E_1 = N_2/N_1$ Where E is the voltage of the primary, E_2 is the output voltage of the secondary, and N_1 and N_2 are the number of turns of the primary and secondary, respectively. Transposing the equation to find the output voltage gives: $E_2 = E_1 N_2/N_1$.

Voltage Transformers

These are the most used types of voltage transformers.

Power Transformers

Power Transformers are used to step-up or step-down voltages and current in many types of power supplies. They range in size from the small transformer shown in ***Figure 8-47*** used in a radio receiver, to the large transformers used to step-down high power line voltage to the 110 – 120 volt level used in homes.

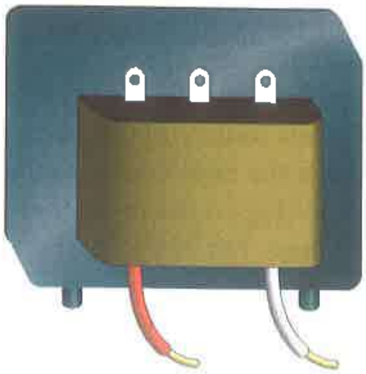


Figure 8-47. Power supply transformer.

Figure 8-48 shows the schematic symbol for an iron core transformer. In this case, the secondary is made up of three separate windings. Each winding supplies a different circuit with a specific voltage which saves the weight, space, and expense of three separate transformers. Each secondary has a midpoint connection, called a center tap which provides a selection of half the voltage across the whole winding. The leads from the various windings are color coded by the manufacturer, as labeled in ***Figure 8-48.***

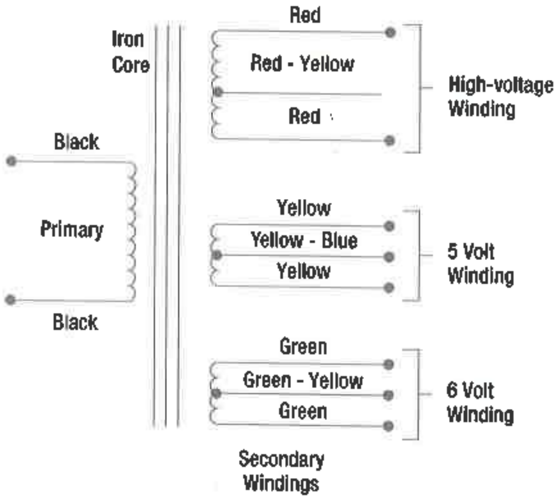


Figure 8-48. Schematic symbol for an iron-core power transformer.

Audio Transformers

Audio transformers resemble power transformers. They have only one secondary and are designed to operate over the range of audio frequencies (20 to 20 000 CPS).

RF Transformers

RF transformers operate in equipment that functions in the radio range of frequencies. The symbol for the RF transformer is the same as for an RF choke coil. It has an air core as shown in ***Figure 8-49.***

Figure 8-49. An air-core transformer.

Autotransformers

Auto transformers are normally used in power circuits; however, they may be designed for other uses. Two different symbols for autotransformers used in power or audio circuits are shown in ***Figure 8-50.*** If used in an RF communication or navigation circuit (***Figure 8-50B***), it is the same, except there is no symbol for an iron core. The autotransformer uses part of a winding as a primary, and depending on whether it is step-up or step-down, it uses all or part of the same winding as the secondary. For example, the autotransformer shown in ***Figure 8-50A*** could use the following choices for primary and secondary terminals as shown.

Primary		Secondary
1–2	used with	1–3
1–2	" "	2–3
1–3	" "	1–2
1–3	" "	2–3
2–3	" "	1–3
2–3	" "	1–2

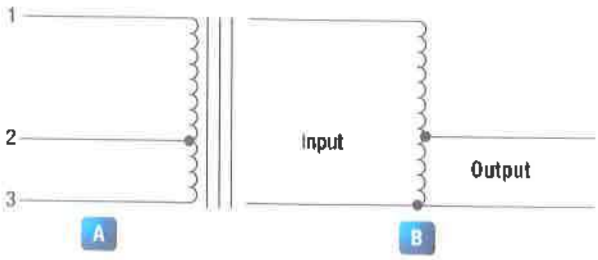


Figure 8-50. Autotransformers.

Current Transformers

Current transformers are used in AC power systems to sense generator line current and to provide a current proportional to the line current for circuit protection and control devices. The current transformer is a ring type transformer using a current carrying power lead as a primary (either the power lead or the ground lead of the AC generator). The current in the primary induces a current in the secondary by magnetic induction. The sides of all current transformers are marked "H1" and "H2" on the unit base. The transformers must be installed with the "H1" side toward the generator in the circuit for proper polarity. The secondary of the transformer should never be left open while the system is being operated. To do so could cause dangerously high voltages and could overheat the transformer. Therefore, the transformer output connections should always be connected by means of a jumper when the transformer is not being used but left in the system.

Transformer Losses

In addition to the power loss caused by imperfect coupling, transformers are subject to copper and iron losses. The resistance of the conductor comprising the turns of the coil causes copper loss. Iron losses are of two types called hysteresis loss and eddy current loss. Hysteresis loss is the electrical energy required to magnetize the transformer core, first in one direction and then in the other, in step with the applied alternating voltage. Eddy current loss is caused by electric currents induced in the transformer core by the varying magnetic fields. To reduce eddy current losses, cores are made of laminations coated with an insulation which reduces the circulation of induced currents.

Power In Transformers

Since a transformer does not add electricity to the circuit but merely changes or transforms the electricity that already exists from one voltage to another, the total amount of energy in a circuit must remain the same. If it were possible to construct a perfect transformer, it would have no loss of power; power would be transferred undiminished from one voltage to another. Since power is the product of volts times amperes, an increase in voltage by the transformer must result in a decrease in current and vice versa. There cannot be more power in the secondary side of a transformer than there is in the primary. The product of amperes times volts remains the same.

The transmission of power over long distances is accomplished by using transformers. At the power source, the voltage is stepped up to reduce line loss during transmission. At the point of utilization, the voltage is stepped down as it is not feasible to use high voltage to operate motors, lights, or other electrical appliances.

RECTIFIERS

Rectifier circuits change AC voltage into DC voltage and are one of the most used type of circuits in aircraft electronics. In ***Figure 8-51*** the resulting DC output is shown. The circuit has a single semiconductor diode and a load resistor. When the AC voltage cycles below zero, the diode shuts off and does not allow current flow until the AC cycles through zero voltage again. The result is pronounced pulsating DC. While this can be useful, half of the original AC voltage is unused.

A full wave rectifier creates pulsating DC from AC while using the full AC cycle. One way to do this is to tap the secondary coil at its midpoint and construct two circuits with the load resistor and a diode in each circuit. In ***Figure 8-52*** the diodes are arranged so that when current is flowing through one, the other blocks current.

When the AC cycles so the top of the secondary coil of the transformer is positive, current flows from ground, through a Load Resistor, Diode 1, and to the upper half of the coil. Current cannot flow through Diode 2 because it is blocked. (***Figure 8-52A***) As the AC cycles through zero, the polarity of the secondary coil changes. (***Figure 8-52B***) Current then flows from ground, through the load resistor, Diode 2, and to the bottom half of the secondary coil. Current flow through Diode 1 is blocked. This arrangement yields positive DC from cycling AC with no wasted current.

Another way to construct a full wave rectifier uses four semiconductor diodes in a bridge circuit. Because the secondary coil of the transformer is not tapped at the center, the resultant DC output is twice that of the two

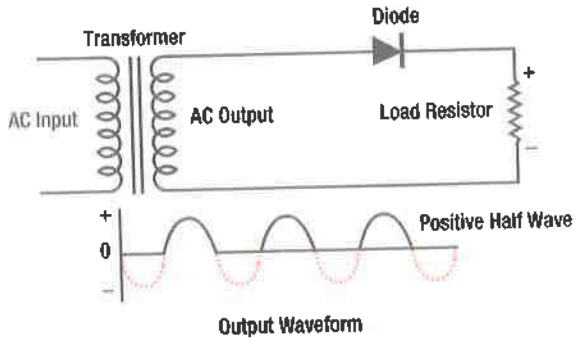


Figure 8-51. A half wave rectifier uses one diode to produce pulsating DC current from AC.

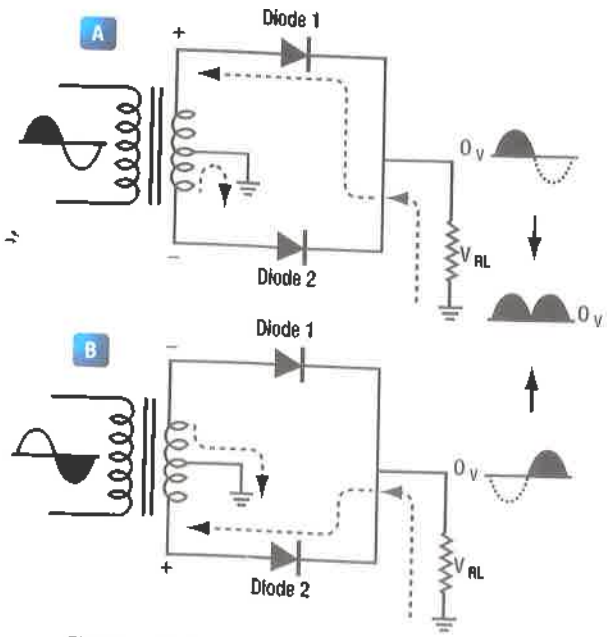


Figure 8-52. A full wave rectifier can be built by center tapping the secondary coil of the transformer.

diodes full wave rectifier. (*Figure 8-53*) During the first half of the AC cycle, the bottom of the secondary coil is negative. Current flows from it through diode D1, then through the load resistor, and through diode D2 on its way back to the top of the secondary coil. When the AC reverses its cycle, the polarity of the secondary coil changes. Current flows from the top of the coil through diode D3, then through the load resistor, and through diode D4 on its way back to the bottom of the secondary coil. The output waveform reflects the higher voltage achieved by rectifying the full AC cycle through the entire length of the secondary coil.

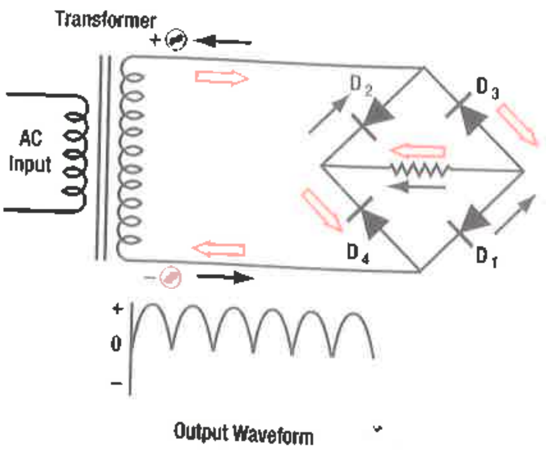


Figure 8-53. The bridge-type four-diode full wave rectifier circuit is most commonly used to rectify single-phase AC into DC avionics.

Use and rectification of three phase AC is also possible on aircraft with a specific benefit. The output DC is very smooth and does not drop to zero. A six diode circuit is built to rectify the typical three phase AC produced by the alternator. (*Figure 8-54*) Each stator coil corresponds to a phase of AC and becomes negative for 120° of rotation of the rotor. When stator 1 or the first phase is negative, current flows from it through diode D1, then through the load resistor and through diode D2 on its way back to the third phase coil.

Next, the second phase coil becomes negative and current flows through diode D3. It continues to flow through the load resistor and diode D4 on its way back to the first phase coil. Finally, the third stage coil becomes negative causing current to flow through diode D5, then the load resistor and diode D6 on its way back to the second phase coil. The output waveform of this three phase rectifier depicts the DC produced. It is a relatively steady, non-pulsing flow equivalent to just the tops of the individual curves. The phase overlap prevents voltage from falling to zero producing smooth DC from AC.

Silicon Controlled Rectifiers

Combination of semiconductor materials is not limited to a two type, three layer sandwich transistor. By creating a four layer sandwich of alternating types of semiconductor material (i.e., PNPN or NPNP), a slightly different semiconductor diode is created. As is the case in a two layer diode, circuit current is either blocked or permitted to flow through the diode in a

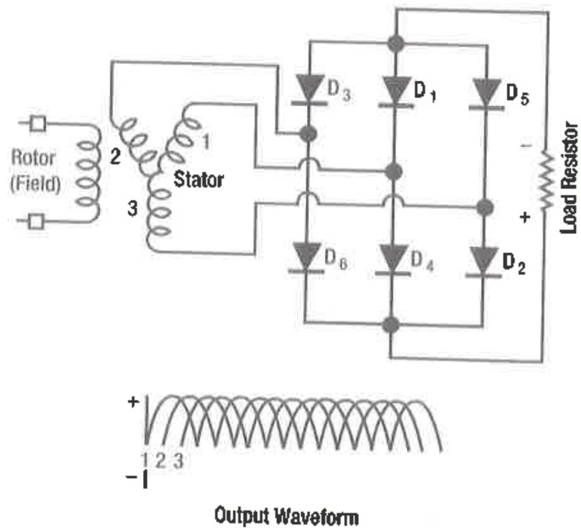


Figure 8-54. A six-diode three-phase AC rectifier.

single direction. Within a four layer diode, sometimes known as a Shockley diode, there are three junctions. The behavior of the junctions and the entire four layer diode can be understood by considering it to be two interconnected three layer transistors. (*Figure 8-55*)

Transistor behavior includes no current flow until the base material receives an applied voltage to narrow the depletion area at the base-emitter junction. The base materials in the four layer diode transistor model receive charge from the other transistor collector. With no other means of reducing any of the depletion areas at the junctions, it appears that current does not flow in either direction. However, if a large voltage is applied to forward bias the anode or cathode, at some point the ability to block the flow breaks down. Current flows through whichever transistor is charged. Collector current then charges the base of the other transistor and then current flows through the entire device.

Some caveats are necessary with this explanation. The transistors that comprise this four layer diode must be constructed of material like that in a Zener diode. That is, it must be able to endure the current flow without burning out. In this case, the voltage that causes the diode to conduct is known as breakover voltage rather than breakdown voltage. Additionally, this diode has the unique characteristic of allowing current flow to continue until the applied voltage is reduced significantly, in most cases until it is reduced to zero. In AC circuits, this would occur when the AC cycles.

While the four layer Shockley diode is useful as a switching device, a slight modification to its design creates a Silicon Controlled Rectifier (SCR). To construct a SCR, an additional terminal known as a gate is added. It provides more control and utility. In four layer semiconductor construction, there are always two junctions forward biased, and one junction reversed biased. The added terminal allows the momentary application of voltage to the reversed biased junction. All three junctions then become forward biased and current at the anode flows through the device. Once voltage is applied to the gate, the SCR become latched or locked on. Current continues to flow until the level drops off significantly, usually to zero. Then, another applied voltage through the gate is needed to reactivate the current flow. (*Figure 8-55 and Figure 8-56*)

Phase control is a key application for SCR. By limiting the percentage of a full cycle of AC voltage that is applied to a load, a reduced voltage result. The firing angle or timing of a positive voltage pulse through the SCRs gate latches the device open allowing current flow until it drops below the holding current which is usually at or near zero as the AC cycle reverses. (*Figure 8-57*)

SCRs are often used in high voltage situations, such as power switching, phase controls, battery chargers, and inverter circuits. They can be used to produce variable DC voltages for motors and are found in welding power supplies. Often, lighting dimmer systems use SCRs to reduce the average voltage applied to the lights by only allowing current flow during part of the AC cycle. This is controlled by controlling the pulses to the SCR gate and eliminating the heat dissipation caused when using resistors to reduce voltage. *Figure 8-58* depicts the timing of the gate pulse that limits full cycle voltage to the load. By controlling the phase during the time the SCR is latched, a reduced average voltage is applied.

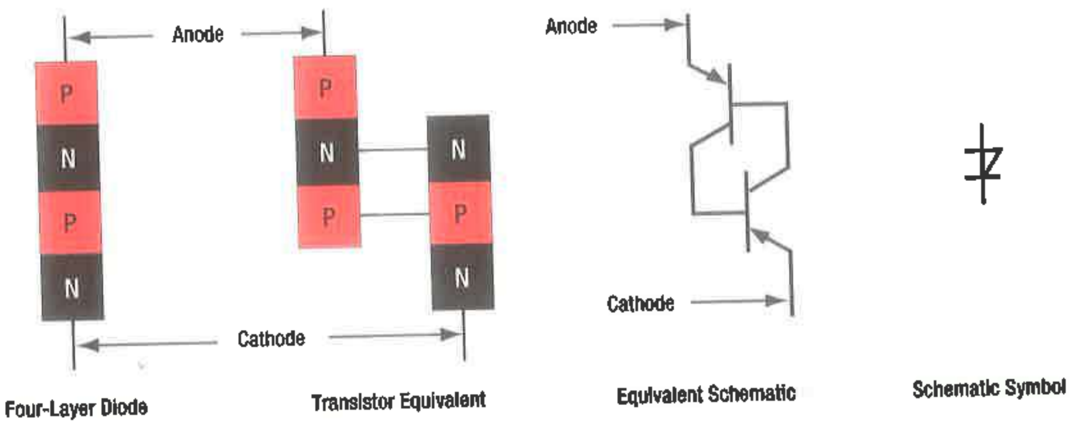


Figure 8-55. A four-layer semiconductor diode behaves like two transistors.

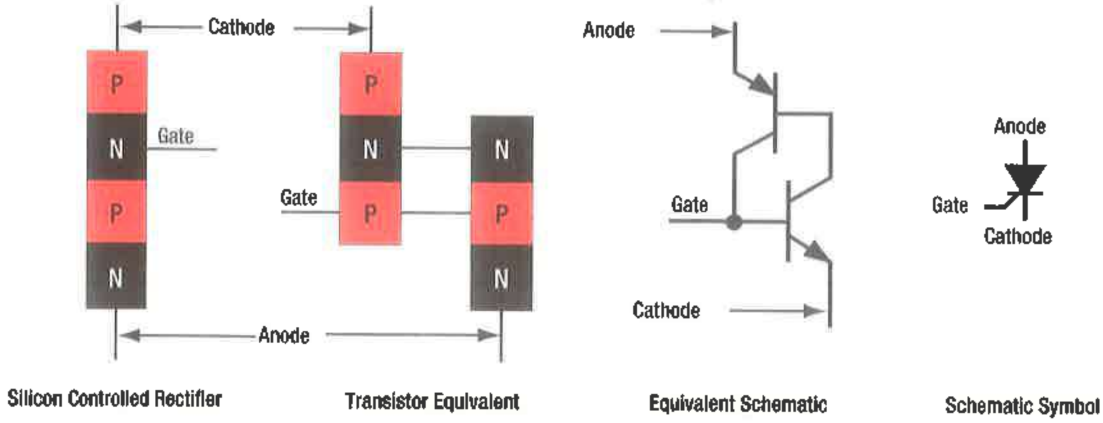


Figure 8-56. A silicon controlled rectifier (SCR) allows current to pass in one direction.

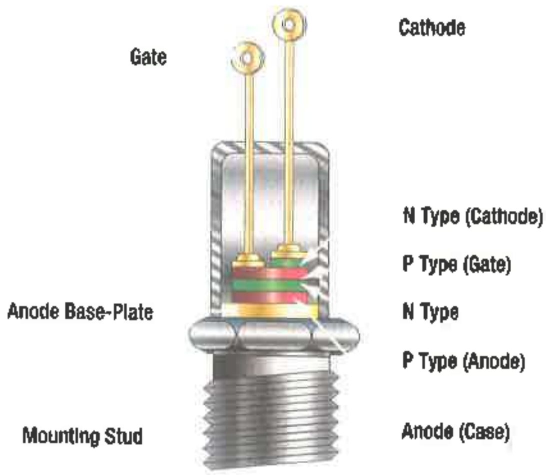


Figure 8-57. Cross-section of a medium-power SCR.

Transformer Rectifiers

Control of the available electric power to numerous electrical devices on an aircraft in any and all situations makes the conversion from AC power to DC power common. Often, Transformer Rectifiers are used for this purpose. They are found between an AC power source and a DC bus and inside battery chargers. The TR not only rectifies AC to produce DC current. It also contains a transformer to adjust the DC output to the precise voltage required.

External/Ground Power

Most aircraft employ an external power circuit that provides a means of connecting electrical power from a ground source to the aircraft. External power is often used for starting the engine or during maintenance

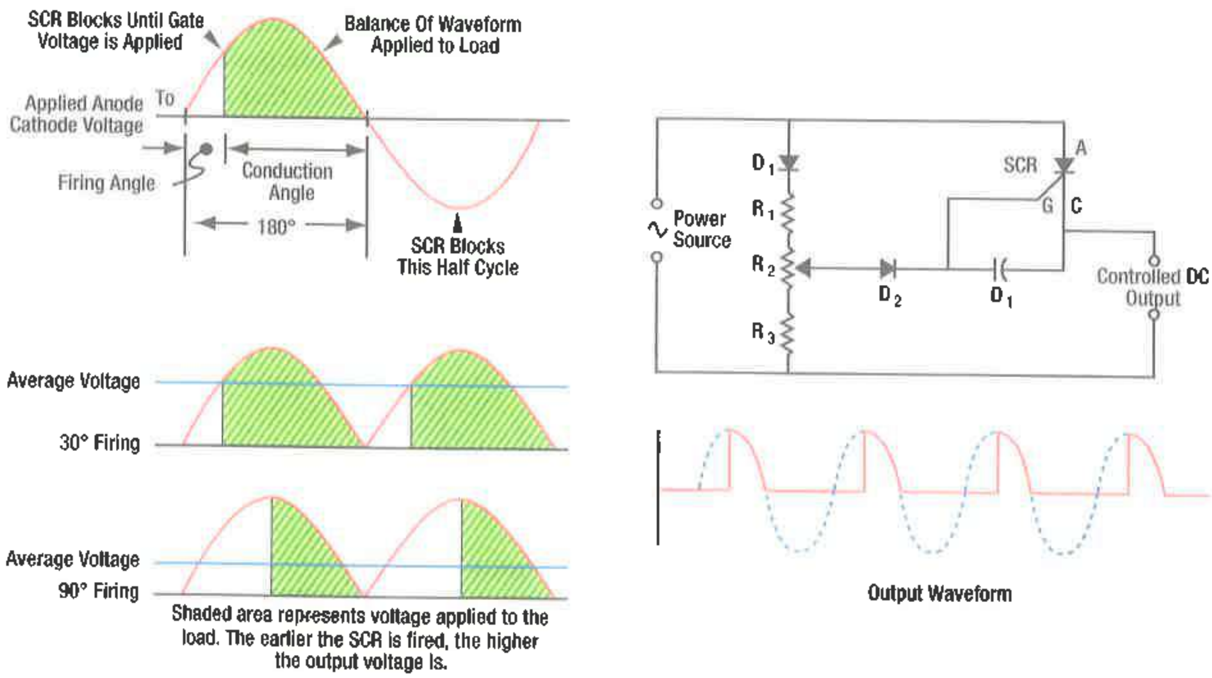


Figure 8-58. Phase control is a key application for SCR.

activities on the aircraft, thus allowing operation of various electrical systems without discharging the battery. External power systems typically consist of an electrical plug located in a convenient area of the fuselage, an electrical solenoid used to connect external power to the bus, and the related wiring. A common external power receptacle is shown in *Figure 8-59*.

Figure 8-60 shows how the external power receptacle connects to the external power solenoid through a reverse polarity diode. This diode is used to prevent any accidental connection in the event the external power supply has the incorrect polarity (i.e., a reverse of the positive and negative connectors).

A reverse polarity connection could be catastrophic to the aircraft electrical system. If a ground power source with a reverse polarity is connected, the diode blocks the current and the external power solenoid does not close. This diagram also shows that external power can be used to charge the aircraft battery or to power the electrical loads. When using external power, the battery master switch must be closed.

On large aircraft, a separate ground handling bus and an APU battery bus are used. Many of these buses divide to power additional buses which distribute power strategically to subsystems of the aircraft.

When the aircraft engines are running, AC generators mounted on and driven by the engines supply the DC buses using Transformer Rectifiers. TRs convert the 115V AC generated into 28V DC power. When the aircraft is on the ground with external power connected, a separate TR converts the AC from the ground power source to 28V DC to power the distribution buses. When the external power source is not used, the main aircraft battery supplies the DC power buses.

Figure 8-59. External power receptacle.

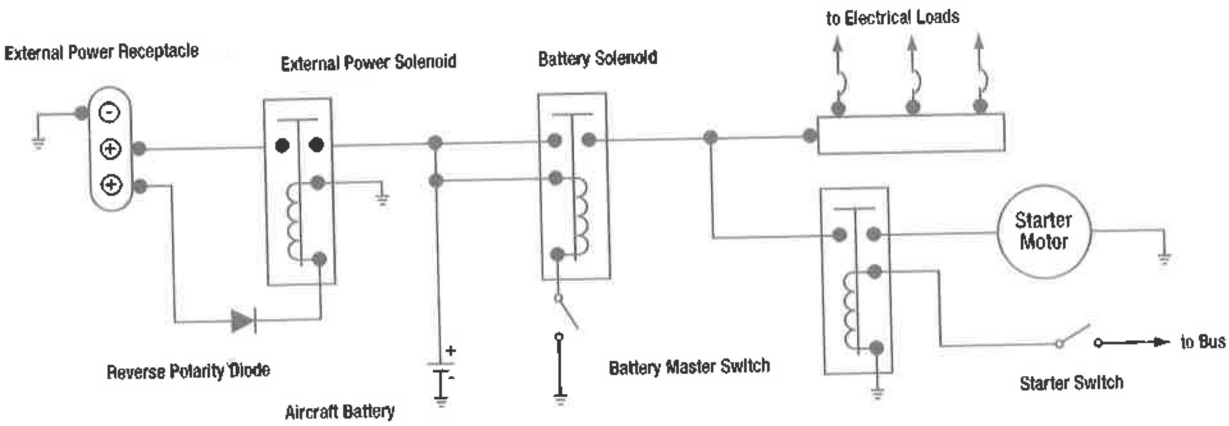


Figure 8-60. A simple external power circuit diagram.

Question: 8-1
What are the two purposes of vents in battery boxes?

Question: 8-2
If a battery has two sets of terminals, what is the purpose of the second smaller set?

Question: 8-3
What device is generally responsible for monitoring the health of the complete electrical system on complex aircraft?

Question: 8-4
What is the primary function of a Constant Field Drive (CFD)?

Question: 8-5
How is the electrical strength output of an alternator controlled?

Question: 8-6
What is the primary (first) source of voltage regulation in a complex aircraft?

Question: 8-7
What are the three basic functions of a Three Unit Regulator?

Question: 8-8
Why are automatic reset breakers not used in aircraft?

ANSWERS

Answer: 8-1
Allow gases to escape and to provide cooling.

Answer: 8-2
To monitor the battery's health.

Answer: 8-3
The Bus Power Control Unit (BPCU).

Answer: 8-4
To maintain the speed (RPM) of the alternator and thus to maintain its 400 Hz frequency output.

Answer: 8-5
By controlling its magnetic field strength.

Answer: 8-6
The Generator Control Unit (GCU).

Answer: 8-7
The voltage regulator section controls generator output; current limiter section limits current; reverse current relay prevents backflow into the generator.

Answer: 8-8
Because the cause of the circuit failure must be checked and corrected before causing additional system damage.

HELICOPTER AERODYNAMICS, STRUCTURES AND SYSTEMS

EQUIPMENT AND FURNISHINGS (ATA 25)

SUB-MODULE 09

PART-66 SYLLABUS LEVELS
CERTIFICATION CATEGORY → B1.3 B1.4

Sub-Module 09
EQUIPMENT AND FURNISHINGS (ATA 25)
Knowledge Requirements

12.9 - Equipment and Furnishings (ATA 25)	
(a) Emergency equipment requirements; Seats, harnesses and belts; Lifting systems;	2
(b) Emergency flotation systems; Cabin layout, cargo retention; Equipment layout; Cabin furnishing installation.	1

12.9 - EQUIPMENT AND FURNISHINGS (ATA 25)

PART A

EMERGENCY EQUIPMENT REQUIREMENTS

Since aircraft leave the surface and fly in the sky, emergency equipment such as fire extinguishers, life rafts, first aid kits, supplemental oxygen, etc., may be required by authorities to be on board for the protection of passengers and crew. Different size aircraft designed for different purposes have emergency equipment requirements suitable for the intended purpose of the aircraft. A primary function of the technician is to ensure all the required equipment is in its specified location and serviceable. These locations are specified in the certification standards of that aircraft.

There are many items on board an aircraft that contribute to safety via designs that are purposeful in an emergency. Among them are:

- Fire Extinguishers
- First-Aid Kits
- Life Jackets
- Emergency Lighting System
- Emergency Locator Transmitter

Life rafts, fire extinguishers and oxygen bottles all have inspection requirements that include a pressure check before each flight. This and other condition inspections are defined and require the technician's signature after checking. These items are installed for a limited period, after which they must be removed for in-depth inspection and recharging. Installation and removal procedures are detailed in the aircraft maintenance manual. The devices must be armed after installation to operate as designed. Technicians must be aware that when working with pressurized rafts and life jackets, specific instructions must be followed to prevent inadvertent deployment and injury.

FIRE EXTINGUISHERS

Fire extinguishers (***Figure 9-1***) are situated throughout the helicopter and are used for dealing with in-flight electric fires. They are mainly of the Halon type which work on most types of fires except liquid. The general information about fire extinguishers is described in *Submodule 10* of this book.

Figure 9-1. A fire extinguisher.

The EASA requirements for hand held fire extinguishers are the following:

- Helicopters shall be equipped with at least one hand fire extinguisher in the flight crew compartment.
- At least one hand fire extinguisher shall be located in, or readily accessible for use in each galley not located in the main passenger compartment.
- At least one hand fire extinguisher shall be available for use in each cargo compartment that is accessible to crew members in flight.
- The type and quantity of extinguishing agent for the required fire extinguishers shall be suitable for the type of fire likely to occur in the compartment where the extinguisher is intended to be used and to minimize the hazard of toxic gas concentration in compartments occupied by persons.
- The helicopter shall be equipped with at least a number of hand fire extinguishers in accordance with ***Table 9-1***, conveniently located to provide adequate availability for use in each passenger compartment.

MAXIMUM OPERATIONAL PASSENGER SEATING (MOPSC)	NUMBER OF EXTINGUISHERS
7-30	1
31-60	2
61-200	3

Table 9-1. Number of handheld fire extinguishers.

FIRST-AID KITS

A first-aid kit (***Figure 9-2***) is a medical kit with different components to provide aid in case of injury during a flight or after landing. It is necessary to frequently

Figure 9-2. A first aid kit.

check the expiration of the various drugs. The EASA requirements for first-aid kits are the following:

- Helicopters shall be equipped with at least one first-aid kit that shall be readily accessible for use and kept up to date.
- A first-aid kit should be equipped with appropriate and sufficient medications and instrumentation. However, these kits should be supplemented by the operator according to the characteristics of the operation (scope of operation, flight duration, number of passengers, etc.).

Maintenance Of First-Aid Kits

To be kept up to date, first-aid kits should be:

- Inspected periodically to confirm that contents are maintained in the condition necessary for their intended use;
- Replenished at regular intervals, in accordance with instructions contained on their labels, or as circumstances warrant;
- Replenished after use at the first opportunity where replacement items are available.

CREW SURVIVAL KITS

The EASA requirements regarding crew survival kits specify that each crew member shall wear a survival suit when operating over water beyond autorotational distance from land, or when the weather report or forecasts indicate that the sea temperature will be less than 10°C during the flight.

Helicopters operated over areas in which search and rescue would be especially difficult shall be equipped with:

- Signaling equipment to make distress signals.
- At least one ELT.
- Additional survival equipment for the route to be flown taking account of the number of persons on board including:
  - 500 ml of water for each 4, or fraction of 4, persons on board.
  - One knife.
  - First-aid equipment.

In addition when polar conditions are expected, the following should be carried:

- A means for melting snow.
- One snow shovel and one ice saw.
- Sleeping bags for 1/3rd of all persons on board and space blankets for the remainder.
- One arctic/polar suit for each crew member.

Life Jackets

A life jacket (***Figure 9-3***) is considered as Personal Protective Equipment. Its main function is to keep the wearer's head above water and keep the airways free, and is considered essential equipment at sea. Life jackets should be worn by all crew members and passengers in the event of over water flight.

In the event of a water landing, the helicopter will not stay on the surface and will dive quickly if not equipped with a flotation system. When the rotor stops rotating due to contact with water, the crew must swim horizontally away from the rotor and inflate the life jacket to ascend and stay on the surface.

Figure 9-3. A life jacket.

A life jacket is not intended to support the entire weight of the person or to keep them out of the water. It only focuses on vertical thrust to support the wearer's head and keep the airways out of the water. Buoyancy is achieved by inflating the jacket with carbon dioxide (CO_2), which is stored under pressure in a canister. Inflation is done manually by pulling on a handle. If there is a lack of pressure in the life jacket, it can be inflated with a mouth inflation valve.

Life jackets should be stored in good temperature and light conditions and protected from wear and tear to be sure they will function properly when needed. They should be frequently checked to ensure that the CO_2 cartridges do not accidentally empty due to a possible leak.

To facilitate rescue operations, life jackets are equipped with an identification light, a battery and a whistle. As an option, the life jackets can also be fitted with fluorescent dyes, shark repellents and special signaling devices. The EASA requirements for life jackets are the following:

- Helicopters shall be equipped with a life jacket for each person on board or an equivalent flotation device for each person on board younger than 24 months.
- Life jackets and other flotation devices must be stowed in a position that is readily accessible from the seat or berth of the person for whose use it is provided, with a safety belt or harness fastened.
- Each life jacket or equivalent individual flotation device shall be equipped with a means of electric illumination for the purpose of facilitating the location of persons.

Life Rafts

Life rafts must be an approved design and radar conspicuous to standard airborne equipment. They must be stowed so as to facilitate their ready use in an emergency. When carrying more than one life raft on board, at least 50% should be able to be deployed by the crew by remote control while seated at their normal station. Those not deployable by remote control should be of a maximum weight to permit handling by one person. (less than 40 kg). Each life raft should contain the following:

- Approved survivor locator light.
- Approved visual signaling device.
- One canopy (for use as a sail, sunshade or rain catcher) or other mean to protect occupants from the elements.
- Radar reflector.
- 20 meter retaining line designed to hold the life raft near the helicopter and releasable if the helicopter becomes totally submerged.
- Sea anchor.
- Survival kit equipped for the route to be flown containing at least:
  - Life raft repair kit;
  - Bailing bucket;
  - Signaling mirror;
  - Police whistle;
  - Buoyant raft knife;
  - Supplementary means of inflation;
  - Sea sickness tablets;
  - First-aid kit;
  - Portable means of illumination;
  - 500 ml of pure water;
  - Sea water desalination kit;
  - Survival booklet in an appropriate language.

EMERGENCY LOCATOR TRANSMITTER (ELT)

Helicopters must be equipped with a minimum of one automatic ELT capable of transmitting simultaneously on 121.5 MHz and 406 MHz. (***Figure 9-4***)

ELT batteries must be replaced (or recharged if the battery is rechargeable) when the equipment has been in use for more than 1 cumulative hour or in the following cases:

- Batteries specifically designed for use in ELTs and having an airworthiness certificate (EASA Form 1 or equivalent) should be replaced or recharged before the end of their useful life in accordance with the maintenance instructions applicable to that ELT.

Figure 9-4. An Emergency Locator Transmitter.

- Batteries not having an airworthiness certificate when used in ELTs should be replaced or recharged when 50% of their useful life has expired.
- The battery useful life (or charge) criteria in (1) and (2) do not apply to batteries (such as water activated batteries) that are essentially unaffected during storage intervals.

The new expiration date for a replaced (or recharged) battery should be legibly marked on the outside of the equipment.

Types of ELTs

The ELT device must be one of the following:

- Automatic Fixed (ELT-AF). An automatically activated ELT that is permanently attached to an aircraft and is designed to aid search and rescue teams in locating the crash site.
- Automatic Portable (ELT-AP). An automatically activated ELT, which is rigidly attached to an aircraft but is readily removable from the aircraft after a crash. It functions as an ELT during the crash sequence. If the ELT does not employ an integral antenna, the aircraft mounted antenna may be disconnected and an auxiliary antenna (stored in the ELT case) attached to the ELT. The ELT can be tethered to a survivor or a life raft.
- Automatic Deployable (ELT-AD). An ELT that is rigidly attached and that is automatically deployed and activated by an impact, or in some cases by water sensors. This type of ELT should float and is intended to aid search and rescue teams in locating the crash site. An ELT-AD may be either a stand alone beacon or an inseparable part of a deployable recorder.

To minimize the possibility of damage in the event of a crash, the automatic ELT should be rigidly fixed to the aircraft structure, as far aft as is practicable, with its antenna and connections arranged so as to maximize the probability of the signal being transmitted after a crash.

EMERGENCY LIGHTING AND MARKING

Helicopters with a Maximum Operational Passenger Seating Configuration of more than 19 shall be equipped with an emergency lighting system having an independent power supply to provide a source of cabin illumination to facilitate the evacuation of the helicopter, and emergency exit markings and location signs visible in either daylight, or in the dark. Additional information on emergency lighting is given in *Sub-Module 08*.

SUPPLEMENTAL OXYGEN

For some helicopters supplemental oxygen equipment may be required. The EASA requirement for supplemental oxygen is that non-pressurized helicopters operated at pressure altitudes above 10 000 ft shall be equipped with supplemental oxygen equipment capable of storing and dispensing the oxygen supplies.

Determination Of Oxygen

The amount of supplemental oxygen for sustenance for a particular operation should be determined on the basis of flight altitudes and flight duration, consistent with the operating procedures, including emergency procedures, established for each operation and the routes to be flown as specified in the operations manual. ***Table 9-2*** is given for use in complex non-pressurized helicopters.

SUPPLY FOR	DURATION AND CABIN PRESSURE ALTITUDE
1. Occupants of flight crew compartment seats on flight crew compartment duty and crew members assisting flight crew in their duties.	The entire flying time at pressure altitudes above 10 000 ft.
2. Required cabin crew members.	The entire flying time at pressure altitudes above 13 000 ft and for any period exceeding 30 minutes at pressure altitudes above 10 000 ft but not exceeding 13 000 ft.
3. Additional crew members and 100 % of passengers(1).	The entire flying time at pressure altitudes above 13 000 ft.
4. 10% of passengers(1).	The entire flying time after 30 minutes at pressure altitudes above 10 000 ft but not exceeding 13 000 ft.

(1) Passenger numbers in Table 1 refer to passengers actually carried onboard including persons younger than 24 months.

Table 9-2. EASA table for complex non-pressurized helicopters.

SEATS, HARNESSES, AND BELTS

SEATS

The manufacturer's seats must adhere to specific rules to receive certification from the regulatory authority. Indeed, this authority must equally approve the interior layouts of the cabin or the cargo holds of a helicopter. Each seat has a use-by date defined during the initial assessment of the supervisory authority. This date is recorded on the Declaration of Design and Performance or specifically underlined in a letter of approval.

In the specifications that the builder must follow, one is the distance between each seat. This distance, defined as minimum pitch, considers the impact zones of the head, trunk and legs to the seat in front, and if necessary, of quickly leaving it in an emergency.

To formalize the minimum acceptable seating standards, normal design extremes are used for certification purposes of all occupied areas. The critical dimension for the seated occupant is the knee length. In addition, the minimum distance and the vertically projected distance between the seats or fixed structure in front of the occupant affects the ease with which the occupant can get up and move from the seat to the main aisle of the cabin. (***Figure 9-5***)

Aircraft seats are covered with flame retardant materials and are individually adjustable back and forth and possibly up and down. Energy attenuating bucket seats are provided for the pilot and front passenger. Energy attenuation occurs when the brittle material inside the back of each seat absorbs energy as the seat descends.

Forward facing seats are made up of seat modules, which are attached to the cabin floor fittings by positive locking clips. Outward facing seat modules can be attached to transmission bulkheads and may have seat support legs that lock into the floor fittings. Outward facing bench seats easily fold up against the bulkhead when not required for passengers to provide additional cabin space for cargo.

Figure 9-5. Helicopter seats.

The EASA requirements for seats, seat belts, restraint systems and child restraint devices are as follows:

- Helicopters shall be equipped with a seat or berth for each person on board who is aged 24 months or more with each seat fitted with a seat belt and each berth fitted with a restraining belt.
- For helicopters first issued a Certificate of Airworthiness (CofA) after 1 August 1999, each safety belt must include an upper torso restraint system for use by each passenger aged 24 months or more; a child restraint device for each person younger than 24 month. A seat belt with an upper torso restraint shall have a single point release.
- Flight crew seats and the seats for the minimum required cabin crew must include two shoulder straps and a seat belt that may be used independently.

EMERGENCY EXIT MARKING

CS-29.811 provides the requirements for emergency exit markings as; Each passenger emergency exit, its means of access, and its means of opening must be conspicuously marked for the guidance of occupants using the exits in daylight or in the dark. Such markings must be designed to remain visible for helicopters equipped for over water flights in case the helicopter becomes capsized and/or the cabin is submerged. The identity and location of each passenger emergency exit must be recognizable from a distance equal to the width of the cabin. (***Figure 9-6***)

The location of each passenger emergency exit must be indicated by a sign visible to occupants approaching along the passenger aisle. There must be locating signs:

- Next to or above the aisle near each floor emergency exit, except that one sign may serve two exits if both exits can be seen readily from that sign.

Figure 9-6. Emergency exit locations showing their banded external markings.

- On each bulkhead or divider that prevents fore and aft vision along the passenger cabin, to indicate emergency exits beyond and obscured by it, except that if this is not possible the sign may be placed at another appropriate location.

Each passenger emergency exit marking and each locating sign must have white letters 25 mm (1 inch) high on a red background 51 mm (2 inches) high, be self or electrically illuminated, and have a minimum luminescence (brightness) of at least 0.51 Candela/m² (160 microlamberts). The colors may be reversed if this will increase the emergency illumination of the passenger compartment.

The location of each passenger emergency exit operating handle and instructions for opening the exit must be shown:

- For each emergency exit, by a marking on or near the exit that is readable from a distance of 0.76 mm (30 inches);
- For each Type I or Type II emergency exit with a locking mechanism released by rotary motion of the handle, by:
  - A red arrow, with a shaft at least 19 mm (3/4 inch) wide and a head twice the width of the shaft, extending along at least 70° of arc at a radius approximately equal to three fourths of the handle length.
  - The word 'open' in red letters 25mm (1 inch) high must placed horizontally near the head of the arrow.

Each emergency exit, and its means of opening, must also be marked on the outside of the helicopter. In addition, the following apply:

- There must be a 51 mm (2 inch) colored band outlining each passenger emergency exit, except small helicopter with a maximum weight of 5 670 kg (12 500 pounds) or less may have a 51 mm (2 inch) colored band outlining each exit release lever or device of passenger emergency exits which are normally used doors.
- Each outside marking, including the band, must have color contrast to be readily distinguishable from the surrounding fuselage surface. The contrast must be such that, if the reflectance of the darker color is 15% or less, the reflectance of the lighter color must be at least 45%. 'Reflectance' is the ratio of the luminous flux reflected by a body to the luminous flux it receives. When the reflectance of the darker color is greater than 15%, at least a 30% difference between its reflectance and the reflectance of the lighter color must be provided.

ESCAPE SLIDES

Helicopters with exits above 1.8 meters above ground must be equipped with slides. Escape Slides are installed at all the helicopter exits to provide quick evacuation for the passengers and the crew in an on-land emergency. The escape slide pressure cylinders are filled with a mixture of carbon dioxide (CO_2) and nitrogen (N).

Slides are inflated automatically. They are built into the door or underneath the door frame. When the door handle is set to ARMED, the girt bar connects the slide to the floor attach fittings. As the door opens, the outboard movement pulls the slide from the slide pack and it begins to fall. A lanyard pulls the reservoir valve to open. The slide inflates fully in approximately 3 to 10 seconds. When used in an emergency water landing, slides can usually be used as rafts. (***Figure 9-7***)

SEAT BELTS AND HARNESSES

CS-29.561 Requirements

Each occupant and each item of mass inside the cabin that could injure an occupant is restrained when subjected to the following ultimate inertial load factors relative to the surrounding structure:

- Upward – 4 g
- Forward – 16 g
- Sideward – 8 g
- Downward – 20g, after the intended displacement of the seat device.
- Rearward – 1.5 g

Figure 9-7. A self deploying emergency exit slide, also suitable as a raft.

Each flight deck and passenger seat is fitted with a seat belt or harness each capable of restraining a person of mass 77 kg (170 pounds).

Passenger Seats

Seat belts and safety harnesses are made from nylon webbing and woven to be extremely strong. They are attached to the entire seat or to a structural part of the aircraft. The seat of each occupant must be equipped with a seat belt or a harness combined with a single trigger point. Typically cargo seats only have lap belts. A typical passenger seat configuration with an over the shoulder harness is shown in ***Figure 9-5***.

If a seat belt is not functional and can not be changed before the flight, the seat may not be used. If not in use, there must be a way to secure the seat belts and harnesses to prevent interference with the operation of the aircraft and for a quick exit if needed.

Crew Seats

The crew seats are usually equipped with a four or five point restraint system that combines a lap safety belt, tie-down strap and double strap shoulder harness with a locking inertia reel. When fastened the seat and harness system must enable the pilot, while seated, to perform all functions required for flight. (***Figure 9-8***)

A special feature of the pilot's belt is an inertia reel which allows forward motion but has an automatic locking mechanism designed to prevent the occupant from tipping forward during rapid deceleration. An adjuster on each shoulder harness permits the pilot or copilot to adjust the tension of the shoulder harness. The lap belts also have an adjustment feature. (***Figure 9-9***)

Some small helicopters may have simple single three point inertia reel seat belts, like in an automobile.

Figure 9-8. A crew seat with a 5 point harness.

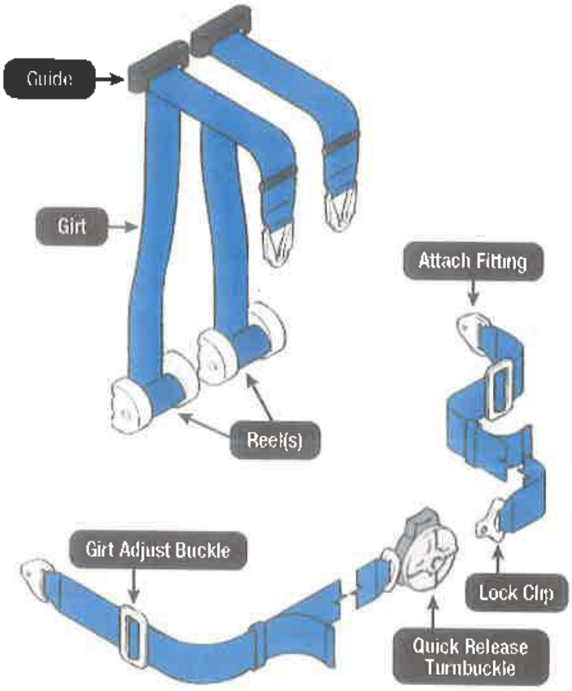


Figure 9-9. A 4 point crew harness with an inertia reel.

LIFTING SYSTEMS

RESCUE HOISTS

In the mountains, above a river or in any difficult access place, the helicopter is the fastest solution in an emergency. The main advantage of the helicopter is that it can remain stationary. In this condition, it is easy to use a winch for different purposes such as the rescue of wounded persons. (***Figure 9-10***) This option on a helicopter can be permanently installed and integrated into the structure of the aircraft or can be removable and installed just when needed. The hoists are installed in the loading area right next to the door or just above the door opening and are powered hydraulically or electrically. (***Figure 9-11***)

Electric Hoists

An electric hoist assembly consists of an electrical motor (AC or DC), a drum and cable assembly and a control unit. The drum and cable assembly is driven by the

Figure 9-10. Hoisting a person into a helicopter.

Figure 9-11. An electric hoist capable of lifting people and other lighter objects.

Figure 9-12. A hoist mechanism rated for 200 Kg (450 pounds).

motor, through a gear train and clutch to raise and lower the usable cable which is terminated with a swivel hook. (***Figure 9-12***)

A fail-safe brake inside the drum engages if electrical power is removed or the load attempts to accelerate faster than the hoist motor. If the load on the hook exceeds a certain limit, an over-tension slipping clutch engages to ensure safe operations. A winding mechanism ensures that the cable is wound evenly and uniformly under all operating conditions.

Constant tension is applied to the cable as it is reeled out. Limit switches sense when the cable has reached its full in or out positions. A second emergency 'full out' limit switch is also provided in case of a failure in the normal 'full out' switch.

A potentiometer, coupled to the drum, continuously senses the cable position for cable payout indications. A cable foul sensor is provided to detect cable mis-wraps. If a mis-wrap occurs, hoisting operations are automatically halted. Two other limit switches (slow-in and slow-out) indicate when automatic deceleration of the hoist is to take place. A backup hoist with a separate and retractable mounting jib is typically provided in the event of a failure of the primary hoist.

Electric Hoist Operation

A hoist may be operated either by the pilot or by crewmen in the cargo area. On the flight deck, two control panels are provided for the primary and for the back-up hoist. The panels located between the seats allows the pilot to enable both the hoist operation and the cable cutter system via a hoist enable switch. A cable cutter switch is also provided. An indicator on the panel indicates when the system is enabled and also if a cable foul has taken place.

The hoist control switches are located on the collective raise or lower the hoist cable operates at a fixed speed. It operates in a regular mode at approximately 200 ft/min and in a slow mode at a speed of 100 ft/min. (***Figure 9-13***)

Hoist controls are also provided for crewmen in the cargo area. Two panels are provided above the cargo door, one for the primary and one for the back-up hoist. These panels allow operation of the hoist at variable speeds using the thumbwheel. A cable payout display indicates the length of cable reeled out. The panel houses three indicator lamps to indicate: when the system is enabled by the pilot, if a cable foul occurs, or if the motor becomes hot. In all cases, the pilot's hoist controls have priority over the crewmen's controls.

Figure 9-13. Hoist controls on the collective lever.

Hover Trim Control
Some helicopters are equipped with a hover trim control to allow a crewman to introduce small corrections into the control system to correct any drifting of the helicopter while hovering. The control is located on a controller's joystick. An additional thumbwheel control allows hoist up and down operations at variable speeds.

Electric Hoist Safety Systems
Safety systems may be activated in the event of a cable foul or an overheated motor.

In the event of a cable foul, the FOUL indication will be provided on both the pilot's and crewmen's control panel and the hoist will cease operations. In the event of a motor overheat, a warning lamp will appear and the hoisting speed will be automatically reduced by approximately 50% until the overheat condition is resolved.

An emergency pyrotechnic cable cutter may also be initiated from the pilot's or crewmen's position. When the Hoist Enable switch is selected ON, the system becomes enabled. Lifting a switch guard and pressing the switch fires the cutter cartridge to release the cable.

Hydraulic Hoists
The advantage of hydraulic generation is to provide more power, thus being able to hoist heavier loads than with an electric hoist, and so be used for a variety of missions in addition to rescue. As with an electric hoist, the pilot and operator control the cable cutting command and only the operator controls the winding and unwinding of the cable.

To get power, a hydraulic motor (*described in Sub-Module 12*) is connected to the general hydraulic power of the helicopter and so converts the hydraulic power into a rotational motion. To change the direction of the cable movement, a selector valve changes the hydraulic fluid flow rate at the same time as the rotation of the hydraulic engine. The problem with this kind of hoist is the possibility of fluid leakage which in this case can drop the pressure. To secure the load, an emergency mechanical brake automatically stops the movement of the cable.

CARGO HOOKS

A cargo hook is used for heavy loads, such as sling loads. This type of lift work is usually very specialized, with special regulations applying. The hook normally has an electrical and a manual release. This is a safety device to ensure that the load may be jettisoned at any time, should difficulties arise.

For example, a Sikorsky S-64 Skycrane is a twin engine heavy lift helicopter with a sling load cargo capability of 20 000 Pounds (9 071 Kgs). (***Figure 9-14***) After electric or manual locking, the helicopter can carry a heavy load such as cargo, water, or various other materials.

Figure 9-14. A Sikorsky Skycrane with a cargo hook.

The cargo hook is fixed to the helicopter on the main frame of the structure and closest to the center of gravity of the helicopter to avoid modifying the balance of the helicopter in flight. (***Figure 9-15***) The major components of a cargo hook systems are:

- Cargo hook;
- Load cell;
- Four-point wire strap assembly;
- Signal processor and load indicator;
- Controls and indications.

The hook assembly consists of the hook which is suspended from a short lateral beam with the hook opening facing to the right. The beam fits into the two structural brackets and is bolted in place. The electrical release is a switch on the pilot's cyclic stick. In the event of an electrical failure, a mechanical release is provided. It is operated by a pull handle centrally located between the front seats. A circuit breaker on the overhead panel completes the kit. A swivel link is sometimes fitted to allow the load to assume a natural stabilized position, with minimum oscillation in flight. Since the hook assembly extends approximately 1 meter beneath the aircraft a hoisting lanyard is provided to enable it to be raised up so that the aircraft can land with the hook clear of the ground.

The difference with the previous hoists, is the possibility for the pilot to work alone and control the hook directly in the cockpit with an information panel. For example, a cockpit indicator displays the weight of the cargo hook load by using microcontroller technology to measure and display the hook load.

Figure 9-15. A Bell Helicopter cargo hook.

The manufacturer defines the various information on load limitations. Each helicopter has different capabilities depending on the power of the engine(s), the size of the rotor(s) and the weight of the helicopter itself.

Four Point Sling Assembly

A four point sling assembly consists of four sheathed wire cables. The wire cable inboard end fitting is a swaged fork fitting and the outboard end fitting is a swaged eye fitting. The outboard fitting is attached to the load cell via shackles. The inboard fitting is attached to the aircraft with assembly fittings and bolts.

Load Cell

The load cell comprised of a load indicator and signal processor is situated in the cabin near the cargo door is the heart of the load weigh system. It is a temperature compensated strain gauge, fitted between the hook and four point sling. The gauge senses the load on the hook by measuring the deflection of the encasing metal. This signal is transmitted to an indicator for display. (***Figure 9-16***)

Although the system should be calibrated annually, the crew can calibrate the system in the field by lifting a known weight that they have entered into the indicator.

Frangible Link

To increase the safety of helicopter long line operations, sometimes a frangible link is included in series with the cargo hook to the line if the load exceeds the capacity of the helicopter. Four different loading scenarios exist that could create an excessive forces.

1. The long line snags something while the helicopter is cruising.

Figure 9-16. A cockpit indicator.

2. The gross weight exceeds the lifting capabilities of the helicopter.
3. In placing or lifting the load it snags on a terrestrial object.
4. A dynamic factor is imparted to the load.

If the load exceeds a preset value, the frangible link separates into two parts and the load releases, thereby preventing damage to the helicopter. A frangible line is commonly used in many systems to protect valuable pieces of equipment by deliberately including a weak member (a "fuse") in series with the load. Shear pins, hydraulic relief valves, and electrical fuses are a few examples in common use. (*Figure 9-17*)

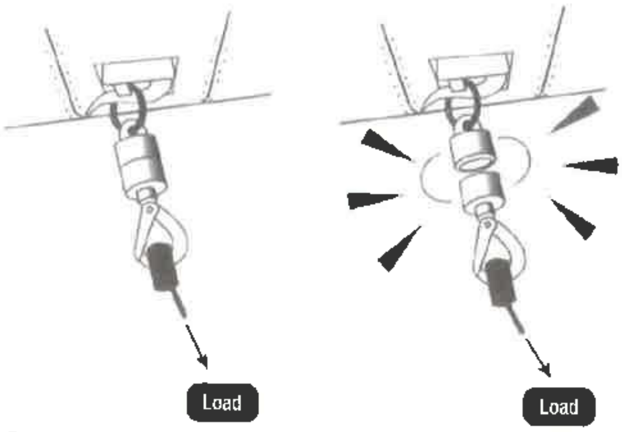


Figure 9-17. A frangible link in series with a cargo hook deliberately releases the line if the load exceeds the capacity of the helicopter.

PART B

EMERGENCY FLOATATION SYSTEMS

When overwater flights are conducted floatation equipment is required. Unless the helicopter is specifically designed for amphibious operations, pop-out floats are often used. The purpose of these floats is to keep the helicopter on the surface of the water for enough time to allow the crew and passengers to exit the aircraft. (*Figure 9-18*)

To activate the emergency flotation system, the pilot can electrically activate the inflation of the emergency float via a switch in the cockpit. Another option for inflating is by a water detector installed under the helicopter to detect submersion. In this case, the gas cartridges are activated and automatically inflate the float allowing the delay of the helicopter sinking. In the event of an electrical malfunction, it is possible for the pilot to activate a mechanical control. A "T" handle marked EMER FLOATS PULL is located on the cockpit floor allows the gas cartridge to be operated mechanically.


Figure 9-18. Emergency floats deployed on a Bell 505.

To avoid accidentally inflating the float, the electrical system must be armed manually with a switch in the cockpit. When this is done, a light informs the pilot that everything is ready to be used in the event of an emergency ditching. A cover protects the handle from accidental jamming and prevents accidental manual activation.

CABIN LAYOUT; CARGO RETENTION

In a helicopter, different cabin arrangements are possible to allow for different roles such as aerial work, external load transport, carrying passengers, search and rescue, etc. In the cockpit, two seats allow the pilot and the copilot to operate the helicopter.

In the case of small helicopters where the passengers are mixed with the crew, three to six seats may be arranged behind the pilot. Seats can be individual or collective, but always with individual seat belts. (*Figure 9-19*)


Figure 9-19. A typical small cabin layout on a Robinson R66.

In medium and large helicopters, a cabin section can have two arrangements; either for passengers ***Figure 9-20*** or without seats for cargo. (***Figure 9-21***)

CARGO RETENTION

When transporting cargo it is critically important that during the flight nothing will move in the cargo area, risking shifting the center of gravity and causing the helicopter to crash, the floor is equipped with tie down rings to which ropes can be attached. (***Figure 9-22***)

A hybrid cargo arrangement with seats allows passengers to travel as on an aircraft. The seating arrangement can be front-to-front, back-to-back or like on an airplane.

EQUIPMENT LAYOUT

All equipment must be strategically positioned in the helicopter for easy access in case of emergency. Some are located with stickers to inform the crew of the position of the equipment. Figure Layout shows neatly and strategically placed items.

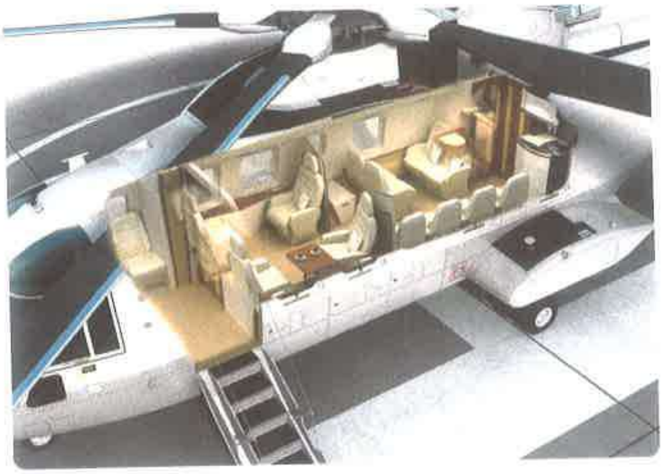


Figure 9-20. A luxury corporate cabin layout.

Figure 9-21. Helicopter cargo layout.

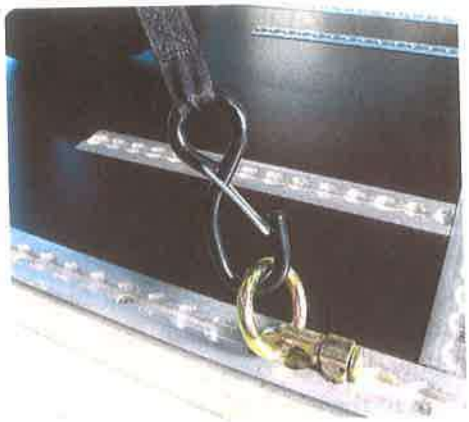


Figure 9-22. Tie-down rings.

The location of the first-aid kit is normally indicated using internationally recognizable signs; typically a red cross. (***Figure 9-23***)

Fire extinguishers are located where electrical fires are most common including in the cockpit attached to the floor near the pilot. In case of large helicopters, a second is installed in the cabin/cargo area. Unless an extinguisher is clearly visible, its location should be indicated by a placard or sign.

Unlike in an airplane a life jacket must be worn at all times during flight over water. This obligation is due to a helicopter's low altitude flight which does not allow time to put one on in case of an emergency.

EASA regulations for helicopters of more than 19 seats require one portable battery powered megaphone readily accessible for use by crew members during an emergency evacuation. The megaphone should be located to be readily accessible at the assigned seat of a crew member assigned with this task.

Figure 9-23. Typical symbols and markings found on a first-aid kit.

CABIN FURNISHING INSTALLATION

All materials used in a passenger cabin must meet the stringent EASA flame resistance requirements and must not emit toxic gases. Most seat cushions and backrest supports are made from a foam like material. When heat is applied to this material, cyanide gas is produced during combustion. To slow this action, the internal parts of the seats are covered with a fire resistant material which is designed to prevent the foam from burning, and therefore allow the occupants a better chance of leaving the helicopter in the event of a cabin fire.

Another important factor is that most fire-resistant qualities are gradually reduced when the seat covers and other materials are dry cleaned. In most cases, manufacturers indicate a maximum number of times that the covers may be dry cleaned before the fire-resistant treatment must be reapplied.

The seat bottoms are normally cushioned and attached with quick attach/detach pins to allow the seat bottom to become an individual flotation device in the event of the helicopter ditching in the sea. Attached to the seat assembly is a seat harness for use by the seat occupant.

How the emergency escape path lighting will be achieved must be decided by the helicopter operators or manufacturers. A popular method in large helicopters is to include small aisle lights in the seat assemblies, which will allow aisle lighting under certain conditions. In medium helicopters, exist lighting will typically be installed at the exit points. (*Figure 9-24*)

Figure 9-24. Emergency egress lighting in a medium size helicopter.

Question: 9-1
For what type of fires, must compatible hand held fire extinguishers be available in helicopters?

Question: 9-2
What is the primary inspection which must be regularly done regarding life jackets?

Question: 9-3
Name the seven items which must be included with a helicopters life raft.

Question: 9-4
In what way must the expiration date of an ELT battery be notated?

Question: 9-5
In what circumstances are all occupants of a helicopter required to wear life jackets?

Question: 9-6
In what circumstances must supplemental oxygen be available in a helicopter?

Question: 9-7
What difference exists between the seat belt system of the flight crew and those of passengers?

Question: 9-8
In which circumstance is a hydraulically operated hoist system preferable to an electric system?

ANSWERS

Answer: 9-1
Halon extinguishers for electrical fires are required.

Answer: 9-2
That its CO_2 cartridges be charged and in good condition.

Answer: 9-3
Locator light, signaling device, sunshade canopy, radar reflector, 20 meters of rope, anchor, and a survival kit.

Answer: 9-4
Written on the outside of the ELT case.

Answer: 9-5
At any time in which the helicopter is not within an autorotational distance to land.

Answer: 9-6
In non-pressurized helicopters when operating above 10 000' altitude.

Answer: 9-7
The system for passengers must be equipped with a single point release mechanism.

Answer: 9-8
When heavy loads must be lifted.

HELICOPTER AERODYNAMICS, STRUCTURES AND SYSTEMS

FIRE PROTECTION (ATA 26)

SUB-MODULE 10

PART-66 SYLLABUS LEVELS
CERTIFICATION CATEGORY → B1.3 B1.4

Sub-Module 10
FIRE PROTECTION (ATA 26)
Knowledge Requirements

12.10 - Fire Protection (ATA 26)	3
Fire and smoke detection and warning systems;	
Fire extinguishing systems;	
System tests.	

12.10 - FIRE PROTECTION (ATA 26)

FIRE PROTECTION

Fire protection systems on modern aircraft include both a fire detection system and a fire extinguishing system. Fire detection may be accomplished in many ways explained below. Fire extinguishing is accomplished with fixed and portable fire agent dispensing systems.

REQUIREMENTS FOR FIRE TO OCCUR

As a refresher from *Module 07*; three things are required to start and maintain a fire. Remove any of these things and the fire extinguishes.

- Fuel: something that will, in the presence of heat, combine with oxygen and thereby releasing more heat, and as a result reducing itself to other chemical compounds.
- Heat: heat accelerates the combination of oxygen with fuel, in turn releases more heat.
- Oxygen: the element which combines chemically with another substance through the process of oxidation. Rapid oxidation, accompanied by a release of heat and light, is called combustion or burning. (*Figure 10-1*)

CLASSES OF FIRES

European Class System

These classes of fire are defined in the European regulation.

- Class A fires involve ordinary combustible materials such as wood, cloth, paper, rubber, and plastics.
- Class B fires involve flammable liquids, such as oils, greases, solvents, and alcohols.
- Class C fires involving flammable gasses such as propane.
- Class D fires involving combustible metals, such as magnesium, sodium and lithium.
- Class E fires involve energized electrical equipment.

Figure 10-1. The fire triangle.

American Class System

These classes of fire are defined in the United States National Fire Protection Association.

- Class A fires involve ordinary combustible materials such as wood, cloth, paper, rubber, and plastics.
- Class B fires involving flammable liquids and flammable gasses.
- Class C fires involving energized electrical equipment.
- Class D fires involving combustible metals, such as magnesium, sodium, and lithium.

FIRE ZONES

Because fire is one of the most dangerous threats to an aircraft, each potential fire zone of a modern aircraft is protected by a fixed fire protection system. A fire zone is an area of an aircraft designated by the manufacturer to require fire detection and/or fire extinguishing equipment along with a high degree of inherent fire resistance. The term "fixed" describes a permanently installed extinguishing system, in contrast to any type of portable fire extinguishing equipment such as a hand held Halon or water fire extinguisher. Typical zones on aircraft that have a fixed fire detection and/or extinguisher system are:

- Engines and Auxiliary Power Units
- Cargo Bays
- Electronics Bays
- Wheel Wells

FIRE PREVENTION

Leaking fuel, hydraulic fluids, deicing, and lubricating fluids can be sources of fire in an aircraft. Such leaks should be noted and corrective action taken when inspecting aircraft systems. Even minute leaks of these fluids are particularly dangerous for they quickly produce an explosive atmospheric condition. Carefully inspect fuel tank installations for signs of external leaks. With integral fuel tanks, the external evidence may occur at some distance from where the fuel is escaping.

Many hydraulic fluids are flammable and should not be permitted to accumulate in the structure. Lagging materials may become highly flammable if soaked with oil of any kind. Any leakage or spillage of flammable fluid in the vicinity of combustion heaters is a serious fire risk, particularly if any vapor is drawn into the heater that passes over the hot combustion chamber.

Oxygen system equipment must be kept absolutely free from traces of oil or grease since these substances spontaneously ignite when in contact with pressurized oxygen. Oxygen servicing cylinders should be clearly marked so they cannot be mistaken for cylinders containing air or nitrogen. Explosions have resulted from this error during maintenance operations.

REQUIREMENTS OF EASA CERTIFICATION SPECIFICATIONS (CS)

CS 27 (small rotorcraft) and CS 29 (large rotorcraft) contain many requirements for fire protection and fire detection including specific design and operational standards for: fire extinguishers, compartment interiors, cargo and baggage compartments, combustion heater fire protection, fire protection of structure, controls, and other parts, flammable fluid fire protection, fuel system crash resistance, fuel jettisoning systems, oil tanks, induction systems, exhaust systems, engine ignition systems, powerplant fire protection, firewalls, fire extinguishing systems and materials, fire extinguishing agents and agent containers, fire detector systems, hand fire extinguishers, electrical system fire and smoke protection, and protection for hydraulic systems.

CS 29.1181 Designated Powerplant Fire Zones Include

- Engine power sections of reciprocating engines;
- Engine accessory sections of reciprocating engines;
- Complete powerplant compartment in which there is no isolation between the engine power section and the engine accessory section;
- Auxiliary power unit compartments;
- Fuel burning heater and other combustion equipment installation described in CS 29.859;
- Compressor and accessory sections of turbine engines;
- Combustor, turbine, and tailpipe sections of turbine engine installations except sections that do not contain lines and components carrying flammable fluids or gases and are isolated from the designated fire zone prescribed in sub-paragraph (a)(6) by a firewall that meets CS 29.1191.

CS 29.1194 Other Surfaces

All surfaces aft of and near engine compartments and designated fire zones, other than tail surfaces not subject to heat, flames, or sparks emanating from a designated fire zone or engine compartment, must be at least fire resistant.

CS 29.1195 Fire Extinguishing Systems

a. Each turbine engine powered rotorcraft and Category A reciprocating engine powered rotorcraft, and each Category B reciprocating engine powered rotorcraft with engines of more than 24 581 cm³ (1 500 cubic inches) must have a fire extinguishing system for each designated fire zone. The fire extinguishing system for a powerplant must be able to simultaneously protect all zones of the powerplant compartment for which protection is provided.

b. For multi engine powered rotorcraft, the fire extinguishing system, the quantity of extinguishing agent, and the rate of discharge must:
   For each auxiliary power unit and combustion equipment, provide at least one adequate discharge; and;
   For each other designated fire zone, provide two adequate discharges.

c. For single engine rotorcraft, the quantity of extinguishing agent and the rate of discharge must provide at least one adequate discharge for the engine compartment.

d. It must be shown by either actual or simulated flight tests that under critical airflow conditions in flight, the discharge of the extinguishing agent in each designated fire zone will provide an agent concentration capable of extinguishing fires in that zone and of minimizing the probability of reignition.

FIRE AND SMOKE DETECTION AND WARNING SYSTEMS

REQUIREMENTS FOR OVERHEAT AND FIRE DETECTION SYSTEMS

Fire detection on aircraft does not rely solely on observation by crew members. Regardless of the type, an ideal fire detector system should include as many of these features as possible:

- No false warnings under any flight or ground condition.

- Rapid indication of a fire and its accurate location.
- Accurate indication of when a fire is out.
- Indication if a fire has re-ignited.
- Continuous indication for the duration of the fire.
- A means to electrically test the detector system from the aircraft cockpit.
- Resistant to damage from exposure to oil, water, vibration, extreme temperatures, or handling.
- Light in weight and easily adaptable to any mounting position.
- Circuitry that operates directly from the aircraft power system without inverters.
- Minimum electrical current requirements when not indicating a fire.
- Cockpit lights that indicate the location of the fire, and with an audible alarm system.
- A separate detector system for each engine.

THERMAL SWITCH SYSTEMS

Many older aircraft have some type of thermal switch or thermocouple system. A thermal switch system has one or more lights energized by the aircraft power system and thermal switches that control the operation of the light(s). These thermal switches are heat sensitive units that complete electrical circuits at a certain temperature. They are connected in parallel with each other but in series with the indicator lights. (*Figure 10-2*)

If the temperature rises above a set value in any one section of the circuit, the thermal switch closes, completing the light circuit to indicate a fire or overheat condition. No set number of thermal switches are required. The exact number and placement is usually determined by the aircraft manufacturer.

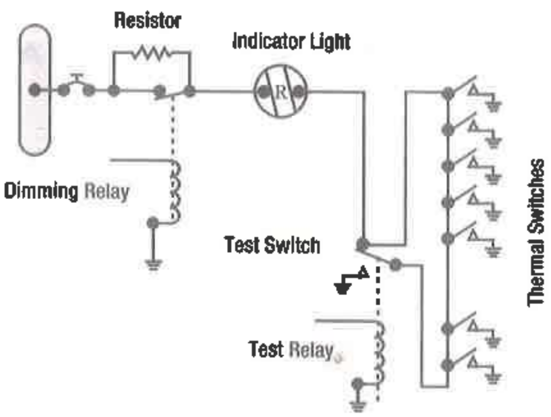


Figure 10-2. Thermal switch fire circuit.

On some installations, all the thermal detectors are connected to one light. On others, there may be one thermal switch for each indicator light. Some warning lights are push-to-test lights. The bulb is tested by pushing it in to check an auxiliary test circuit. The circuit shown in *Figure 10-2* includes this type of test relay. With the relay contact in the position shown, there are two possible paths for current flow from the switches to the light. Energizing the test relay completes a series circuit and checks all the system wiring and the indicator light bulb. Also included in the circuit shown in *Figure 10-2* is a dimming relay. By energizing the dimming relay, the circuit is altered to include a resistor in series with the light. In some installations, several circuits are wired through the dimming relay, and all the warning lights may be dimmed at the same time.

THERMOCOUPLE SYSTEMS

The thermocouple fire warning system operates on an entirely different principle from the thermal switch system. A thermocouple depends on the rate of temperature rise and does not give a warning if an engine slowly overheats, or a short circuit develops. The system consists of a relay box, warning lights, and thermocouples. The wiring system of these units may be divided into the three circuits as shown in *Figure 10-3*.

- Detector Circuit
- Test Circuit
- Alarm Circuit

The relay box contains two relays; the sensitive relay and the slave relay plus a thermal test unit. A single relay box may contain from one to eight identical circuits, depending on the number of potential fire zones. The relays control the warning lights, and in turn, the thermocouples control the operation of the relays. Each circuit consists of several thermocouples in series with each other and its sensitive relay.

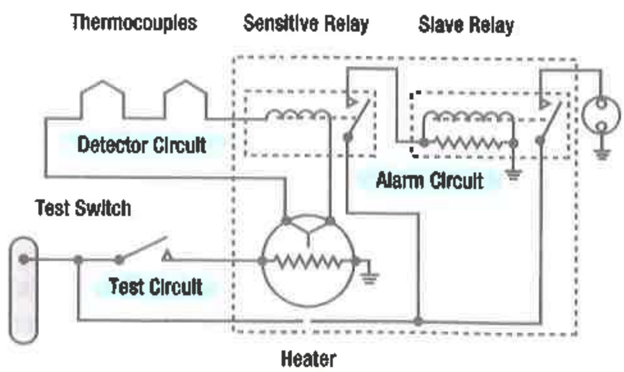


Figure 10-3. Thermocouple fire warning circuit.

Thermocouples are constructed of two dissimilar metals, such as Chromel and Constantan. The point at which these metals are joined and exposed to the heat of a fire is called a hot junction. There is also a reference junction enclosed in a dead air space between two insulation blocks. A metal cage surrounds the thermocouple to give mechanical protection without hindering the free movement of air to the hot junction. If the temperature rises rapidly, the thermocouple produces a voltage from the temperature difference between the reference junction and the hot junction. If both junctions are heated at the same rate, no voltage results.

In an engine compartment, there is a normal and gradual rise in temperature from engine operation. Because it is gradual, both junctions heat at the same rate and no warning signal is given. However, if there is a fire, the hot junction heats more rapidly than the reference junction. The ensuing voltage causes a current to flow within the detector circuit. Any time the current is greater than 4 milliamperes (0.004 ampere), the sensitive relay closes. This completes a circuit from the aircraft power system to the coil of the slave relay.

The slave relay then closes and completes the circuit to the warning light to give a visual fire warning. The total number of thermocouples used in individual detector circuits depends on the size of each fire zone and the total circuit resistance which usually does not exceed 5 ohms. As shown in ***Figure 10-3***, the circuit has two resistors. The resistor connected across the slave relay terminals absorbs the coil self induced voltage to prevent arcing across the points of the sensitive relay. The contacts of the sensitive relay are so fragile that they burn or weld if arcing is permitted.

When the sensitive relay opens, the circuit to the slave relay is interrupted and the magnetic field around its coil collapses. The coil then gets voltage through self induction, but with the resistor across the coil terminals, there is a path for current flow as a result of this voltage, and so eliminating arcing at the sensitive relay contacts.

CONTINUOUS LOOP SYSTEMS

Large aircraft almost exclusively use continuous thermal sensing elements for powerplant protection. These systems offer superior detection performance and coverage, and they have the proven ruggedness to survive in the harsh environment of modern turbofan engines. A continuous loop detector or sensing system permits more complete coverage of a fire hazard area than any of the spot type temperature detectors. Two widely used types of continuous loop systems are the thermistor type detectors, such as the Kidde and the Fenwal systems, and the pneumatic pressure detector, such as the Lingberg system. (Lindberg system is also known as Systron-Donner and, more recently, Meggitt Safety Systems.)

Fenwal System

The Fenwal system uses a slender Inconel tube packed with thermally sensitive eutectic salt and a nickel wire center conductor. (***Figure 10-4***) Lengths of these sensing elements are connected in series to a control unit. The elements may be of equal or varying length and of the same or different temperature settings.

The Fenwal system control unit, operating directly from the power source, applies a small voltage to the sensing elements. When an overheat condition occurs at any point along the element length, the resistance of the eutectic salt within the sensing element drops sharply, causing current to flow between the outer sheath and the center conductor. This current flow is sensed by the control unit, which produces a signal to actuate the output relay and activate the alarms. When the fire has been extinguished or the critical temperature lowered below the set point, the Fenwal system automatically returns to standby alert, ready to detect any subsequent fire or overheat condition. The Fenwal system may be wired to employ a loop circuit. In this case, should an open circuit occur, the system still signals fire or overheat. If multiple open circuits occur, only that section between breaks becomes inoperative.

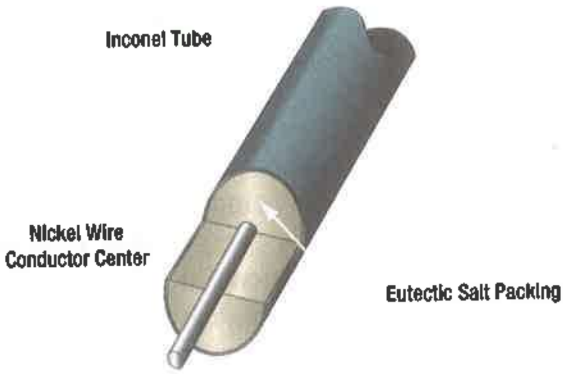


Figure 10-4. Fenwal sensing element.

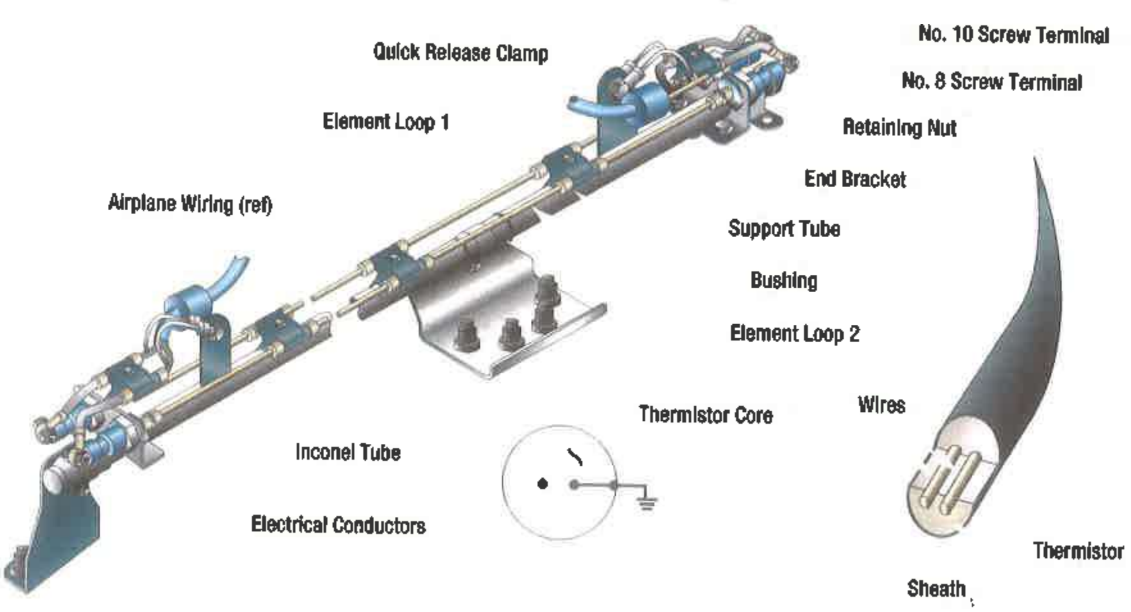


Figure 10-5. Kidde continuous-loop system.

Kidde System

In the Kidde continuous loop system, two wires are embedded in an Inconel tube filled with a thermistor core material. (***Figure 10-5***) The two electrical conductors go through the length of the core. One conductor has a ground connection to the tube, and the other conductor connects to the fire detection control unit. As the temperature of the core increases, electrical resistance to the ground decreases.

The fire detection control unit monitors this resistance. If the resistance decreases to the overheat set point, an overheat indication occurs in the flight deck. Typically, a 10 second time delay is incorporated for the overheat indication. If the resistance decreases more to the fire set point, a fire warning occurs. When the fire or overheat condition is gone, the resistance of the core material in a Kidde detector system increases to the reset point and the flight deck indications disappear. The rate of change of resistance identifies an electrical short or a fire. The resistance decreases quicker with an electrical short than with a fire.

Sensing Elements

The resistance of a sensor varies inversely as it is heated. As a sensor's temperature is increased, its resistance decreases. Each sensor is composed of two wires embedded in thermistor material that is encased in a heavy wall Inconel tube for high strength at elevated temperatures. The electrical connectors at each end of the sensor are ceramic insulated. The Inconel tubes are shrouded in a perforated stainless steel tube and supported by Teflon impregnated asbestos bushings at intervals. The shroud protects the sensor from breakage due to vibration, abrasion against structures, and damage from maintenance activity. The resistance of a sensor also varies inversely with its length, the increments of length being resistances in parallel. The heating of a short length of sensor out of a given length requires that the short length be heated above the temperature alarm point, so the total resistance of the sensor decreases to the alarm point. This characteristic permits integration of all temperatures throughout the length of the installation rather than sensing only the highest local temperature.

The two wires encased within the thermistor material of each Inconel tube form a variable resistance network between themselves, between the detector wire and the Inconel tube, and between each adjacent incremental length of sensor. These variable resistance networks are monitored by the application of 28 volts DC to the detector wire from the detector control unit.

Sensing Element Fault Indication

Provision is made in a control unit to output a fault signal which activates a fault indicator whenever the short discriminator circuit detects an electrical short in the sensing element loop. This is a requirement for transport aircraft because such a short would disable the fire detection system.

Combination Fire And Overheat Warning

The analog signal from the thermistor sensing element permits the control circuits to be arranged to give a two level response from the same sensing element loop. The first is an overheat warning at a temperature level below the fire warning, indicating a general engine compartment temperature rise such as would be caused by leakage of hot bleed air or combustion gas into the engine compartment. It could also be an early warning of fire and would alert the crew to appropriate action to reduce the engine compartment temperature. The second level response is at a level above that attainable by a leaking hot gas and is the fire warning.

Temperature Trend Indication

The analog signal produced by the sensing element loop as its temperature changes is converted to signals suitable for flight deck display to indicate engine bay temperature increases above normal. A comparison of the readings from each loop system also provides a check on the condition of the fire detection system, because the two loops should normally read alike.

Support Tube Mounted Sensing Elements

For those installations where it is desired to mount the sensing elements on the engine, the support tube mounted element solves the problem of providing sufficient element support points and greatly facilitates the removal and reinstallation of the sensing elements for engine or system maintenance.

Most modern installations use the support tube concept of mounting sensing elements for better maintainability, as well as increased reliability. The sensing element is attached to a pre-bent stainless steel tube by closely spaced clamps and bushings where it is supported from vibration damage and protected from pinching and excessive bending. The support tube mounted elements can be furnished with either single or dual sensing elements. Being pre-bent to the designed configuration assures its installation in the aircraft precisely in its designed location where it has the necessary clearance to prevent chafing against the engine. The assembly requires only a few attachment points. Should its removal for engine maintenance be necessary, it is quickly and easily accomplished. Should the assembly require repair or maintenance, it is easily replaced with another assembly, leaving the repair for the shop. Should a sensing element be damaged, it is easily replaced in the assembly.

Dual Loop Systems

Dual loop systems are two complete fire detection systems with their output signals connected so that both must signal to result in a fire warning. This arrangement, called AND logic, results in increased reliability against false fire warnings from any cause. Should one of the two loops be found inoperative at the preflight integrity test, a cockpit selector switch disconnects that loop and allows the signal from the other loop alone to activate the fire warning. Since the single operative loop meets all fire detector requirements, the aircraft can be safely dispatched and maintenance deferred to a more convenient time. However, should one of the two loops become inoperative in flight and a fire subsequently occur, the fire signaling loop activates a cockpit fault signal that alerts the flight crew to select single loop operation to confirm the possible occurrence of fire.

FIRE DETECTION SYSTEM MAINTENANCE

An inspection and maintenance program for all types of continuous loop systems should include the following visual checks. (Note that these procedures are examples and should not be used to replace the manufacturer instructions.) Sensing elements of a continuous loop system should be inspected for the following:

- Cracked or broken sections caused by crushing or squeezing between inspection plates, cowl panels, or engine components.
- Abrasion caused by rubbing of the element on cowling, accessories, or structural members.
- Pieces of safety wire, or other metal particles, that may short the spot detector terminals.
- Condition of rubber grommets in mounting clamps that may be softened from exposure to oils or hardened from excessive heat.
- Dents and kinks in sensing element sections. Limits on the element diameter, acceptable dents and kinks, and degree of smoothness of tubing contour are specified by manufacturers. No attempt should be made to straighten any acceptable dent or kink, since stresses may be set up that could cause tubing failure. (***Figure 10-6***)
- Nuts at the end of the sensing elements should be inspected for tightness and safety wired. (***Figure 10-7***)
- Loose nuts should be re-torqued to the value specified by the manufacturer's instructions. Some types of sensing element connection joints require the use of copper crush gaskets. These should be

replaced any time a connection is separated.

- If shielded flexible leads are used, they should be inspected for fraying of the outer braid. The braided sheath is made up of many fine metal strands woven into a protective covering surrounding the inner insulated wire. Continuous bending of the cable or rough treatment can break these fine wires, especially those near the connectors. Sensing element routing and clamping should be inspected carefully. (***Figure 10-8***) Long, unsupported sections may permit excessive vibration that can cause breakage. The distance between clamps on straight runs, usually about 20-25 cm, is specified by each manufacturer. At end connectors, the first support clamp usually is located about 12 cm from the end connector fittings. In most cases, a straight run of 2-3 cm is maintained from all connectors before a bend is started, and an optimum bend radius of 7-8 cm is normally adhered to.
- Interference between a cowl brace and a sensing element can cause rubbing. This interference may cause wear and short the sensing element.
- Grommets should be installed on the sensing element so that both ends are centered on its clamp. The split end of the grommet should face the outside of the nearest bend. Clamps and grommets should fit the element snugly. (***Figure 10-9***)

FIRE DETECTION SYSTEM TROUBLESHOOTING

The following troubleshooting procedures represent the most common difficulties encountered in engine fire detection systems.

- Intermittent alarms are most often caused by an intermittent short in the detector system wiring. Such shorts may be caused by a loose wire that occasionally touches a nearby terminal, a frayed wire brushing against a structure, or a sensing element rubbing against a structural member long enough to wear through the insulation. Intermittent faults often can be located by moving wires to recreate the short.
- Fire alarms and warning lights can occur when no engine fire or overheat condition exists. Such false alarms can be most easily located by disconnecting the engine sensing loop connections from the control unit. If the false alarm ceases when the engine sensing loop is disconnected, the fault is in

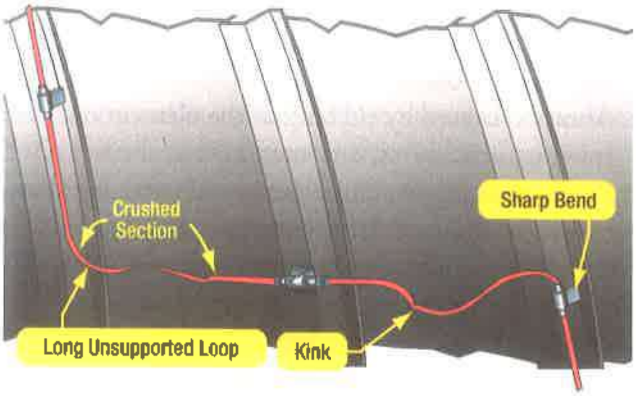


Figure 10-6. Sensing element defects.

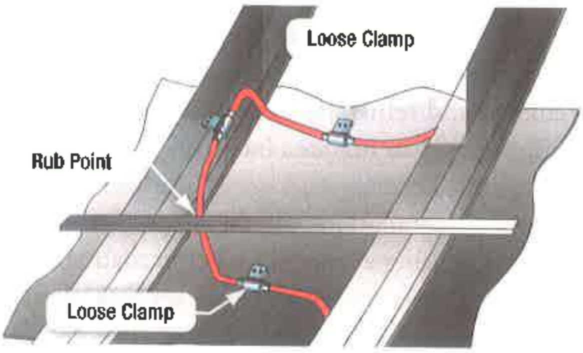


Figure 10-8. Rubbing interference.

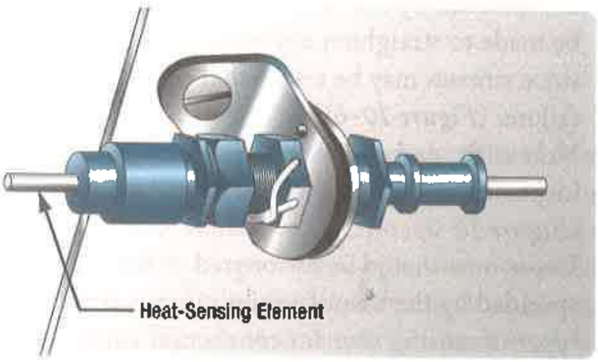


Figure 10-7. Connector joint fitting attached to the structure.

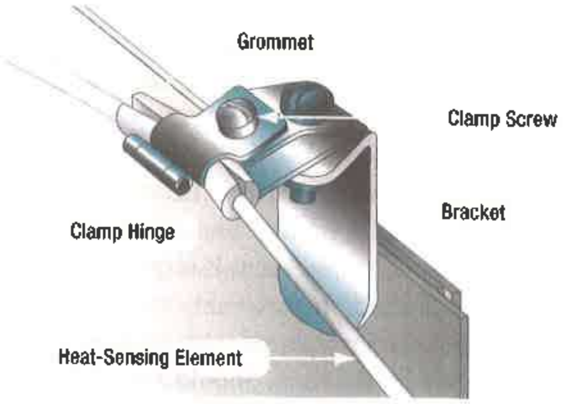


Figure 10-9. Inspection of fire detector loop clamp.

the disconnected sensing loop, which should be examined for areas that have been bent into contact with hot parts of the engine. If no bent element can be found, the shortened section can be located by isolating the connecting elements consecutively around the entire loop.

- Kinks and sharp bends in the sensing element can cause an internal wire to short intermittently to the outer tubing. The fault can be located by checking the sensing element with an ohmmeter while tapping the element in the suspected areas to produce the short.
- Moisture in the detection system seldom causes a false fire alarm. However, if moisture does cause an alarm, the warning persists until the contamination is removed or boils away, and the resistance of the loop returns to its normal value.
- Failure to obtain an alarm signal when the test switch is actuated may be caused by a defective test switch or control unit, the lack of electrical power, an inoperative indicator light, or an opening in the sensing element or connecting wiring. When the test switch fails to provide an alarm, the continuity of a two wire sensing loop can be determined by opening the loop and measuring the resistance. In a single wire continuous loop system, the center conductor should be grounded.

FLAME DETECTORS

Optical sensors, often referred to as flame detectors, are designed to produce an alarm when they detect the presence of prominent specific radiation emissions from hydrocarbon flames. The two types of optical sensors available are infrared (IR) and ultraviolet (UV), based on the specific emission wavelengths that they are designed to detect. IR based optical flame detectors are used primarily on helicopter engines. These sensors have proven to be very dependable and economical for this application. (***Figure 10-10***)

When radiation emitted by the fire crosses the airspace between the fire and the detector, it impinges on the detector front face and window. The window allows a broad spectrum of radiation to pass into the detector where it strikes the sensing device filter. The filter allows only radiation in a tight waveband centered on 4.3 micrometers in the IR band to pass on to the radiation sensitive surface of the sensing device. The radiation striking the sensing device minutely raises its temperature causing small thermoelectric voltages to be generated. These voltages are fed to an amplifier whose output is connected to various analytical electronic processing circuits. The processing electronics are tailored exactly to the time signature of all known hydrocarbon flame sources and ignore false alarm sources, such as incandescent lights and sunlight. Alarm sensitivity level is accurately controlled by a digital circuit.

SMOKE DETECTION SYSTEMS

A smoke detection system monitors the lavatories and cargo compartments for the presence of smoke, which is indicative of a fire condition. Smoke detection instruments that collect air for sampling are mounted

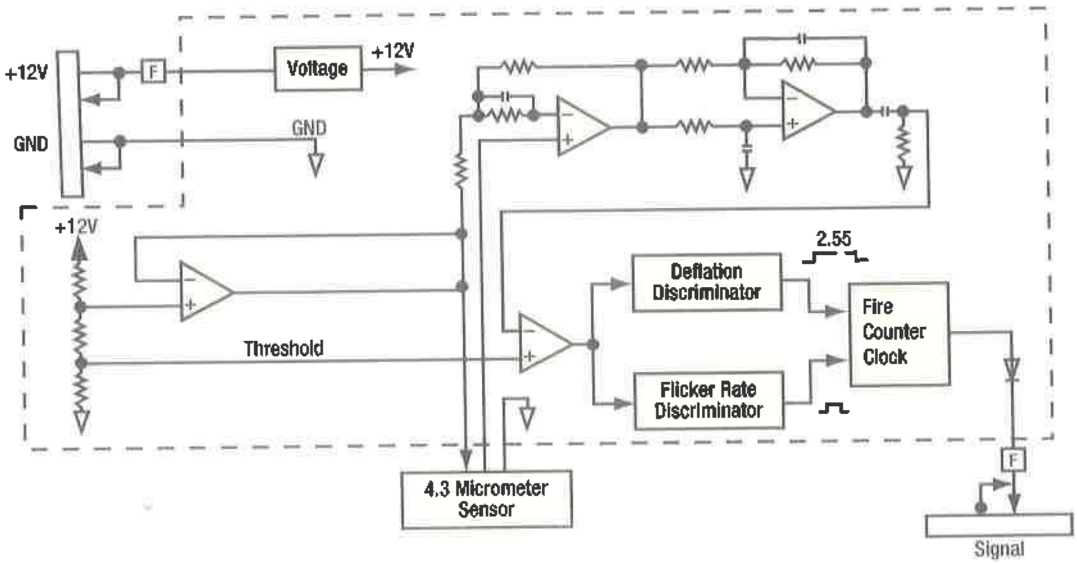


Figure 10-10. Infrared (IR) based optical flame detector.

in the compartments in strategic locations. A smoke detection system is used where the type of fire anticipated is expected to generate a substantial amount of smoke before temperature changes are sufficient to actuate a heat detection system. Two common types used are light refraction and ionization.

Light Refraction Type

The light refraction type of smoke detector contains a photoelectric cell that detects light refracted by smoke particles. Smoke particles refract the light to the photoelectric cell. When it senses enough of this light, it creates an electrical current that sets off a light.

Ionization Type

Some aircraft use an ionization type smoke detector. The system generates an alarm signal (both horn and indicator) by detecting a change in ion density due to smoke in the cabin. The system is connected to 28 volt DC electrical power. Alarm output and sensor sensitive checks are performed simply with the test switch on the control panel.

CARBON MONOXIDE DETECTORS

Carbon monoxide (CO) is a toxic, colorless, odorless and tasteless gas produced from the incomplete combustion of carbon containing materials such as: gasoline, kerosene and other fossil fuels. Piston engine helicopters produce high concentrations of CO that are carried away from the aircraft through the exhaust system. Poor sealing of the cabin, or leaks into the heating or ventilation system from the exhaust, can provide pathways for CO to enter the cabin. While piston engines produce the highest concentration of CO, exhausts from turbine engines also contain CO. Carbon monoxide exposure may lead to accidents due to fatal levels of CO in the blood of the passengers. Exposure to CO can lead to damage to the brain, heart and nervous system.

In order to mitigate this potentially fatal risk, EASA recommends that pilots and operators of aircraft with internal combustion engines or combustion heaters equip the aircraft with a CO detector which must be maintained in accordance with the manufacturer's maintenance instructions. There are many types of CO detectors.

Carbon Monoxide Chemical Detectors

These simple detectors are pieces of cardboard with a small orange colored circle in the middle. If there is a high level of carbon monoxide in the vicinity, the circle changes color from orange to black by a chemical effect. The advantage of such detectors is that they are cheap and provide a basic protection. The disadvantage is that they have no alarm sound and that the users have to keep looking at them to notice if the color has changed. They also have short expiration dates and must be replaced frequently depending on the environment.

Carbon Monoxide Monitors (Sensors)

CO sensors are designed with advanced sensor elements to continuously monitor ambient air for the presence of toxic gas. If detected, the monitor alerts the user to a potentially unsafe exposure with visual, vibrating, and audible alarms.

FIRE EXTINGUISHING SYSTEMS

All types of aircraft store fire extinguishing agents on board to be accessed and deployed manually or automatically by the flight crew if a fire occurs. Storage of a fire extinguishing agent is either in fixed containers, portable containers, or both. Fixed containers are typically spherical in shape and are permanently installed in the aircraft. Apparatus to expel and direct the agent onto a fire is part of a fixed container system. Portable containers are also used. These are stored using quick release latches so that a user may quickly grab a container and hand carry it to the fire for deployment usually by squeezing a trigger type handle.

FIRE EXTINGUISHER AGENTS

Various agents are manufactured and used on aircraft. They are used in both fixed and portable systems. The following is a list of extinguishing agents and the class of fires for which each is appropriate.

Water - Class A

Water deprives the fire of oxygen and cools the material below its ignition temperature. It soaks the burning material to prevent it from igniting again once the fire is extinguished. A water fire extinguisher should only be used on Class A fires. Portable water fire extinguishers may be found in the passenger cabin, however should never be used on electrical fires. Since water is conductive, spraying water on an electrical fire could cause electrocution and will certainly cause damage to the electrical equipment. Note that water fire extinguishers have antifreeze as well as water inside to ensure service should temperatures drop below freezing 0°C.

Carbon Dioxide (CO_2) - Class B or C

Carbon dioxide acts as a blanketing agent. It smothers a fire and deprives it of oxygen. Caution must be exercised when using a CO_2 fire extinguisher in a confined area, as the operator of the CO_2 extinguisher may also be deprived of oxygen. While considered only mildly toxic, it can cause unconsciousness and death by suffocation if the victim is allowed to breathe CO_2 in fire extinguishing concentrations for 20-30 minutes. Because of this, CO_2 is not recommended for hand held fire extinguishers for internal aircraft use.

Carbon dioxide is an effective extinguishing agent. It is most often used in fire extinguishers that are available on the ramp to fight fires on the exterior of the aircraft, such as engine or APU fires. CO_2 has also been used for engine fires on older aircraft. It can extinguish flammable fluid fires and fires involving electrical equipment, although Halon is preferred for electrical fires.

Carbon dioxide is noncombustible and does not react with most substances. It has a boiling point of 78.8°C. As such, it provides its own vapor pressure for discharge from the storage vessel. An exception is in extremely cold climates where a booster charge of nitrogen may be added to winterize the system. Carbon dioxide is about 1.5 times as heavy as air, which causes it to replace air above burning surfaces and maintain a smothering atmosphere. Thus it is effective as an extinguishing agent because it dilutes the air and reduces the oxygen content so that combustion is no longer supported. Under most conditions, some cooling effect is also realized.

CO_2 is not effective on fires involving chemicals containing their own oxygen supply, such as cellulose nitrate (used in some aircraft paints). Also, fires involving magnesium and titanium cannot be extinguished by CO_2. Once used, a carbon dioxide fire extinguisher must be replaced.

Dry Powder Chemicals - Class B, C or D

While effective on Class B and C fires, dry powder extinguishers are the best for use on Class D fires. The method of operation of dry powder extinguishers varies. Some containers use gas cartridge charges or store the agent under pressure within the container to force the powder charge out of the container. Dry powder may also come in a large barrel from which it is applied by hand using a scoop or bucket. Examples of dry powder chemicals are sodium bicarbonate, potassium bicarbonate, and ammonium phosphate.

Dry powder is not recommended for use on aircraft except on metal fires. This is primarily because the leftover chemical residues and dust often make cleanup difficult and can damage electronic or other delicate equipment. As such, dry powder is useful for Class D aircraft wheel and brake fires.

Halogenated Hydrocarbons - Class A, B, or C

Halogenated hydrocarbon (Halon) fire extinguishing agents come in many chemical formulas. Halon 1211 (Bromochlorodifluoromethane) and Halon 1301 (Bromotrifluoromethane) are commonly used in aviation depending on the application. Halon extinguishing agents smother a fire and deprive it of oxygen. They are volatile with part of their effect due to cooling of the burning materials through rapid expansion of the agent. Halon 1301 and 1211 are less toxic than other Halon formulas and are highly effective. They are stored in pressurized containers. Halon 1301 creates satisfactory vapor pressure to expel itself. Halon 1211 has a higher boiling point and may require a nitrogen or a 1301 charge to pressurize adequately for effective discharge.

Halon is extremely effective on a per unit weight basis over a wide range of environmental conditions. It is electrically nonconducting, evaporates rapidly, leaves no residue, and requires no cleanup or neutralization. NOTE: Do not use Halons on a Class D fire as it may react vigorously with the burning metal.

For over 45 years, Halons have been practically the only extinguishing agents used in civil aircraft. However, because Halon is a global warming agent, its production has been banned by international agreement. However, aviation has been granted an exemption because of its unique requirements. Halon replacement agents that are acceptable for environmental protection are now available. Some of these are the halocarbons HCFC Blend B, HFC-227ea, and HFC236fa.

FIXED CONTAINER FIRE EXTINGUISHING SYSTEMS

Most extinguishing systems use perforated tubing or discharge nozzles to distribute the extinguishing agent. High Rate of Discharge (HRD) systems use open end tubes to deliver a large quantity of extinguishing agent in 1 to 2 seconds. The most common extinguishing agent still used today is Halon 1301.

Containers

Most fixed fire extinguishing agent containers on high performance aircraft are the HRD type. They typically store a liquid halogenated extinguishing agent and a pressurized gas (typically nitrogen) to assist in the propulsion of the agent from the container. The containers are normally manufactured from stainless steel or titanium. Most aircraft containers are spherical in design, which provides the lightest weight possible. However, cylindrical shapes are available where space limitations are a factor. Each container incorporates a temperature or pressure sensitive safety relief diaphragm that prevents over-pressurization of the container in the event of exposure to excessive temperatures. (***Figure 10-11 and Figure 10-12***)

Figure 10-11. Built-in non-portable fire extinguisher containers (HRD bottle).

Discharge Valves

Discharge valves are installed on the containers. A cartridge (squib) with a frangible disk valve is installed in the outlet of the discharge valve assembly. Special assemblies having solenoid operated or manually operated seat type valves are also available. Two types of cartridge disk release techniques are used. Standard release types use a slug driven by explosive energy to rupture a segmented closure disc. For high temperature or hermetically sealed units, a direct explosive impact

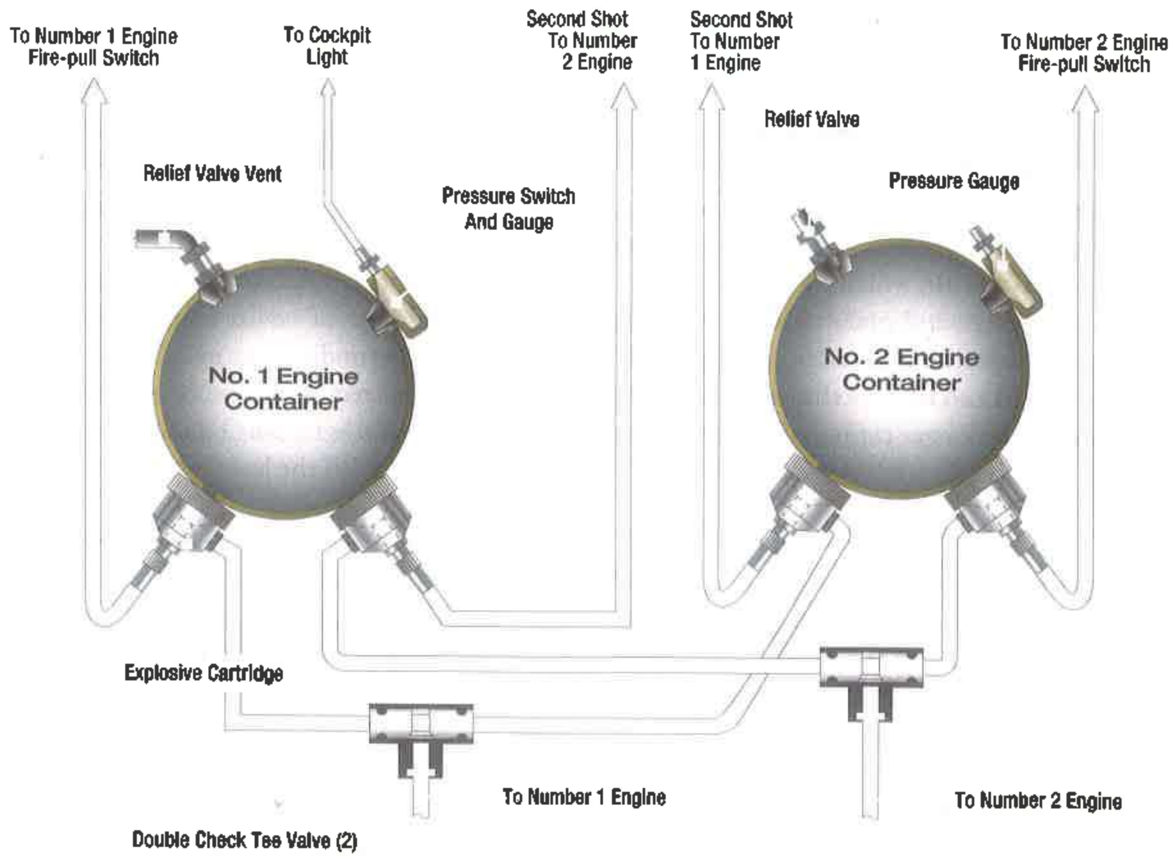


Figure 10-12. Diagram of fire extinguisher containers (HRD bottles.

Figure 10-13. Discharge valve (left) and cartridge, or squib (right).

cartridge is used that applies fragmentation impact to rupture a prestressed corrosion resistant steel diaphragm. Most containers use conventional metallic gasket seals that facilitate refurbishment following discharge. (***Figure 10-13***)

Pressure Indication

A wide range of diagnostics are utilized to verify the fire extinguisher agent charge status. A simple visual indication gauge is available; typically a vibration resistant helical Bourdon type indicator. A combination gauge switch visually indicates actual container pressure and provides an electrical signal if container pressure is lost, precluding the need for discharge indicators. A ground checkable diaphragm type low pressure switch is commonly used on hermetically sealed containers. The Kidde system has a temperature compensated pressure switch that tracks the container pressure variations with temperature by using a hermetically sealed reference chamber.

Two Way Check Valve

Two way check valves are required in a two shot system to prevent the extinguisher agent from a reserve container from backing up into the previously emptied main container.

Discharge Indicators

Discharge indicators provide immediate visual evidence of container discharge on fire extinguishing systems. Two kinds of indicators can be furnished: thermal discharge and normal discharge.

Thermal Discharge Inditator (Red Disk)

The thermal discharge indicator is connected to the fire container relief fitting and ejects a red disk to show when container contents have dumped overboard due to excessive heat. The agent discharges through the opening when the disk blows out. This gives the flight and maintenance crews an indication that the fire extinguisher container needs to be replaced before the next flight. (***Figure 10-14***)

Normal Discharge Indicator (Yellow Disk)

If the flight crew activates the fire extinguisher system, a yellow disk is ejected from the skin of the aircraft fuselage. This is an indication for the maintenance crew that the fire extinguishing system was activated by the flight crew, and the container needs to be replaced before the next flight.

Fire Switch

The engine and APU fire switches are typically installed on the center overhead panel or center console on the flight deck. (***Figure 10-15***) When an engine fire switch is activated, the following happens: the engine stops because the fuel control shuts off, the engine is isolated from the aircraft systems, and the fire extinguishing system is activated. Some aircraft use fire switches that

Figure 10-14. Discharge indicators.

Figure 10-15. Engine and APU fire switches on the cockpit center overhead panel.

need to be pulled and turned to activate the system, while others use a push-type switch with a guard. To prevent accidental activation of the fire switch, a lock is installed that releases the fire switch only when a fire has been detected. This lock can be manually released by the flight crew if the fire detection system malfunctions.

FIRE EXTINGUISHER SYSTEM MAINTENANCE

Regular maintenance of fire extinguisher systems typically includes such items as the inspection and servicing of fire extinguisher bottles (containers), removal and reinstallation of cartridge and discharge valves, testing of discharge tubing for leakage, and electrical wiring continuity tests. The following are some of the most typical maintenance procedures:

Container Pressure Check

Fire extinguisher containers are checked periodically to determine that the pressure is between the prescribed minimum and maximum limits. Changes of pressure with ambient temperature must also fall within prescribed limits. The graph shown in ***Figure 10-16*** is typical of the pressure temperature curve graphs that provide maximum and minimum gauge readings. If the pressure does not fall within the graph limits, the extinguisher container is replaced.

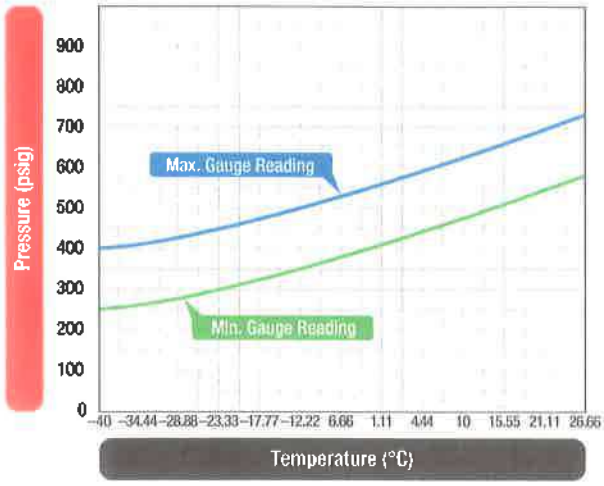


Figure 10-16. Fire extinguisher container pressure-temperature chart.

Discharge Cartridges

The service life of discharge cartridges is calculated from the manufacturer date stamp which is usually placed on the face of the cartridge. The service life recommended by the manufacturer is usually in terms of years; typically 5 years or more. To determine the unexpired service life, it is usually necessary to remove the electrical leads and discharge line from the plug body which can then be removed from the extinguisher container.

Agent Containers

Care must be taken in the replacement of cartridges and discharge valves. Most new extinguisher containers are supplied with their cartridge and discharge valve disassembled. Before installation on the aircraft, the cartridge must be assembled properly with the discharge valve and the valve connected to the container, usually by means of a swivel nut that tightens against a packing ring gasket. (***Figure 10-17***)

If a cartridge is removed from a discharge valve for any reason, it should not be used in another discharge valve assembly, since the distance the contact point protrudes may vary with each unit. Thus, continuity might not exist if a used plug that had been indented with a long contact point were installed in a discharge valve with a shorter contact point.

PORTABLE FIRE EXTINGUISHERS

By regulation there must be a least one hand held portable fire extinguisher for use on the flight deck that is located within easy access of the pilot while seated. In addition, there must be at least one hand held fire extinguisher located conveniently in any passenger cabin

Figure 10-17. Components of fire extinguisher container.

Figure 10-18. A typical aviation Halon extinguisher with a label stating instructions and allowable use.

Figure 10-19. A gauge showing an extinguisher is need of service.

capable of accommodating 6 or more people or 2 in any cabin capable of accommodating 31 or more people. Each extinguisher for use in any personnel compartment must be of the type designed to minimize the hazard of toxic gas concentrations.

PORTABLE EXTINGUISHER TYPES

Several types of portable fire extinguishers are available to fight a fire. The most common types are Halon 1211 and water.

Halon Fire Extinguishers

Halon extinguishers are most common for internal use as they are relatively non-toxic and may be used for a wide range of internal fire possibilities including Class A fires as well as electrical and flammable liquid fires. An additional benefit is that Halon extinguishing agent leaves no residue after discharge. (***Figure 10-18***)

Halon extinguishers are rechargeable. A pressure gauge, typically marked with green or red arcs shows when you must recharge or replace the fire extinguisher. (***Figure 10-19***)

To operate the extinguisher, pull the handle locking pin. Hold the extinguisher upright and squeeze the handle and lever together. Point the nozzle flow at the base of the fire.

Pressurized Water Extinguishers

Pressurized water extinguishers may be used in the cabin to extinguish Class A non-electrical fires. However, they may not be used on the flight deck or in the presence of any electrical equipment or devices.

The water extinguisher is rechargeable. An antifreeze is added to the water to prevent freezing. A carbon dioxide cartridge is mounted within the extinguisher to provide pressurization for discharge of the water. (***Figure 10-20***)

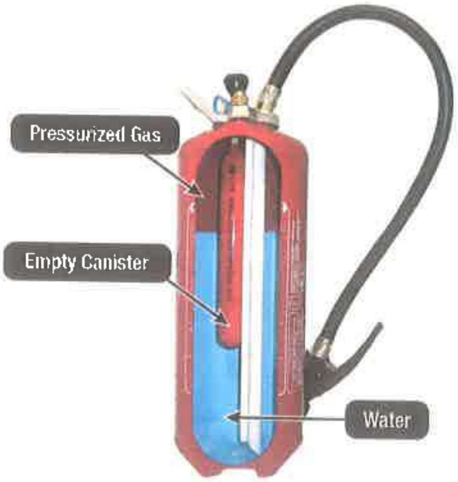


Figure 10-20. A CO_2 gas canister provides pressure to expel water on a water type extinguisher.

Figure 10-21. A maintenance record label notes its service status and filled weight.

To operate the extinguisher, turn the cartridge. This punctures the cartridge and pressurizes the water container. Push the trigger and aim the nozzle flow at the base of the fire

Non-Suitable Extinguishers

CO_2 (carbon dioxide) is not suitable for use inside the flight deck or passenger cabin due to its toxicity. Dry Chemical type extinguishers are also not suitable for internal use due to their corrosive effects on electrical equipment and possible visual obscuration if used on the flight deck.

MAINTENANCE AND INSPECTION

The pre-installation checks, installation procedures and inspections of portable extinguishers may vary between types but in general the following points are common:

- Before placing an extinguisher in its stowage brackets, it should be inspected for general condition and signs of fluid leakage. In addition, its weight and service information should be checked for currency as typically noted on its label or an attached tab. (***Figure 10-21***)
- The expiration dates of extinguishers should be checked against the date of manufacture to ensure they are within the specified service life. Extinguishers having expendable containers should be fitted with new containers at the time expired date.
- In certain types of water extinguishers, safety pins are provided to lock the triggers when the extinguishers are in transit. Such pins must be removed before installation of the extinguishers.
- Copper tell-tale wire and any other tamper-proof seals, should be checked to ensure that they are intact. If the wire and seal of an extinguisher have been broken it must be withdrawn from service for a weight check.
- Where a dust cap is provided on the discharge nozzle of an extinguisher, check to ensure that the cap is free to be forced off should the extinguisher be needed. If overly tight, the nozzle should be smeared with a light application of silicone grease.

SYSTEM TESTS

CONTINUOUS LOOP FIRE DETECTION SYSTEM

The integrity of a continuous loop fire detection system may be tested by actuating a test switch on the flight deck. This switches one end of the sensing element loop from its control circuit to a test circuit built into the control unit which simulates the sensing element resistance change due to fire. (***Figure 10-22***)

If the sensing element loop is unbroken, the resistance detected by the control circuit is that of the simulated fire, and the alarm is activated. The test demonstrates, in addition to the continuity of the sensing element loop, the integrity of the alarm indicator circuit and the proper functioning of the control circuits. The thermistor properties of the sensing element remain unchanged for the life of the element as no irreversible changes take place when heated. The element functions properly as long as it is electrically connected to the control unit.

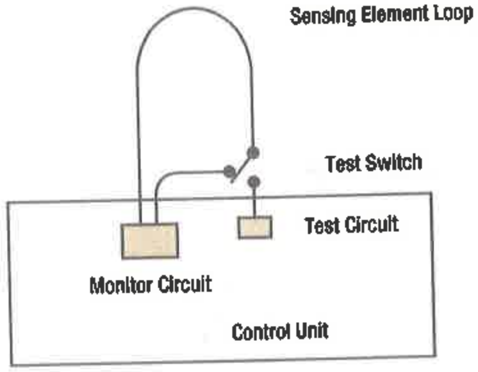


Figure 10-22. Continuously loop fire detection system test circuit.

Dual Loop Systems Automatic Self Interrogation

Dual loop systems automatically perform the loop switching and decision making functions required of the flight crew upon appearance of the fault indication in the cockpit. This function is known as automatic self interrogation. Automatic self interrogation eliminates the fault indication and assures the immediate appearance of a fire indication should fire occur while at least one of the dual loop systems is operative. Should the control circuit from a single loop signal activate, the self interrogation circuit automatically tests the functioning of the other loop. If it tests operative, the circuit suppresses the fire signal because the operative loop would have signaled if a fire existed. If however, the other loop tests inoperative, the circuit outputs a fire signal. The interrogation and decision take place in milliseconds, so that no delay occurs if a fire really exists.

Fire Detection Control Unit (Fire Detection Card)

The control unit for the simplest type of system typically contains the necessary electronic resistance monitoring and alarm output circuits housed in a hermetically sealed aluminum case fitted with a mounting bracket and electrical connector. For more sophisticated systems, control modules are employed that contain removable control cards with circuitry for individual hazard areas and/or unique functions. In the most advanced applications, the detection system circuitry controls all aircraft fire protection functions, including fire detection and extinguishing for engines or APUs.

Question: 10-1
Which types of fire extinguishing agents are usable on electrical fires inside a helicopter?

Question: 10-2
What are the 4 principle designated fire zones on an aircraft?

Question: 10-3
What is the principle difference between a Fenwall and Kidde type loop system?

Question: 10-4
What is the basic operating principle of continuous loop fire detectors?

Question: 10-5
What is the principle goal of the design of support brackets for lines of fire detection sensors?

Question: 10-6
Where on a helicopter will you most likely find an Infrared optical flame detector?

Question: 10-7
While conducting a walk-around of a helicopter, you notice a yellow disc protruding from the airframe. What does this indicate?

Question: 10-8
While inspecting a fixed fire extinguisher container, while all looks intact and leak free, you notice a slight change of pressure from the previous day. What is the likely cause?

ANSWERS

Answer: 10-1
Halon and Halocarbons only.

Answer: 10-2
Engine compartments, cargo bays, electronic bays, wheel wells.

Answer: 10-3
The Fenwall sensing element has one imbedded electrical conductor, while the Kidde system uses two.

Answer: 10-4
The electrical resistance of the sensing elements decreases as heat is applied causing current to flow.

Answer: 10-5
To prevent the lines from breaking or chafing on the structure due to vibrations.

Answer: 10-6
In the engine compartment.

Answer: 10-7
The flight crew has discharged an extinguisher container.

Answer: 10-8
The outside ambient temperature has changed.

HELICOPTER AERODYNAMICS, STRUCTURES AND SYSTEMS

FUEL SYSTEMS (ATA 28)

SUB-MODULE 11

PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY → B1.3 B1.4

Sub-Module 11

FUEL SYSTEMS (ATA 28)

Knowledge Requirements

12.11 - Fuel Systems (ATA 28) 3

System layout;

Fuel tanks;

Supply systems;

Dumping, venting and draining;

Cross-feed and transfer;

Indications and warnings;

Refueling and defuelling.

12.11 - FUEL SYSTEMS

INTRODUCTION

All powered aircraft require fuel on board to operate the engine(s). A fuel system consisting of storage tanks, pumps, filters, valves, fuel lines, metering and monitoring devices must be designed and certified under the strict guidelines of EASA. Each system must provide an uninterrupted flow of contaminant free fuel regardless of the aircraft attitude. Since fuel load can be a significant portion of the aircraft's weight, a sufficiently strong airframe must be designed. Varying fuel loads and weight shifts during maneuvers must not negatively affect control of the aircraft in flight. Although the technician is rarely involved with designing fuel systems, an overview of these requirements gives insight into how an aircraft fuel system operates.

EASA Requirement CS 29.951
General
Each fuel system must be constructed and arranged to ensure fuel flow at the rate and pressure established for each engine and APU under all likely operating conditions, including the maneuvers for which certification is requested.

Fuel systems must be arranged so that no engine or pump may draw fuel from more than one tank at a time and to prevent the introduction of air into the system. Each fuel system for a turbine engine must be capable of sustained operation throughout its flow and pressure range, with fuel initially saturated with water at 27°C, and having 0.20 cm³ of free water per liter added and cooled to the most critical condition for icing likely to be encountered in operation.

EASA Requirement CS 29.953
Fuel System Independence
For Category A Rotorcraft: The powerplants must be arranged and isolated from each other to allow operation, so that the malfunction of any engine, or any fuel system component that can affect one engine will not prevent the continued safe operation of the remaining engine(s).

For Multi Engine Category B Rotorcraft: Each fuel system must meet the requirement of the above, however separate fuel tanks need not be provided for each engine.

EASA Requirement CS 29.954
Fuel System Lightning Protection
The fuel system must be designed and arranged to prevent the ignition of fuel vapor within the system by:

- Direct lightning strikes to areas having a high probability of stroke attachment.
- Swept lightning strokes to areas where swept strokes are highly probable.
- Corona and streaming at fuel vent outlets.

EASA Requirement CS 29.959
Unusable Fuel Supply
The unusable fuel supply for each tank must be established as not less than the quantity at which the first evidence of malfunction occurs under the most adverse fuel feed condition occurring under any intended flight maneuvers involving that tank.

EASA Requirement CS 29.961
Fuel System Hot Weather Operation
Each suction lift fuel and other fuel systems conducive to vapor formation must be shown to operate within certification limits when using fuel at its most critical temperature for vapor formation.

FUEL SYSTEM LAYOUT

Helicopter fuel systems vary, and can be simple or complex depending on each aircraft. Always consult the manufacturer's maintenance manual for complete fuel system information on the aircraft on which maintenance is being performed. Each aircraft fuel system must store and deliver clean fuel to the engine(s) at a pressure and flow rate able to sustain operations regardless of the operating conditions of the aircraft. Fuel systems can be regarded as these subsystems:

- Storage
- Venting
- Distribution
- Feed
- Indicating

GRAVITY FEED SYSTEMS

Simple gravity feed systems have vented fuel tanks with an outlet strainer and shutoff valve. Fuel flows from the tank through a main filter to the carburetor. (***Figure 11-1***)

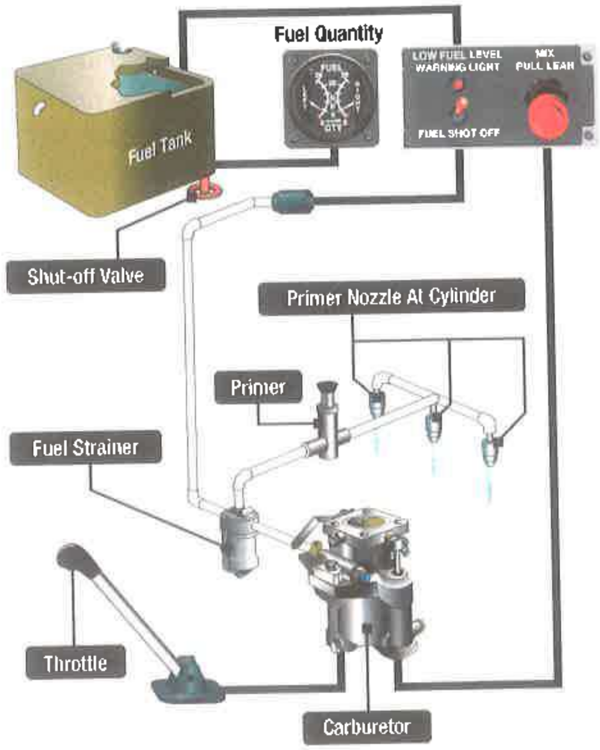


Figure 11-1. A gravity feed fuel system.

PRESSURE FEED FUEL SYSTEM

A slightly more complex system for a light turbine powered helicopter is shown in *Figure 11-2*. Since there is only one fuel tank, two in-tank electric boost pumps send fuel through a shutoff valve rather than a selector valve. Fuel flows through an airframe filter, to an engine filter and then to the engine driven fuel pump. The fuel tank is vented and contains an electrically operated sump drain valve. A pressure gauge is used to monitor boost pump output pressure and differential pressure switches warn of fuel filter restrictions. Fuel quantity is monitored through the use of two in-tank fuel probes with transmitters.

COMPLEX FUEL SYSTEMS

Larger, heavy, multiengine transport helicopters have complex fuel systems similar to jet transport fixed wing aircraft. They may feature multiple fuel tanks, cross feed systems, and pressure refueling.

FUEL TANKS

EASA Requirement CS 29.963
Fuel Tanks - General

a. Each fuel tank must be able to withstand the vibration, inertia, fluid, and structural loads to which it may be subjected in operation.
b. Each flexible fuel tank bladder or liner must be approved as suitable for the particular application and must be puncture resistant.
c. Each integral fuel tank must have facilities for inspection and repair of its interior.
d. The maximum exposed surface temperature of all components of the fuel tank must be less by a safe margin than the lowest expected auto ignition temperature of the fuel or fuel vapor in the tank.
e. Each fuel tank installed in personnel compartments must be isolated by fume and fuel proof enclosures that are drained and vented to the exterior of the rotorcraft. The design and construction of the enclosures must provide necessary protection for the tank, must be crash resistant during a survivable impact, and must be adequate to withstand loads to be expected in personnel compartments.

EASA Requirement CS 29.967
Fuel Tank Installation

a. Each fuel tank must be supported so that tank loads are not concentrated on unsupported tank surfaces.
b. Spaces adjacent to tank surfaces must be adequately ventilated to avoid accumulation of fuel or fumes in those spaces due to minor leakage. If the tank is in a sealed compartment, ventilation may be limited to drain holes that prevent clogging and that prevent excessive pressure resulting from altitude changes. If flexible tank liners are installed, the venting arrangement for the spaces between the liner and its container must maintain the proper relationship to tank vent pressures for any expected flight condition.
c. The location of each tank must be isolated from the engines by a firewall and there must be at least 13 mm of clear airspace between each tank or reservoir and each firewall isolating a designated fire zone, unless equivalent means are used to prevent heat transfer from the fire zone to the flammable fluid.
d. No rotorcraft skin immediately adjacent to a major air outlet from the engine compartment may act as the wall of an integral tank.

EASA Requirement CS 29.969
Fuel Tank Expansion Space

Each fuel tank or group of tanks with interconnected vent systems must have an expansion space of not less

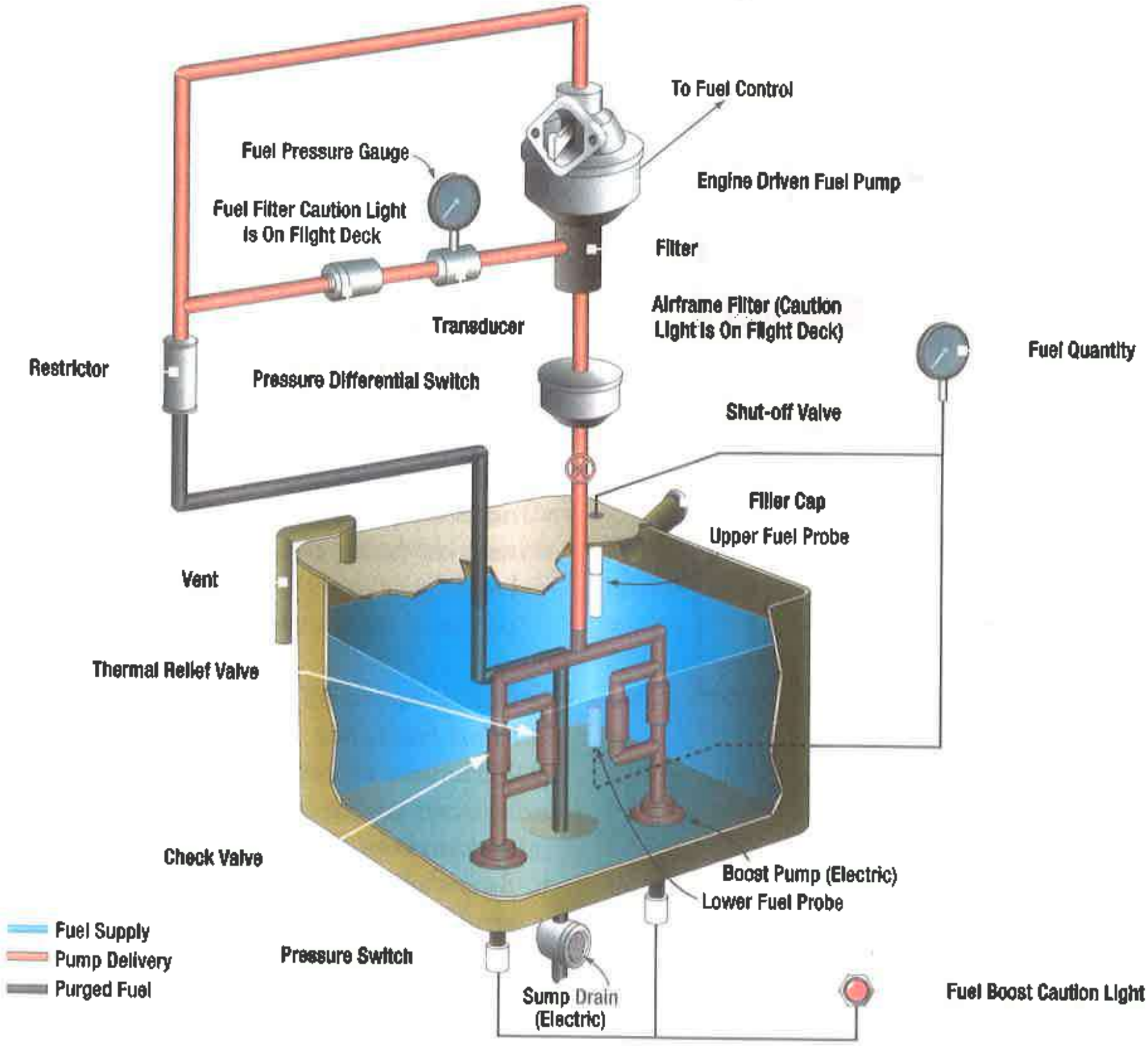


Figure 11-2. A pressure feed fuel system.

than 2% of the combined tank capacity. It must be impossible to fill the expansion space inadvertently with the rotorcraft in the normal ground attitude.

EASA Requirement CS 29.971
Fuel Tank Sump

a. Each fuel tank must have a sump with a capacity of not less than the greater of 0.10% of the tank capacity, or 0.24 liters and be effective with the rotorcraft in any normal attitude, and located so that the sump contents cannot escape through the tank outlet opening.
b. Each fuel tank must allow drainage of hazardous quantities of water from each part of the tank to the sump with the rotorcraft in any ground attitude expected in service.
c. Each fuel tank sump must have a drain to allow complete drainage of the sump on the ground.

EASA Requirement CS 29.973
Fuel Tank Filler Connection

a. Each fuel tank filler connection must prevent the entrance of fuel into any part of the rotorcraft other than the tank itself and must be crash resistant during a survivable impact. Each recessed filler that can retain any appreciable quantity of fuel must have a drain that discharges clear of the entire rotorcraft. Each filler cap must provide a fuel tight seal under the fluid pressure expected in normal operation and in a survivable impact.
b. Each filler cap or filler cap cover must warn when the cap is not fully locked or seated on the filler connection.

EASA Requirement CS 29.977
Fuel Tank Outlet

a. There must be a fuel strainer for the fuel tank outlet or for the booster pump. For reciprocating engines, this strainer must have 3 to 6 meshes per cm. For turbine engine powered rotorcraft, the strainer must prevent the passage of any object that could restrict fuel flow.

b. The clear area of each outlet strainer must be at least five times the area of the outlet line.

c. The diameter of each strainer must be at least that of the fuel tank outlet.

d. Each finger strainer must be accessible for inspection and cleaning.

GENERALITIES ON FUEL TANKS

Typically, a helicopter has only one or two fuel tanks located near the Center of Gravity (CG) of the aircraft, which is near the main rotor mast. Thus, the tank, or tanks, are usually located in or near the aft fuselage. Some helicopter fuel tanks are mounted above the engine allowing for gravity fuel feed. Others use fuel pumps and pressure feed systems.

Small rotorcraft are usually single engine types with a single tank, unlike large helicopters which have multiple engines and need separate tanks for redundant safety. Unlike transport airplanes which store fuel directly in the sealed wing or central structure, large rotorcraft have bladder tanks.

Integral fuel tanks must have access panels for inspection and repairs of the tanks or other system components. When performing maintenance on an integral fuel tank, all fuel must be emptied from the tank and strict safety procedures must be followed. Fuel vapors must be purged from the tank and respiratory equipment must be used. A full time lookout must be positioned just outside of the tank to assist if needed.

Aircraft using integral fuel tanks have sophisticated fuel systems including in-tank boost pumps. Usually at least two pumps are in each tank to deliver fuel to the engine(s) under positive pressure. On various aircraft, these in-tank boost pumps are also used to transfer fuel to other tanks. Additional or rigid auxiliary tanks can be added in the cargo area to increase the flight range.

FUEL TANK CONSTRUCTION

The location, size, shape, and construction of fuel tanks vary with the type and intended use of the helicopter. Some fuel tanks are equipped with dump valves that make it possible to jettison fuel during flight in order to reduce the weight of the helicopter to its specified landing weight. Some helicopters are also equipped with auxiliary fuel tanks. (*Figure 11-3*)

Integral Fuel Cells

Integral tanks have the advantage of using the maximum amount of space for the fuel and of having a minimum amount of weight. All of the rivets and nut plates are sealed, and sealant is used around all of the inspection openings. The sealant is spread along each seam individually rather than sloshing the entire tank. (*Figure 11-4*)

Bladder Fuel Tanks

A fuel tank made up of a reinforced flexible material, known as a bladder tank, can be used instead of a rigid tank. Flexible bladders are constructed of a multi-layered sandwich type lamination of rubber, nylon, and woven fabric. The cell liner material is a gum rubber inner liner and a high strength rubberized fabric outer ply for tear and puncture resistance. (*Figure 11-5*)

Figure 11-3. An auxiliary fuel tank.

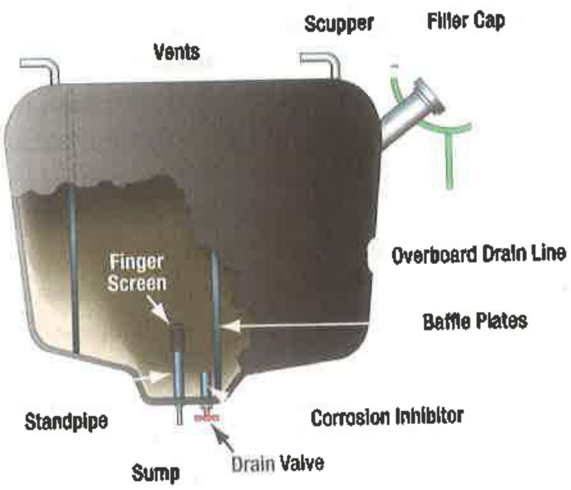


Figure 11-4. A rigid removable fuel tank.

A bladder tank contains most of the features and components of a rigid tank but does not require as large an opening in the aircraft skin to install. The fuel bay is prepared by covering all sharp edges of the metal structure with a chafe-resisting tape. The bladder can then be rolled up and put into the prepared bay through a small opening such as an inspection port. Once inside, it can be unfurled to its full size. Bladder tanks must be attached to the structure with clips or other fastening devices. They should lie smooth and unwrinkled in the bay. It is especially important that no wrinkles exist on the bottom surface so that fuel contaminants are not blocked from settling into the tank sump. An opening in the bladder is then secured to the inspection opening and is covered with an inspection plate.

Figure 11-5. An aircraft bladder fuel tank.

Bladder fuel tanks are used on aircraft of all sizes. They are strong and have a long life with seams only around features such as vents, sump drains, filler spouts, etc. If a bladder tank develops a leak, a patch can be made according to the manufacturer's instructions. The cell can also be removed and sent to a repair station familiar with and equipped to perform such repairs. (*Figure 11-6*)

Figure 11-6. Bladder fuel tanks, each with its own filler, placed in compartments within an aircraft structure.

Bladder tanks should never be allowed to stand empty for an extended period of time. Their soft, flexible nature requires that they remain wet. Should it become necessary to store a bladder tank empty of fuel for an extended period, it is common to wipe the inside of the tank with a coating of clean engine oil. Follow the manufacturer's instructions for dry storage procedures for fuel cells.

FUEL TANK REPAIR

Whether bladder type or integral, all fuel tanks have the potential to develop leaks. General notes for repair of each tank type follow. Note that at the time a tank is repaired, a thorough inspection should be made. Corrosion, such as that caused by water and microbes should be identified and treated at this time, even if it is not the cause of the leak.

Integral Tanks

Occasionally, an integral tank develops a leak at a seam or access panel. This can often be repaired by transferring fuel to another tank so the panel can be removed, and then the seal replaced. Use of the proper sealing compound and bolt torque are required. Other integral fuel tank leaks can be more challenging and time consuming to repair. They occur when the sealant used to seal the tank seams loses its integrity. To repair, fuel first needs to be transferred out of the tank. A checklist for fuel tank preparation and detailed procedures are given in the manufacturer's maintenance manual. Once the location of the leak is determined, the old tank sealant is removed, and new sealant is applied. Remove old sealant with a nonmetallic scraper. Aluminum wool can be used to remove the final traces. After cleaning the area with the recommended solvent, apply new sealant as instructed by the manufacturer. Observe cure time and leak checks before refilling the tank.

Bladder Tanks

Bladder tanks that develop leaks can also be repaired. Most commonly they are patched using patch material and adhesive as approved by the manufacturer. As with integral tanks, a patch has a required overlap of the damaged area. A damage completely penetrating through the bladder is repaired with an external plus an internal patch. Synthetic bladder tanks have a limited service life. At some point, they will seep fuel beyond acceptable limits and need to be replaced. It is important to ensure that bladder tanks are correctly secured in place with the proper fasteners when reinstalling them in the aircraft after a repair.

SUPPLY SYSTEMS

The fuel system in a helicopter is made up of two components: supply and control. Fuel is supplied from the tanks to the engines and APU through a distribution system consisting of fuel lines that connect various valves, pumps, heat exchangers, and indication system components. These fuel feed, cross feed and transfer systems are all part of the distribution system.

FUEL FEED

The fuel feed system is the heart of the fuel system as it supplies the engines with fuel. Large helicopters supply fuel to the engines with pressure pumps connected directly to the engine that are located on the top of the helicopter at the front or rear of the main gearbox. To help the fuel rise from the tanks that are below the structure, booster pumps are directly installed to push and facilitate the fluid flow. For each engine, pumps pressurize the fuel through a shutoff valve. A manifold or connecting tube generally allows any tank to feed any engine using cross feed valves. Boost pump bypass valves allow fuel to flow in the event of a pump failure. Check valves allow fuel to flow only in the correct direction to the engines. Note that the engines are designed to run without a booster pump running. However, each fuel shutoff valve must be on open to allow any flow to the engines from the tanks.

Non-Pressurized Fuel Systems

The supply system consists of a fuel tank or tanks, fuel quantity gauges, a shut-off valve, fuel filter, a fuel line to the engine, and possibly a primer and fuel pumps. (***Figure 11-1***) The tanks are usually mounted to the airframe as close as possible to the CG. Thus, as fuel is burned off, there is a negligible effect on the CG. A drain valve located on the bottom of the tank allows the pilot to drain water and sediment that may have collected in the tank. A fuel vent prevents the formation of a vacuum in the tank, and an overflow drain allows fuel to expand without rupturing the tank. The fuel travels from the tank through a shut off valve, which provides a means to completely stop fuel flow to the engine in the event of an emergency or fire. The shut off valve remains in the open position for all normal operations. The schematic shown in ***Figure 11-7*** depicts the fuel system of a Bell 205 and is typical of medium sized helicopters.

Fuel is contained in interconnected cells, located near the helicopter's center of gravity to eliminate the need for fuel management. Gravity transfers fuel between the cells automatically, eliminating the need for transfer pumps. The cells contained within the fuselage are enclosed by structural panels which often form the rear seat base, internal bulkheads, and the outer skin. On helicopters incorporating stub wings, additional fuel cells are contained within them. These outboard cells are surrounded by fiberglass panels and aluminum alloy skin. Each fuel compartment is lined with a flexible bladder. Interconnect and vent lines between the wing cells and those in the fuselage are designed to break away and self seal to prevent fuel spillage should the wing become separated from the fuselage. Fuel lines within the fuselage are designed to stretch up to 50%.

In normal operation, each engine draws fuel from its own tank by the engine driven suction pumps. The two under the seat tanks are isolated from one another to insure that one fuel leak will not affect both engines. An interconnect valve assures that all fuel is available for single engine operation. An electrical fuel shutoff valve for each engine is located on the forward firewall. A thermal relief valve permits the return of fuel to the tanks after shutdown, removing internal pressure from the engine fuel controls. Each feed tank incorporates a sump drain and prime pump assembly.

The primer pumps are used only during engine starting to purge air from the fuel system, and are deactivated when the start cycle has been completed. The sump drains are electrically activated by environmentally sealed push button switches located low on the fuselage, just ahead of the wings and also serve as manual defuel valves. The two boost pumps are interconnected to supply fuel through a common line. These pumps are typical submersible centrifugal pumps. Each is identical and provided with a single line to the engine. Before these lines join, a check valve is located in each, so one pump does not cycle fuel through the other in case of pump failure. Either pump can supply sufficient fuel. A pump failure would immediately be indicated by a pressure switch located in each outlet port.

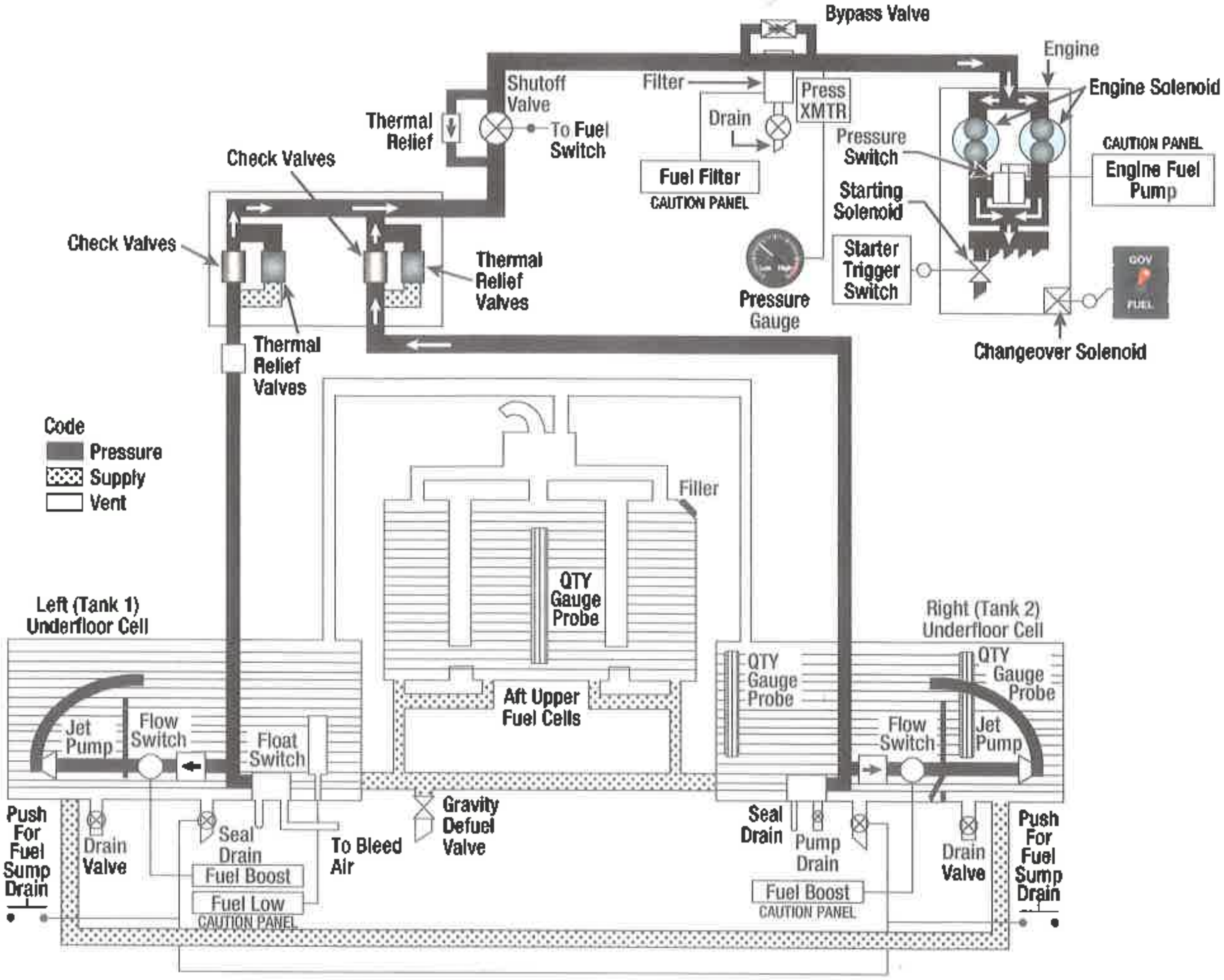


Figure 11-7. A three tank system of a Bell 205 with their various pathways, vents, valves, etc.

An electrically operated shutoff valve is incorporated into the single line. The fuel then passes through a fuel filter unit equipped with a bypass caution light which operates by pressure differential. The fuel then proceeds to the engine filter before entering the engine. Additional provisions are made in the system for a fuel pressure gauge, vent system, and fuel quantity indicator.

Pressurized Fuel Systems

Most non-gravity feed fuel systems contain both an electric pump and a mechanical engine driven pump. The electrical pump is used to maintain positive fuel pressure to the engine pump and may also serve as a backup in the event of mechanical pump failure. The electrical pump is controlled by a switch in the cockpit. The engine driven pump is the primary pump that supplies fuel to the engine and operates any time the engine is running. A fuel filter removes moisture and other sediment from the fuel before it reaches the engine. (***Figure 11-8***)

Figure 11-8. (A) An engine driven fuel pump from Eaton Aerospace. (B) An electric fuel boost pump from Crane Aerospace.

FUEL SYSTEM COMPONENTS

EASA Requirement CS 29.993

Fuel System Lines And Fittings

a. Each fuel line must be installed and supported to prevent excessive vibration and to withstand loads due to fuel pressure, valve actuation, and accelerated flight conditions.

b. Each fuel line connected to components of the rotorcraft between which relative motion could exist must have provisions for flexibility.

c. Each flexible connection in fuel lines that may be under pressure or subjected to axial loading must use approved flexible hose assemblies.

d. No flexible hose that might be adversely affected by high temperatures may be used where excessive temperatures will exist during operation or after engine shutdown.

Lines And Fittings

As seen above, even fuel lines and fittings have standards to ensure proper operation. Aircraft fuel lines can be rigid or flexible depending on their location and application. Rigid lines are often made of aluminum alloy and are connected by standardized fittings. However, in the engine compartment, wheel wells, and other areas which are more subject to damage from debris, abrasion and heat, stainless steel lines are often used.

Flexible hoses are used in areas where vibration exists between components, such as between the engine and the aircraft structure. Flexible fuel hoses have a synthetic rubber interior with a reinforcing fiber braid covered by a synthetic exterior. (*Figure 11-9*) Some have a braided stainless steel exterior. (*Figure 11-10*)

Sometimes manufacturers additionally wrap both flexible and rigid fuel lines to provide even further protection from abrasion and from fire. A fire sleeve cover is held over the line with steel clamps at the end fittings. (***Figure 11-11***)

The diameters of all hoses and lines are determined by the fuel flow requirements of the aircraft fuel system. In all cases, use only fuel line which is approved for fuel. No other hose should be substituted.

Aircraft fuel line fittings are usually standardized types. Both flared and flareless fitting are used. Problems with leaks at fittings can occur. Technicians are cautioned to not over tighten a leaky fitting. In case the proper torque does not stop a leak, depressurize the line, disconnect the fitting, and visually inspect it for a cause. The fitting or line should be replaced if needed. Replace all aircraft fuel lines and fittings with approved replacement parts from the manufacturer. If a line is manufactured in the shop, only approved components may be used.

Several installation procedures for fuel hoses and rigid fuel lines exist. Hoses should be installed without twisting. The writing printed on the outside of the hose is used as a lay line to monitor fuel hose twist. Separation should be maintained between all fuel hoses and electrical wiring. Never clamp wires to a fuel line. When separation is not possible, always route the fuel line below any wiring. In this way, if a fuel leak develops, it does not drip onto the wires. Metal fuel lines and all aircraft fuel system components need to be electrically bonded and grounded to the aircraft structure. This is important because fuel flowing through the fuel system

Figure 11-9. A typical flexible aircraft fuel line.

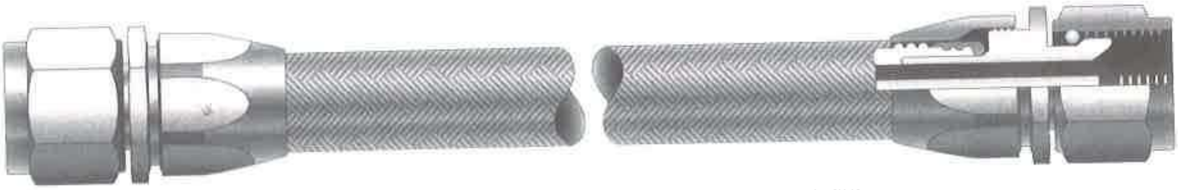
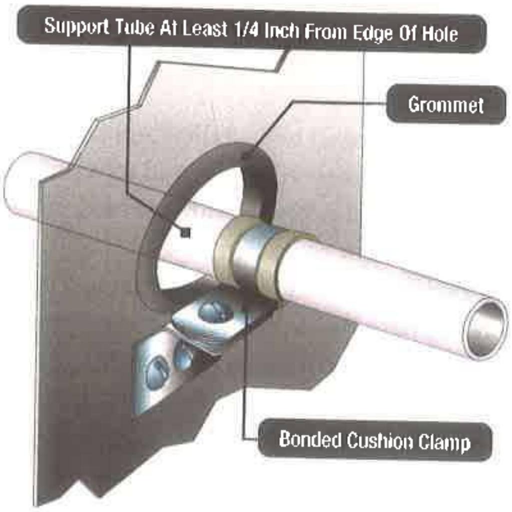


Figure 11-10. A braided stainless steel exterior fuel line with fittings.

Figure 11-11. Exterior fuel hose wrap.



Tubing OD (Inch)	Approximate Distance Between Supports (Inches)
1/8 to 3/16	9
1/4 to 5/16	12
3/8 to 1/2	16
5/8 to 3/4	22
7 to 1-1/4	30
1-1/2 to 2	40

Figure 11-12. Rigid metallic fuel lines.

generates static electricity that must have a place to flow to ground rather than build up. Special bonded cushion clamps are used to secure rigid fuel lines in place. They are supported at intervals shown in ***Figure 11-12***.

All fuel lines should be supported so that there is no strain on the fittings. Clamp lines so that fittings are aligned. Never draw two fittings together by threading. They should thread easily, and a wrench should be used only for tightening the threads. Additionally, a straight length of rigid fuel line should not be made between two components mounted to the airframe. A small bend in the line is needed to absorb any strain from vibration or expansion and contraction due to temperature changes.

FUEL VALVES

EASA Requirement CS 29.1189
Fuel Shutoff Means

a. There must be means to shut off or otherwise prevent fuel, oil, deicing fluid, and other flammable fluids from flowing into or through any designated fire zone.
b. The closing of any fuel shutoff valve for any engine may not make fuel unavailable to the remaining engines.
c. For Category A rotorcraft no quantity of flammable fluid may drain into any designated fire zone after shutoff has been accomplished.
d. The operation of any shutoff may not interfere with the emergency operation of any other equipment, such as the means for declutching the engine from the rotor drive.
e. Each shutoff valve and its control must be designed, located, and protected to function properly under any condition likely to result from fire in a designated fire zone.
f. Except for ground-use-only APU installations, there must be means to prevent inadvertent operation of each shutoff and to make it possible to reopen it in flight after being closed.

EASA Requirement CS 29.995
Fuel Valves
In addition to meeting the requirements of shutoff means, each fuel valve must be supported so that no loads resulting from their operation or from accelerated flight conditions are transmitted to the lines attached to the valve.

Generalities On Fuel Valves

There are many types of fuel valves used in aircraft fuel systems. They are used to shut off fuel flow or to route the fuel to a desired location. Most simply open and close and are known by different names related to their location and function in the fuel system (e.g., shutoff valve, transfer valve, cross feed valve). Fuel valves can be manually operated, solenoid operated, or operated by electric motor. A feature of all aircraft fuel valves is that they have a means for always identifying the position of the valve. Motor and solenoid operated valves use annunciator lights to indicate valve position in addition to the switch position. Flight Management System (FMS) fuel pages also display the position of the fuel valves graphically in diagrams called up on the flat screen monitors. NOTE: Many valves have an exterior lever that indicates the valve's position. Maintenance personnel can manually position this valve using this same lever. (***Figure 11-13***)

Figure 11-13. This motor-operated gate valve has a red position indicating lever.

Manual Valves

On each engine fuel circuit, manually operated gate valves can be used, particularly as fire control valves, requiring no electrical power to shut off the fuel flow when the emergency fire handle is pulled. The valves are positioned in the fuel supply line of each engine. ***Figure 11-14*** shows an example of a handle located on the top of the cockpit, easily recognizable with its black and yellow colors.

Figure 11-14. A manual fuel cut-off valve handle.

Gate valves use a sealed door or blade that slides in the path of the fuel, blocking its flow when closed. ***Figure 11-15*** shows a typical manual gate valve. When the handle is rotated, the actuator arm inside the valve moves the valve blade down between the seals and into the fuel flow path. A thermal bypass valve is incorporated to relieve excess pressure build-up against the closed valve due to temperature increases.

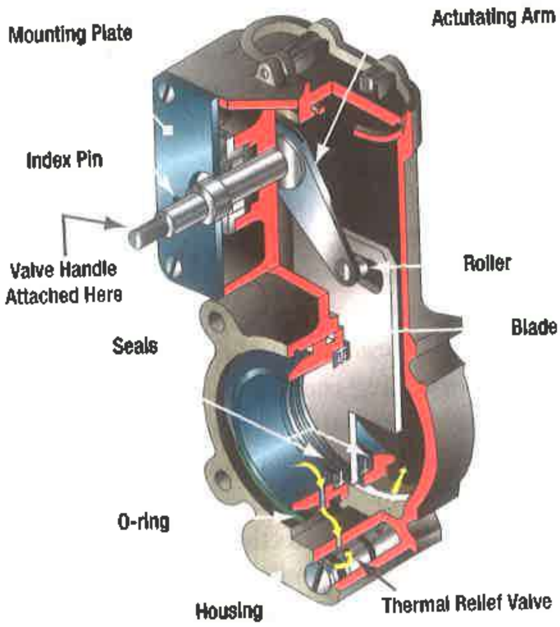


Figure 11-15. A hand operated gate valve.

Motor Operated Valves

The use of electric motors to operate fuel valves are common on large aircraft due to the remote location from the cockpit. The types of valves used are basically the same as manually operated valves, except electric motors are used to actuate them. The two most common motor operated valves are the gate type and the plug type. Motor operated gate valves use a geared reversible motor to turn the actuating arm that moves the fuel gate into or out of the path of the fuel. As with the manually operated gate valve, the gate (or blade) is sealed. An override lever allows the technician to observe the position of the valve and manually position it. (***Figure 11-16***)

Less common are motorized plug type valves. With plug valves, a motor is used to rotate the plug or drum rather than it being rotated manually.

Solenoid Operated Valves

An additional way to operate a remotely located fuel valve is using electric solenoids. A poppet type valve is opened by the magnetic pull developed when an opening solenoid is energized. A spring forces a locking stem into a notch in the stem of a poppet to lock the valve in the open position. Fuel then flows through the opening vacated by the poppet.

To close the poppet and shut off flow, a closing solenoid is energized. Its magnetic pull overcomes the force of the locking spring and pulls the stem out of the notch in the poppet. A spring behind the poppet forces it back onto its seat. A characteristic of solenoid operated valves is that they open and close very quickly. (***Figure 11-17***)

Fire Shut Off Valves

The function of the fire shut off valve (or low pressure shut off) is to be able to cut off the fuel supply line to an engine in case of an external fire or any other situation which makes it necessary to isolate the engine. (***Figure 11-18***)

The valve can be closed either mechanically or electrically depending on the helicopter. There is at least one fire shut off valve in each supply line. The fire shut off valve of the APU is generally operated electrically. This is because the APU is electrically switched off by means of its own monitoring system. If valves are electrically driven, two actuators are often installed for redundancy, with each supplied from different power sources.

FUEL PUMPS

EASA Requirement CS 29.955
Fuel Flow

a. The fuel system for each engine must provide the engine with at least 100% of the fuel required under all operating conditions approved for that rotorcraft. Unless equivalent methods are used, compliance must be shown by testing during which the following provisions are met.
  - The fuel pressure, corrected for accelerations, must be within the limits specified by the engine's type certificate data sheet.
  - The fuel level in the tank may not exceed that

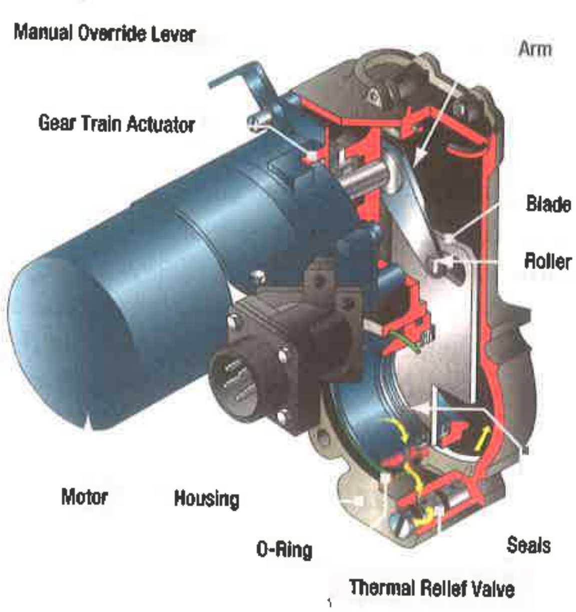


Figure 11-16. An electric motor driven gate valve.

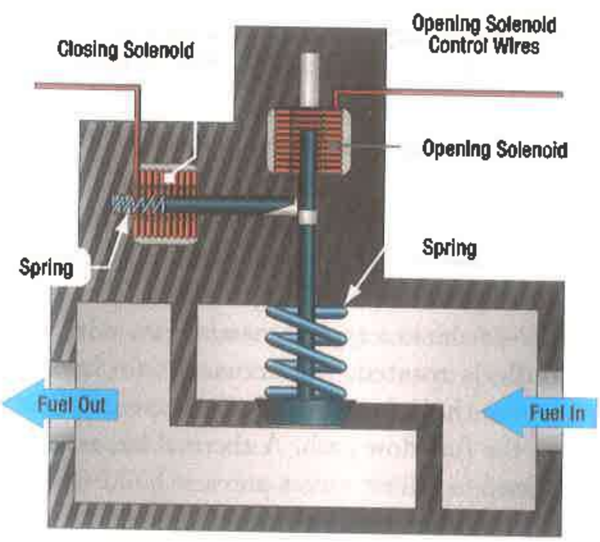


Figure 11-17. A solenoid operated fuel valve.

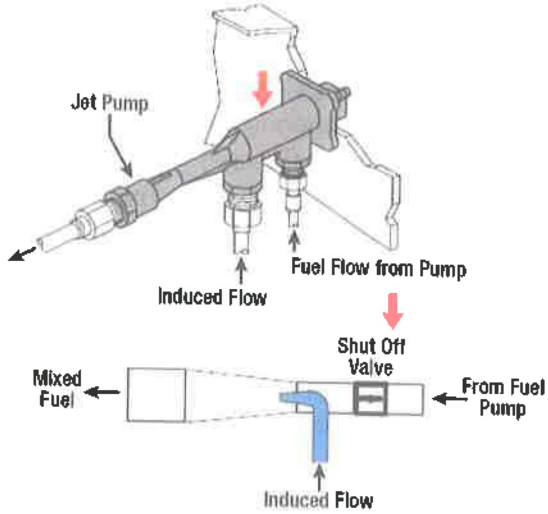


Figure 11-18. Shut-off principle.

established as the unusable fuel supply for that tank.
- The fuel head between the tank and the engine must be critical with respect to rotorcraft flight attitudes.
- The fuel flow transmitter, if installed, and the critical fuel pump (for pump fed systems) must be installed to produce the critical restriction to fuel flow expected from component failure.
- The fuel filter required is blocked to the degree necessary to simulate the accumulation of fuel contamination required to activate the required indicator.

b. If normal operation of the fuel system requires fuel to be transferred to another tank, the transfer must occur automatically via a system which has been shown to maintain the fuel level in the receiving tank within acceptable limits during flight or surface operation.

c. If an engine can be supplied with fuel from more than one tank, the fuel system must be designed to prevent interruption of fuel flow to the engine, without attention by the flight crew, when any tank supplying fuel to that engine is depleted of usable fuel during normal operation and any other tank that normally supplies fuel to that engine alone contains usable fuel.

EASA Requirement CS 29.991
Fuel Pumps

a. Compliance with fuel flow specified in CS 29.955 above must not be jeopardized by failure of any one pump or component required for pump operation except the engine served by that pump.

b. The following fuel pump installation requirements apply:
   - When necessary to maintain the proper fuel pressure a connection must be provided to transmit the carburetor air intake static pressure to the proper fuel pump relief valve connection, and the gauge balance lines must be independently connected to the carburetor inlet pressure to avoid incorrect fuel pressure readings.
   - The installation of fuel pumps having seals or diaphragms that may leak must have means for draining leaking fuel.
   - Each drain line must discharge where it will not create a fire hazard.

Generalities On Fuel Pumps

Fuel pumps are part of most aircraft fuel systems. Standards exist for main and emergency pumps. Operation of any fuel pump may not affect engine operation by creating a hazard, regardless of the engine's power setting or the working status of any other fuel pump.

Turbine engines require dedicated fuel pumps for each engine. Any pump required for operation is considered a main fuel pump. The power supply for the main pump for each engine must be independent of the power supplies for the main pumps of any other engine. There must also be a bypass feature for each positive displacement pump. Emergency pumps must be immediately available to supply fuel to the engine if any main pump fails. The power supply for each emergency pump must also be independent of the power supply for each corresponding main pump. If both the main fuel pump and the emergency pump operate continuously, there must be a means to indicate a malfunction of either pump to the flight crew.

All rotorcraft, due to the lower position of the tanks and the upper position of the engine, have at least one fuel pump to deliver clean fuel under pressure to the fuel metering device of each engine. Engine driven pumps are the primary delivery device. Electrically driven auxiliary pumps are used on many aircraft as well. Sometimes known as boost pumps, auxiliary pumps provide fuel under positive pressure to the engine driven pump and for starting when the engine driven pump is not yet providing sufficient fuel delivery. They are also used to back up the engine driven pump during takeoff and at high altitude to guard against vapor lock. On many large aircraft, boost pumps are also used to move fuel from one tank to another.

Reciprocating engine helicopters often also contain a small hand operated pump called a primer. A primer allows fuel to be pumped directly into the intake port of the cylinders prior to engine start. The primer is useful in cold weather when fuel in the carburetor is difficult to vaporize.

Centrifugal Boost Pumps

The most common type of auxiliary fuel pump, especially in large and high performance aircraft, is the centrifugal pump. Centrifugal pumps are electric motor

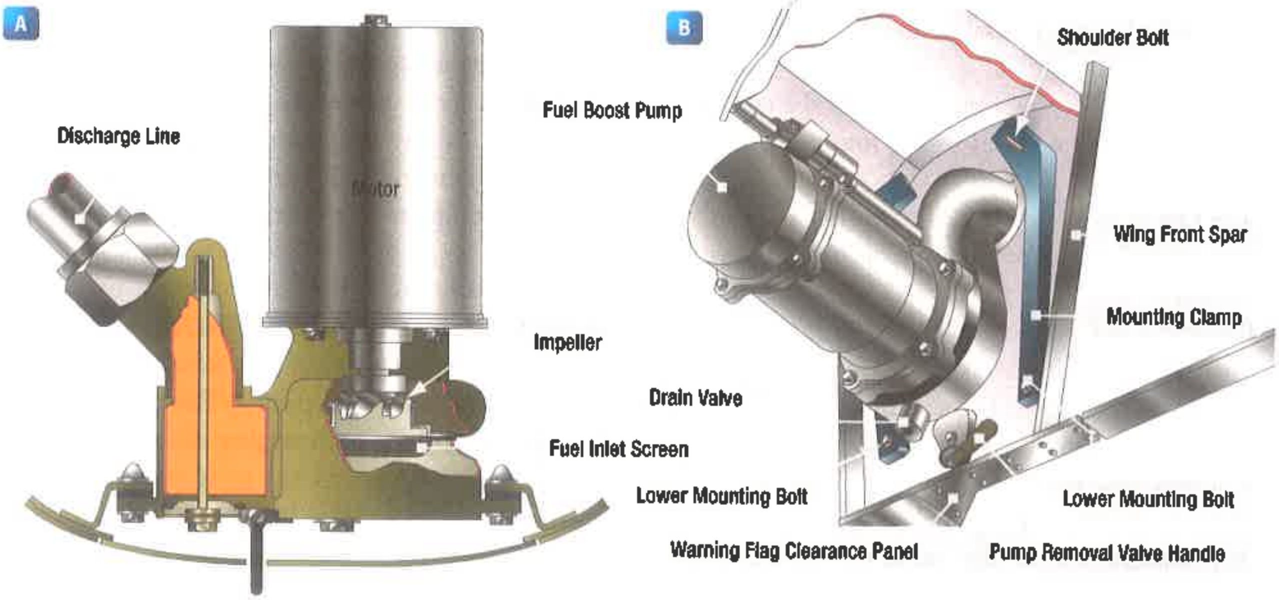


Figure 11-19. A centrifugal fuel boost pump.

driven and are most often submerged in the fuel tank or located just outside of the bottom of the tank with the inlet of the pump extending into the tank. If the pump is mounted outside the tank, a pump removal valve is installed so the pump can be removed without draining the fuel tank. (*Figure 11-19*)

A centrifugal boost pump is a variable displacement pump. It takes in fuel at the center of an impeller and expels it to the outside as the impeller turns. (*Figure 11-20*) An outlet check valve prevents fuel from flowing back through the pump. A fuel feed line is connected to the pump outlet. A bypass valve may

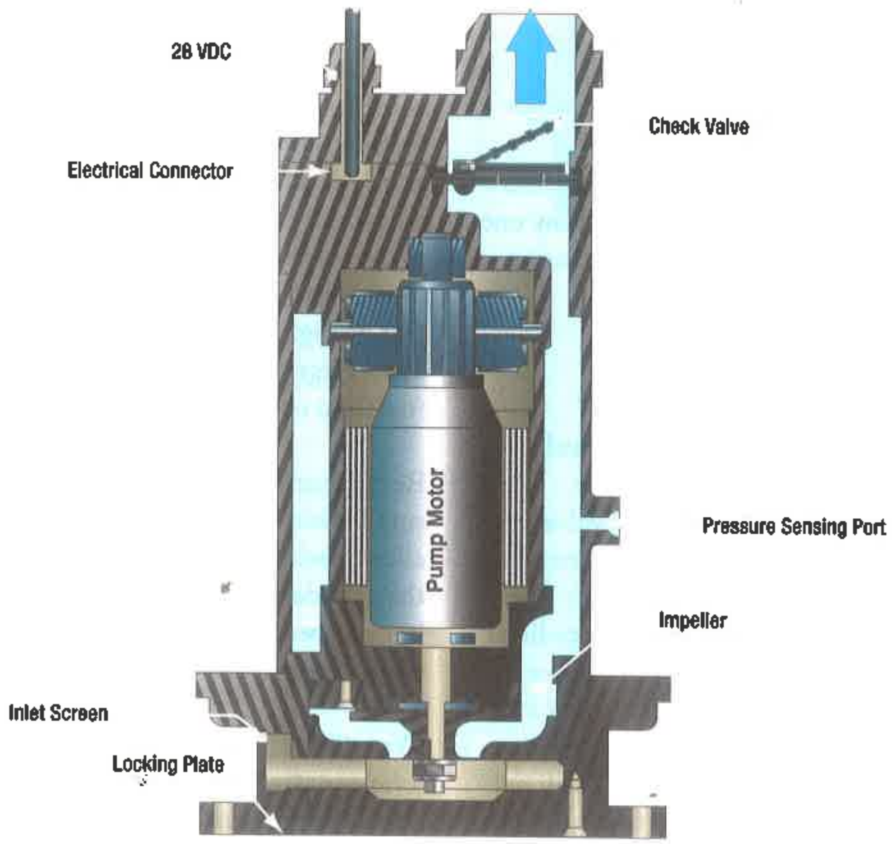


Figure 11-20. The internal workings of a centrifugal fuel boost pump.

be installed in the fuel feed system to allow the engine driven pump to pull fuel from the tank if the boost pump is not operating.

Some centrifugal pumps operate at more than one speed, as selected by the pilot, depending on the phase of aircraft operation. Centrifugal fuel pumps located in fuel tanks ensure positive pressure throughout the fuel system regardless of temperature, altitude, or flight attitude thus preventing vapor lock.

Submerged pumps have fuel proof covers for the electric motor since the motor is in the fuel. Centrifugal pumps mounted on the outside of the tank do not require this. In this case a shutoff valve is provided so the pump can be changed without draining the tank. The inlet of both types of centrifugal pumps are covered with a screen to prevent the ingestion of foreign matter. (***Figure 11-21***)

Ejector Pumps

Fuel tanks with in-tank pumps, such as centrifugal pumps, are constructed to always maintain a fuel supply at the pump inlet. This ensures that the pump does not cavitate and that the pump is always cooled by the fuel. The section of the fuel tank dedicated for the pump installation may be partitioned off with baffles that contain flapper type check valves. These allow fuel to flow inboard to the pump during maneuvers but does not allow it to flow outboard. Some aircraft use ejector pumps to help ensure that liquid fuel is always at the inlet of the pump. A relatively small diameter line circulates pump outflow back into the section of the tank where the pump is located. The fuel is directed through a venturi that is part of the ejector. As the fuel rushes through the venturi, low pressure is formed. An inlet, or line that originates outside of the tank pump area allows fuel to be drawn into the ejector assembly where it is pumped into the fuel pump tank section. Together, with baffle check valves, ejector pumps keep a positive head of fuel at the inlet of the pump. (***Figure 11-22***)

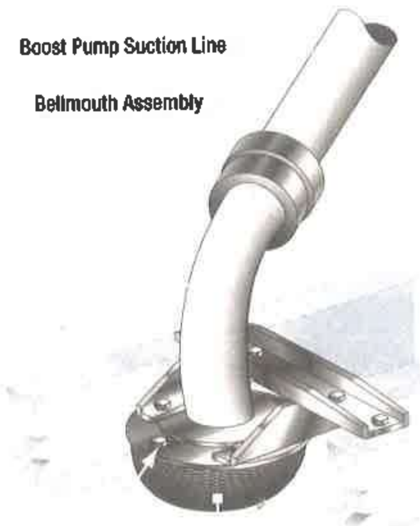


Figure 11-21. A typical fuel boost pump inlet screen installation.

FUEL FILTERS

EASA Requirement CS 29.997
Fuel Strainer or Filter
There must be a fuel strainer or filter between the fuel tank outlet and the inlet of the first fuel system component which is susceptible to fuel contamination, including but not limited to the fuel metering device or an engine positive displacement pump, whichever is nearer the fuel tank outlet. This fuel strainer or filter must:

a. Be accessible for draining and cleaning and must incorporate a screen or element which is easily removable.
b. Have a sediment trap and drain, except if the strainer or filter is easily removable for drain purposes.
c. Be mounted so that its weight is not supported by the connecting lines or by the inlet or outlet connections of the filter itself.
d. Provide a means to remove from the fuel any contaminant which would jeopardize the flow of fuel through fuel system components required for proper engine operation.

Two main types of fuel cleaning device are utilized on aircraft. Fuel strainers are usually constructed of relatively coarse wire mesh. They are designed to trap large pieces of debris and prevent their passage through the fuel system. Fuel strainers do not inhibit the flow of water.

Fuel filters are usually fine mesh. In various applications, they can trap fine sediment that can be only thousands of an inch in diameter and help trap water as well. The technician should be aware that the terms 'strainer' and 'filter' are sometimes used interchangeably even though they are not the same thing.

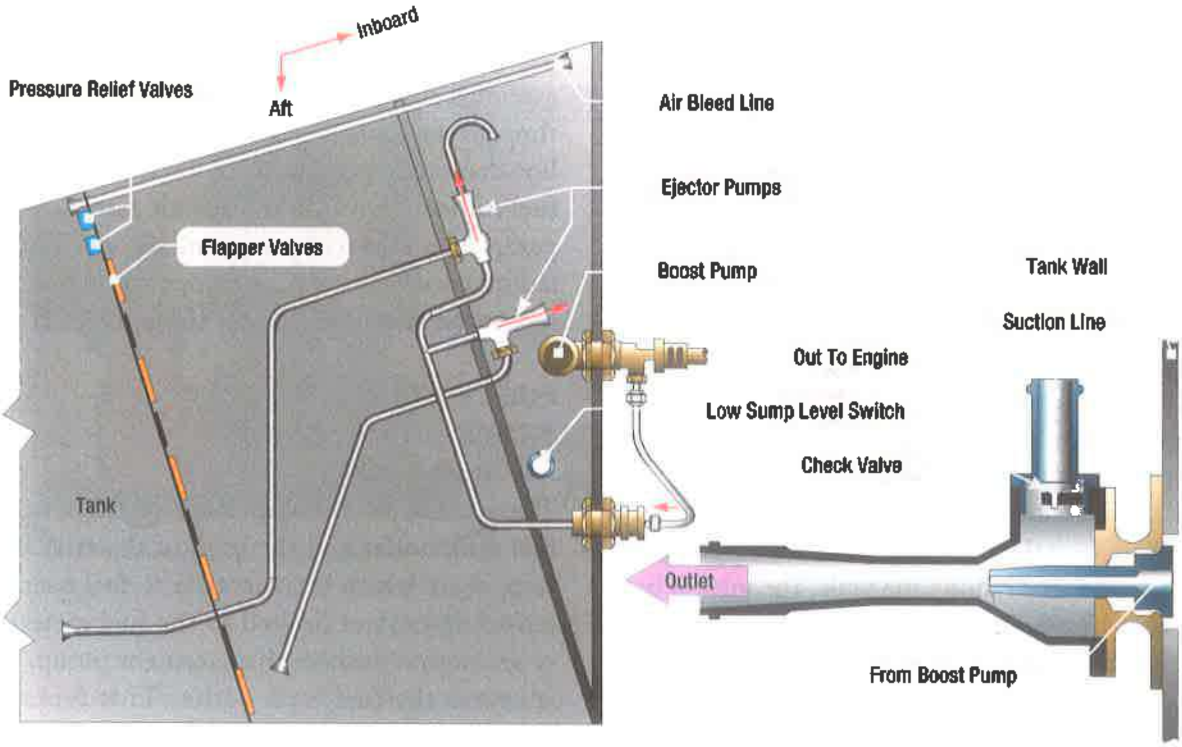


Figure 11-22. An ejector pump uses a venturi.

Micronic filters are commonly used on turbine powered aircraft. This type of filter captures extremely fine particles in the range of 10-25 microns. A micron is 1/1000 of a millimeter. (***Figure 11-23***) As turbine fuel control units are extremely close tolerance devices, it is imperative that fuel is contaminant free. The changeable cellulose filter mesh type shown in ***Figure 11-24*** can block particles 10-200 microns in size and absorbs water when present.

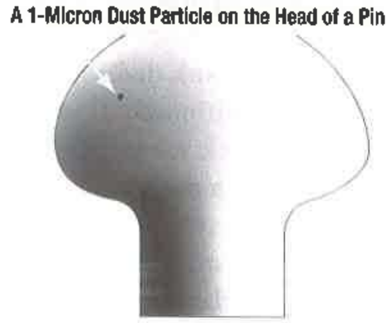


Figure 11-23. Size comparison.

The small size of the micronic filter mesh raises the possibility of the filter being blocked by debris or water. Therefore, a bypass valve is included in the filter assembly that bypasses fuel through the unit should pressure build up from blockage. In addition to a fuel filter installed between the tank and engine driven pump, fuel filters are often used between the engine driven pump and the fuel metering device on turbine engines. A common type is shown in ***Figure 11-25***.

The differential pressure type indicator compares the input pressure of the fuel filter to the output pressure. A circuit is completed when a preset difference occurs. Thus, an indicator is illuminated in the cockpit should a blockage cause the bypass to open or the inlet and outlet pressures to vary significantly.

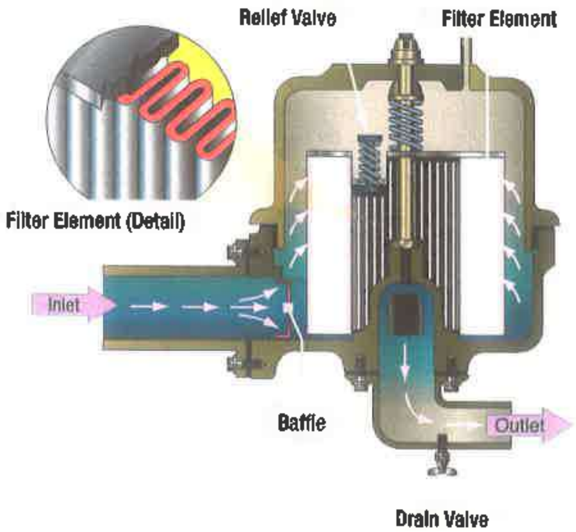


Figure 11-24. A typical micronic fuel filter.

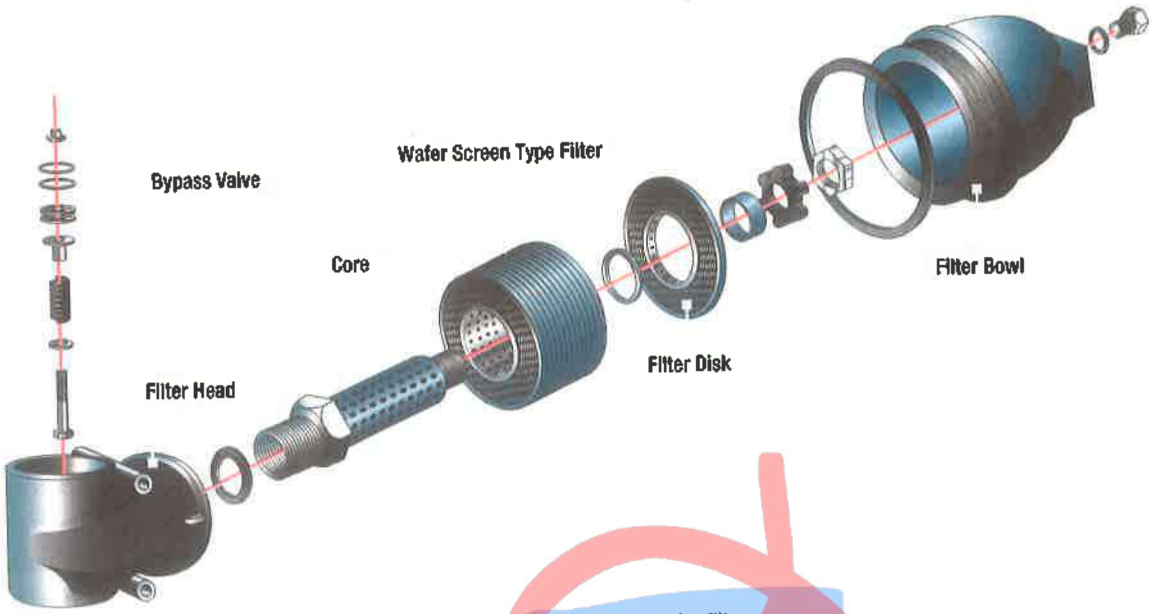


Figure 11-25. A micronic wafer filter.

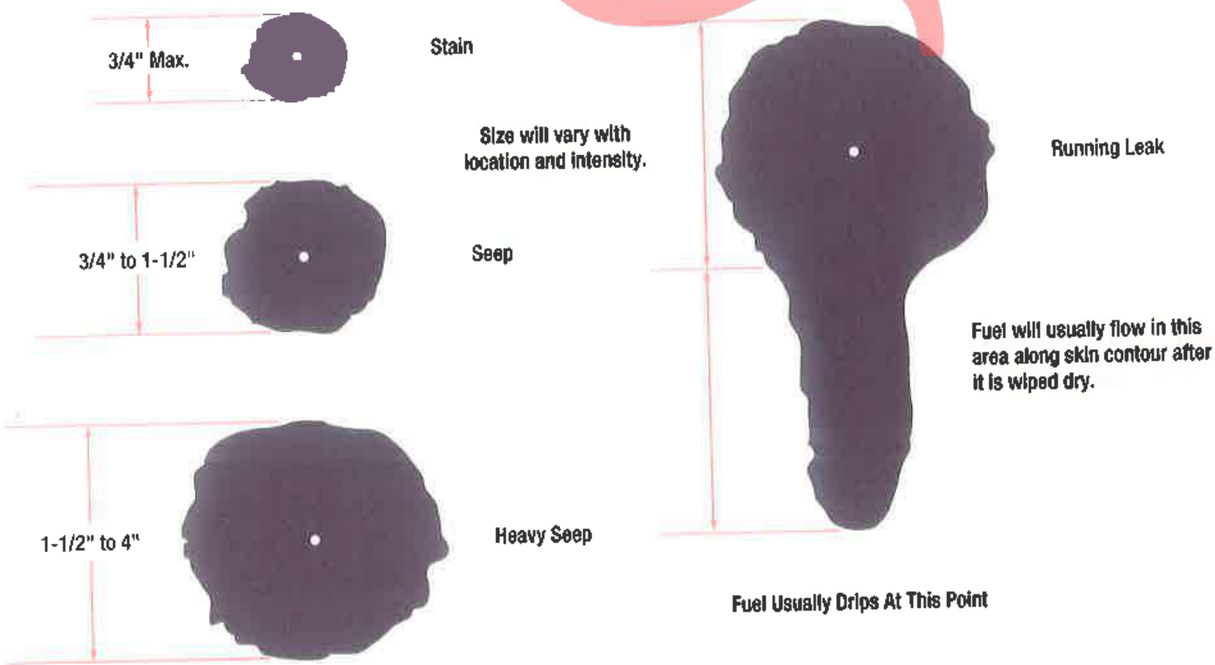


Figure 11-26. The surface area of collected fuel from a leak.

GASKETS, SEALS AND PACKINGS

Fuel leaks can be classified as stains, seeps, heavy seeps, and running leaks. (*Figure 11-26*) In 30 minutes, the surface area of the collected fuel from a leak is a certain size. This is used as the classification standard. When the area is less than 2 cm in diameter, the leak is said to be a stain. From 2-3 cm in diameter, the leak is classified as a seep. Heavy seeps form an area from 3-12 cm in diameter. Running leaks pool and actually drip from the aircraft. They may follow the contour of the aircraft for a long distance.

A leak can often be repaired by replacing a gasket or seal. When this occurs or a component is replaced or reassembled after a maintenance operation, a new gasket, seal, or packing must be installed. Do not use the old one(s). Always be sure to use the correct replacement as identified by part number. Also, most gaskets, seals, and packing have a limited shelf life. They should be used only if they are within the service life stamped on the package.

Remove the entire old gasket and clean all mating surfaces. Clean surfaces and grooves, allow a tight seal. Inspect new gaskets and seals for any flaws. Follow the manufacturer's instructions for replacement, including cleaning procedures and any sealing compound that you may need to apply during replacement. Torque assembly bolts evenly to provide even pressure and prevent pinching.

TROUBLESHOOT THE FUEL SYSTEM

Knowledge of the fuel system and how it operates is essential when troubleshooting. Manufacturers produce diagrams and descriptions in their maintenance manuals to aid the technician including troubleshooting charts and flow diagrams that can be followed. As with all troubleshooting, a logical sequence of steps to narrow the problem to a specific component or location should be followed. Defects within the system can often be located by tracing the fuel flow from the tank through the system to the engine. Each component must be functioning as designed and the cause of the defect symptom must be ruled out sequentially.

DUMPING, VENTING AND DRAINING

EASA Requirement CS 29.1001
Fuel Jettisoning

If a fuel jettisoning system is installed the following apply:

a. The fuel jettisoning system and its operation must be free from fire hazard from fuel or vapors which impinge on any part of the rotorcraft during fuel jettisoning. Controllability of the rotorcraft must remain satisfactory throughout the jettison operation.
b. Means must be provided to prevent jettisoning fuel below the amount required for an all engine climb at maximum power from sea level to 5 000 ft. altitude and cruise thereafter for 30 minutes.
c. The controls for any fuel jettisoning system must allow flight personnel to safely interrupt fuel jettisoning during any part of the jettisoning operation.
d. The fuel jettison system must be designed to comply with the powerplant installation requirements. For each powerplant and auxiliary power unit installation, it must be established that no failure or malfunction will jeopardize the safe operation of the rotorcraft.

DUMPING

Fuel dumping is familiar for fixed wing aircraft operations and is used to jettison the volatile liquid to lighten the aircraft for landing, or in various emergency situations. For a helicopter, this is far less common. The main reason for this is that if a helicopter loaded with fuel can lift itself into the air, it can very likely touch down again with a similar load, even if very little fuel has been burnt. The exception could conceivably be a helicopter rolling takeoff, where forward speed provides additional lift and can allow liftoff at higher weights. However, in that scenario, a heavily loaded helicopter could also get back to the ground safely using a rolling landing.

In the case of some military helicopters, such as the MH-47, the situation is a little different due to their high altitude capabilities and ability to refuel in flight. Operating at high altitude where the rotors generate less lift in the thinner air is a perennial challenge and there are sometimes reasons for the crew to want to lighten the fuel load. Such helicopters are equipped with fuel dumping pipes as seen on *Figure 11-27.*

FUEL VENT SYSTEMS

EASA Requirement CS 29.975
Fuel Tank and Carburetor Vapor Vents

a. Fuel tank vents. Each fuel tank must be vented from the top part of the expansion space so that venting is effective under normal flight conditions.
b. Carburetor vapor vents. Each carburetor with vapor elimination connections must have a vent line to lead vapors back to one of the fuel tanks.

The purpose of the ventilation system is to prevent structural damage to the fuel tanks by under pressure or over pressure. Regulations say that each fuel tank must have an expansion space of not less than 2% of

Figure 11-27. MH47-Fuel-dump pipes.

the combined tank capacity. This means that inflow and outflow of air must always be possible when using the fuel and when refueling and defueling.

A series of vent tubing and channels connects all tanks to vent space into surge tanks (if present) or vent overboard. Venting must be configured to ensure the fuel may vent regardless of the attitude of the aircraft or the quantity of fuel on board.

The ventilation system consists of lines, ports, valves, vent surge tanks and ram air ports. The ram air ports ensure that there is positive pressure on the fuel in each tank. Between the ram air ports and the surge tank, there is sometimes a flame arrestor (a fine meshed screen) in the line that keeps flames from entering the tank. (***Figure 11-28***) Check valves ensure that the vent lines will be emptied back to the tank if fuel should be present in them. A ventilation system is designed in such a way that, when putting too much fuel in the tank (overflow), the fuel flows to the surge tanks via this system. If fuel exists in the surge tanks, they will be emptied when the engine(s) is/are running.

The air release valve releases air trapped in the engine fuel feed line, while preventing fuel from flowing out. The valve also prevents air flow into the fuel systems. It is installed at the highest point of the fuel line. (***Figure 11-29***)

FUEL SYSTEM DRAINS

EASA Requirement CS 29.999
Fuel System Drains

a. There must be at least one accessible drain at the lowest point in each fuel system to completely drain the system with the rotorcraft in any expected ground attitude.
b. Each drain, including the drains in the fuel tank sump must discharge clear of all parts of the rotorcraft, have means to ensure positive closure in the off position, and have a valve that is readily accessible to be easily opened and closed, and located or protected to prevent fuel spillage in the event of a landing with landing gear retracted.

Draining of water, sediment and fuel residue from the tank is done via a drain valve as shown in ***Figure 11-30***. This valve is typically found on the underside of the fuselage. Since water has a higher density than fuel, it will accumulate at the lowest point of the tank. Therefore, the drain valves are usually located at the lowest point. Where drain valves are not placed in that

Figure 11-28. Flame arrestors.

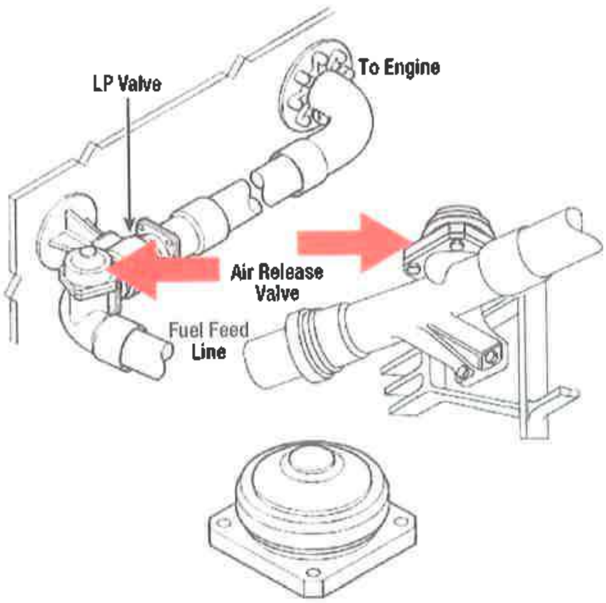


Figure 11-29. Air release valve.

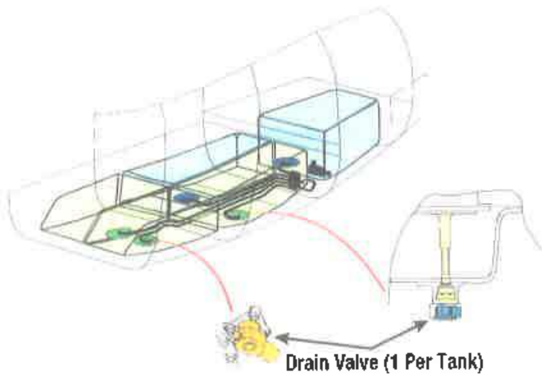


Figure 11-30. Fuel drain valve location.

location, indirect valves with a vacuum apparatus must be used. (***Figure 11-31***) Prior to draining existing water from a fuel tank it is important to let the water settle for a certain time. Drain valves are usually equipped with a check valve. This means if the valve assembly is leaking, it can be removed from the valve body for repair without emptying the tank.

CROSS FEED AND TRANSFER

EASA Requirement CS 29.957
Flow Between Interconnected Tanks

a. Where tank outlets are interconnected to allow fuel to flow between them due to gravity or flight accelerations, it must be impossible for fuel to flow between tanks in quantities great enough to cause overflow from the tank vent.
b. If fuel can be pumped from one tank to another in flight, the design of the vents and transfer system must prevent structural damage to tanks from overfilling, and there must be means to warn the crew before overflow through the vents occurs.

LATERAL BALANCE SYSTEMS

When consuming fuel in flight, the aircraft center of gravity can change and disturb flight stability. The transfer of fuel to maintain the desired center of gravity of the aircraft is essential. Transfer pumps are used to send fuel between tanks for this purpose. During flight, the crew must check the quantity of fuel in each tank group and must balance it to keep the center of gravity at the good position.

CROSS FEED AND TRANSFER

In the case of a helicopter equipped with several tanks, the fuel may pass through one or more tanks before reaching the engine. Cross feeding is therefore considered as that transfer between either several tanks of the same group by gravity or by ejectors; or between two different groups of tanks when carried out by a pump and a cross feed circuit. (***Figure 11-32***) In this case, the cross feed is only used in the event of an engine failure.

The fuel transfer system is a series of fuel lines and valves that allow the movement of fuel from one tank to another on board the aircraft. Two booster pumps, one in each longitudinal tank, send fuel to the corresponding engine. Gravity empties the transverse and rear tank into the longitudinal tank which always remains full. Due to an additional tank the quantity is not equal in each group and can unbalance the helicopter. Dedicated transfer boost pumps move fuel from one group to another to balance that lateral weight.
(***Figure 11-33 and Figure 11-34***)

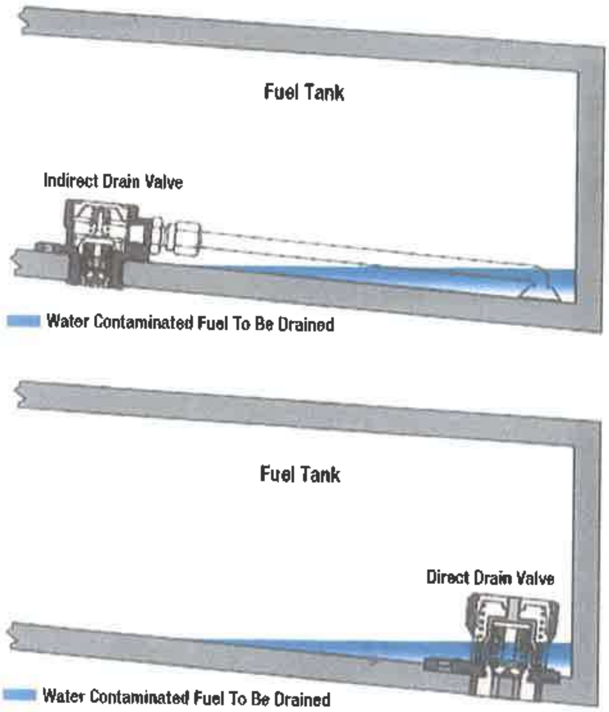


Figure 11-31. Direct and indirect drain valves.

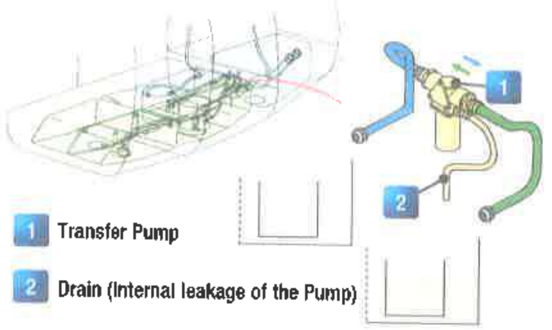


Figure 11-32. Transfer pump.

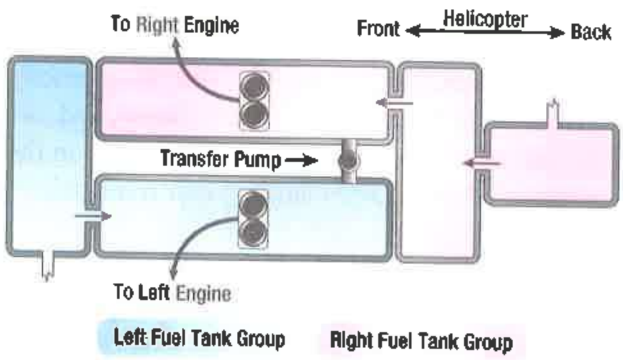


Figure 11-33. Cross-feed and transfer systems.

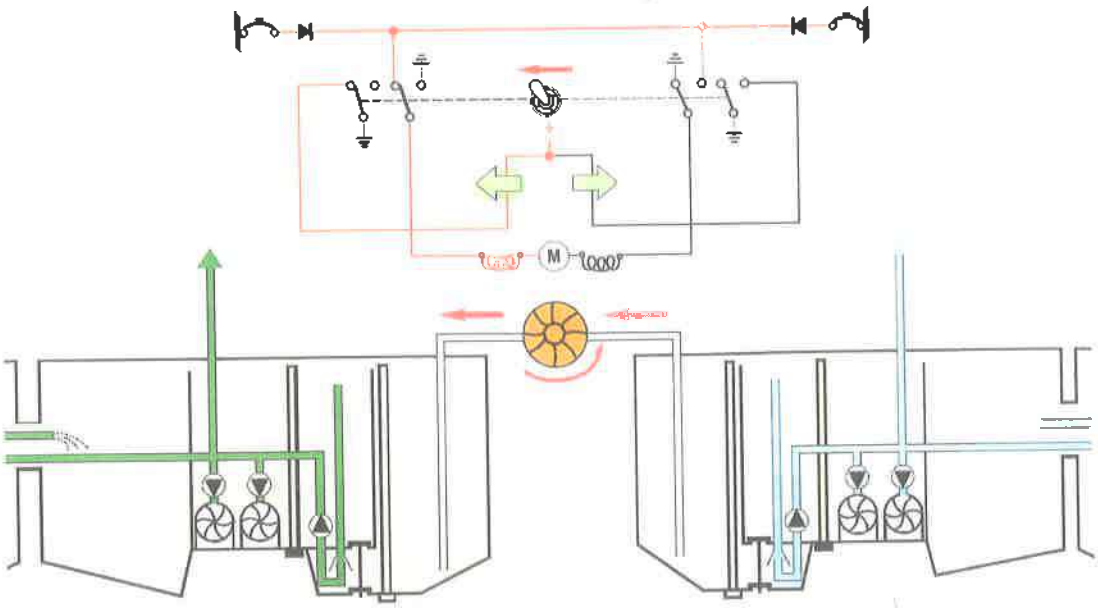


Figure 11-34. Cross-feed and transfer schematic.

In the event of an engine failure, all the fuel can be switched to the right or to the left, allowing the greatest range. The main engine pump is capable of drawing fuel from the tank to the engine without a booster pump. For safety reasons, if an electrical failure occurs stopping the electric booster pump, the engine must continue to run.

In ***Figure 11-34*** the green arrow indicates the direction of transfer from one group to another. If the transfer pump control selector is "to the left", the fuel is sucked from the low point of the tank and transferred from the right group to the left group.

INDICATIONS AND WARNINGS

Fuel indicating systems monitor a variety of parameters. True fuel flow indicators for each engine are used as the primary means of monitoring fuel delivery. The fuel filter bypass warning lights are necessary on the fuel system to inform the pilot of the situation. The indicator is located on the instrument panel or is displayed on a Multi Function Display (MFD). Valve position indicators and various warning lights and annunciations are also used. Low fuel pressure warning lights are also common. The sensors for these are in the boost pump outlet line, giving an indication of possible boost pump failure.

FUEL QUANTITY INDICATING SYSTEMS

Fuel quantity indications for all tanks are important on all types of aircraft. Capacitance type fuel quantity indication systems and a fuel totalizer are often used as discussed below.

Ratiometer Type

Most electric fuel quantity indicators operate with Direct Current (DC) and use variable resistance in a circuit to drive a ratiometer type indicator. The movement of a float in the tank moves a connecting arm to the wiper on a variable resistor in the tank unit. This resistor is wired in series with one of the coils of a ratiometer type fuel gauge in the instrument panel. Changes to the current flowing through the tank unit resistor changes the current flowing through one of the coils in the indicator. This alters the magnetic field in which the indicating pointer pivots. The calibrated dial indicates the corresponding fuel quantity. (***Figure 11-35***)

Digital indicators are available that work with the same signal from the tank unit. They convert the variable resistance into a digital display in the cockpit instrument. (***Figure 11-36***) Fully digital instrumentation systems convert the variable resistance into a signal to be processed in a computer and displayed on a flat screen panel.

Capacitance Type

Large and high performance aircraft typically utilize electronic fuel quantity systems. These systems have the advantage of having no moving parts in the tank sending units. Variable capacitance transmitters are installed in the fuel tanks extending from the top to the bottom of each tank in the usable fuel. Several of these fuel probes, as they are sometimes called, may be installed in a large tank. (***Figure 11-37***)

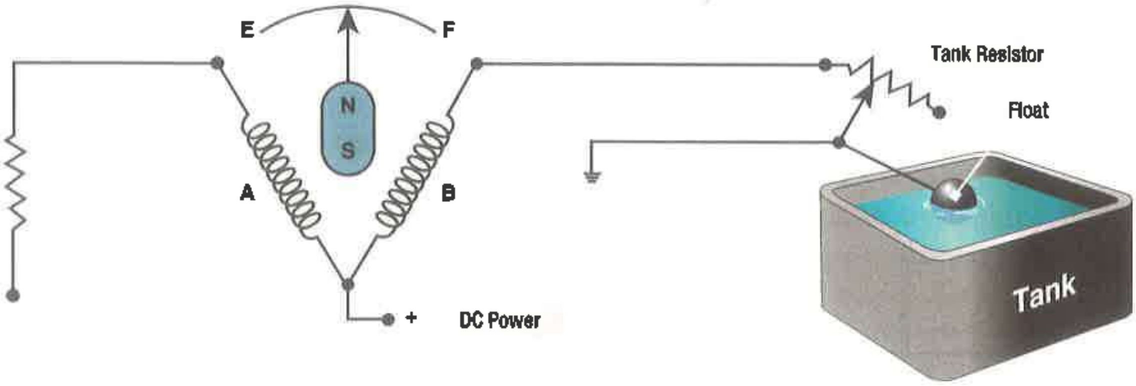


Figure 11-35. A DC electric fuel quantity indicator.

Figure 11-36. Digital fuel quantity gauges.

Figure 11-37. A fuel tank transmitter.

Each probe is wired in parallel. As the level of the fuel changes, the capacitance of each unit changes. The capacitance transmitted by all the probes in a tank are totaled and compared in a bridge circuit by a microchip computer in the cockpit indicator. As the aircraft maneuvers, some probes are in more fuel than others due to the attitude of the aircraft. The indication remains steady because the total capacitance transmitted by all the probes remains the same. A trimmer is used to match the capacitance output with the precalibrated indicator.

A capacitor is a device that stores electricity. The amount it can store depends on three factors: the area of its plates, the distance between the plates, and the dielectric constant of the material separating the plates.

A fuel tank probe contains two concentric plates that are a fixed distance apart. Therefore, the capacitance of a unit can change if the dielectric constant of the material separating the plates varies. The units are open at the top and bottom so they can assume the same level of fuel as is in the tanks. Therefore, the material between the plates is either fuel (if the tank is full), air (if the tank is empty), or some ratio of fuel and air depending on how much remains in the tank. ***Figure 11-38*** shows a simplified illustration of this construction.

The bridge circuit that measures the capacitance of the tank probes uses a reference capacitor for comparison. When voltage is induced into the bridge, the capacitive reactance of the probes and the reference capacitor can be equal or different. The magnitude of the difference is translated into an indication of the fuel quantity in the tank calibrated in pounds. ***Figure 11-39*** represents the nature of this comparison bridge circuit.

The use of tank capacitors, a reference capacitor, and a microchip bridge in the fuel quantity indicators is complicated by the fact that temperature affects the dielectric constant of the fuel. A compensator unit (mounted low in the tank so it is always covered with fuel) is wired into the circuit. It modifies current flow to reflect temperature variations of the fuel, which affect its density and thus capacitance of the probes. (***Figure 11-40***)

An amplifier is also needed in older systems. The amplitude of the electric signals must be increased to move the servo motor in the analog indicator. Additionally, the dielectric constant of different turbine engine fuels approved for a particular aircraft may also vary. Calibration is required to overcome this. A fuel summation unit is part of the capacitance indication system. It is used to add the tank quantities from all indicators. This total fuel quantity can be used by the crew and by flight management computers for calculating optimum airspeed and engine performance limits for climb, cruise, descent, etc. Capacitance type fuel quantity system test units are available for troubleshooting and ensuring proper functioning and calibration of the indicating components.

Mechanical Type

Many aircraft with capacitance type fuel indicating systems also use a mechanical indication system to cross check fuel quantity indications and to ascertain the amount of fuel on board the aircraft when electrical power is not available. A handful of fuel measuring sticks, (or drip sticks) are mounted throughout each tank. When pushed and rotated, the drip stick can be lowered until fuel begins to exit the hole on the bottom of each stick. This is the point at which the top of the stick is equal to the height of the fuel. The sticks are fitted with a calibrated scale. By adding the indications of all the drip sticks and converting to pounds or gallons via a chart supplied by the manufacturer, the quantity of the fuel in the tank can be ascertained. (***Figure 11-41***)

FUEL FLOW METERS

A fuel flow meter indicates the amount of fuel used by the engine in real time. This can be useful to the pilot for ascertaining engine performance and for flight planning

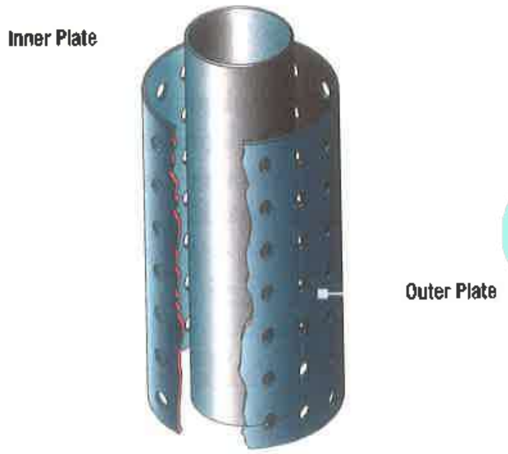


Figure 11-38. The capacitance of tank probes varies in a capacitance.

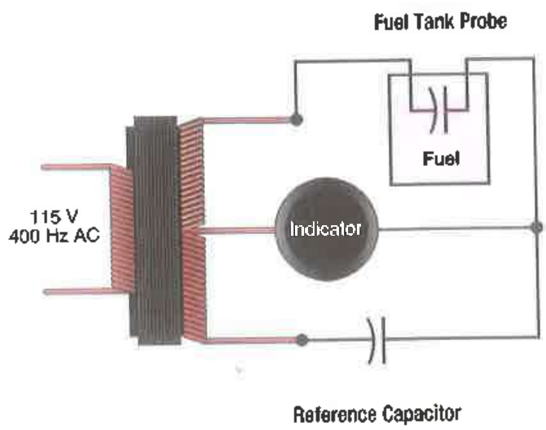


Figure 11-39. A simplified capacitance bridge.

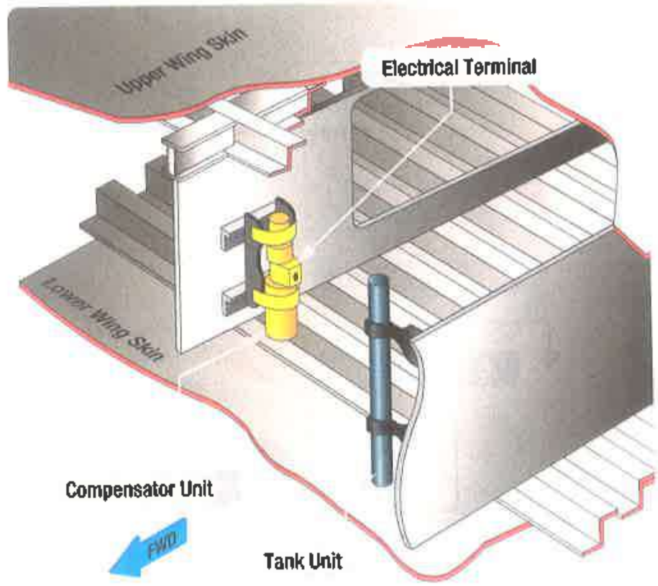


Figure 11-40. A fuel quantity tank unit and compensator unit.

calculations. The type of fuel flow meter used on an aircraft depends primarily on the powerplant being used and its associated fuel system.

Measuring fuel flow accurately is complicated by the fact that the fuel mass changes with temperature or with the type of fuel used in turbine engines. Turbine engine aircraft experience the greatest range of fuel density from temperature variation and fuel composition. A fuel flow device measures fuel mass for an accurate flow indication in the cockpit. This indicator takes advantage of the direct relationship between fuel mass and viscosity. Fuel is swirled by a cylindrical impeller that rotates at a fixed speed. The outflow deflects a turbine just downstream of the impeller. The turbine is held with calibrated springs.

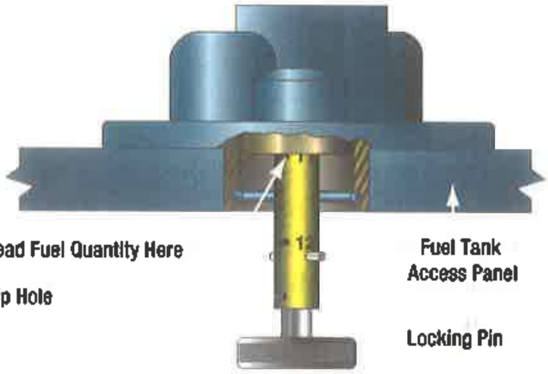


Figure 11-41. A fuel drip stick.

Since the impeller motor swirls the fuel at a fixed rate, any variation of the turbine's deflection is caused by the volume and viscosity of the fuel. The viscosity component represents the fuel's mass. (*Figure 11-42*)

An Alternating Current (AC) synchro system is part of the mass fuel flow meter. It is used to position a pointer against the cockpit indicator scale calibrated in pounds per hour. With accurate fuel flow knowledge, numerous calculations can be performed to aid the pilot's situational awareness and flight planning.

Most high performance aircraft have a fuel totalizer that electronically calculates and displays information, such as total fuel used, total fuel remaining on board the aircraft, total range and flight time remaining at the present airspeed, rate of fuel consumption, etc. Various types of fuel flow sensors/transmitters are available in new aircraft and for retrofit to older aircraft. Increasing use of microprocessors and computers on aircraft enable the integration of fuel temperature and other compensating factors to produce highly accurate information.

Thermal dispersion technology provides flow sensing with no moving parts and digital output signals. These sensors consist of two Resistance Temperature Detectors (RTDs). One is a reference RTD that measures the

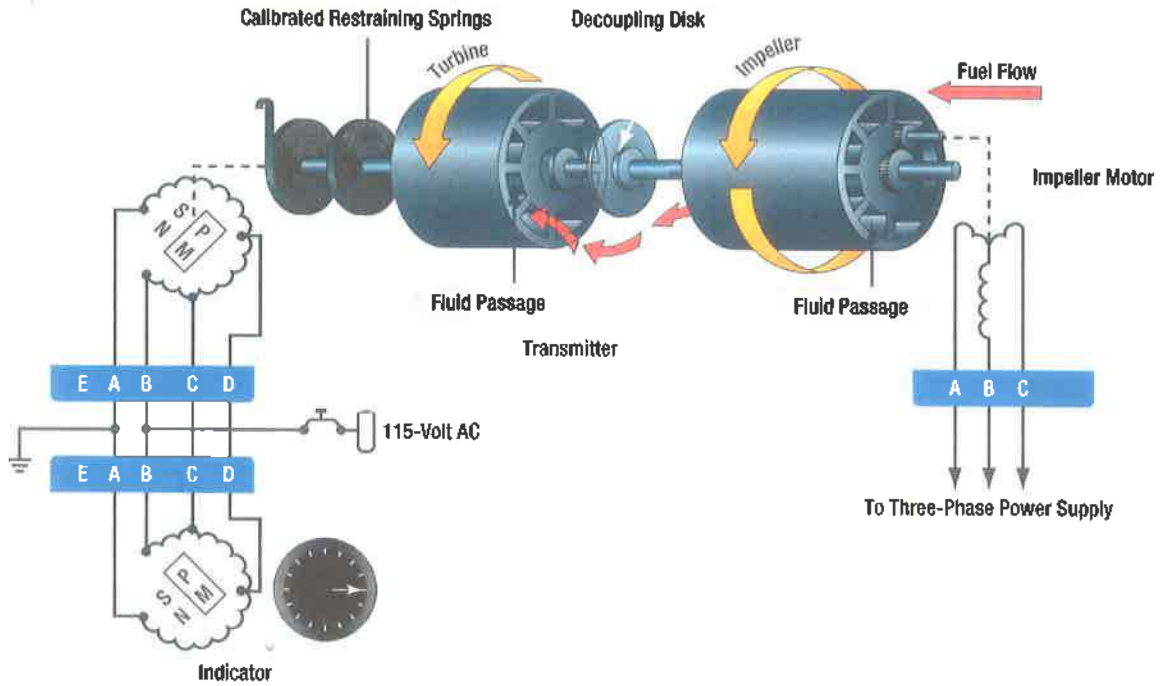


Figure 11-42. A mass flow fuel flow indicating system.

temperature of the fuel. The other is the active RTD which is heated by an adjacent element to a temperature higher than the fuel. As the fuel flows, the active element cools proportionally to the fuel flow. The temperature difference between the two RTDs is highest at no flow. The RTDs are connected to an electronic assembly that supplies power to the heater and uses sensing circuitry and a microprocessor to control a constant temperature difference between the heated and unheated RTDs. The electrical current to the heater is proportional to the mass flow of the fuel. (***Figure 11-43***)

FUEL HEATING

Fuel heaters heat the fuel so that combustion is facilitated. The most common types are air/fuel heaters and oil/fuel heaters. An air/fuel heater uses hot air bleed air from the compressor to heat the fuel. An oil/fuel exchanger heats the fuel with hot engine oil. It not only heats the fuel, but also cools the engine oil. (***Figure 11-44***)

FUEL PRESSURE GAUGES

Monitoring fuel pressure can give the pilot early warning of a fuel system related malfunction. Verification that the fuel system is delivering fuel to the fuel metering device can be critical. Light aircraft utilize a direct reading Bourdon tube pressure gauge. It is connected into the fuel inlet of the fuel metering device with a line extending to the back of the gauge in the cockpit. A more complex aircraft may have a sensor with a transducer located at the fuel inlet to the metering device that sends electrical signals to the cockpit. (***Figure 11-45***)

In aircraft equipped with an auxiliary pump, the fuel pressure gauge indicates the auxiliary pump pressure until the engine is started. When the auxiliary pump is switched off, the gauge indicates the pressure developed by the engine driven pump. Modern aircraft may use a variety of sensors including solid state types and those with digital output signals. These can be processed in the instrument's microprocessor, if so equipped, or in a computer and sent to the display unit. (***Figure 11-46***)

Figure 11-43. Fuel flow sensing units using thermal dispersion technology.

Figure 11-45. A typical fuel gauge.

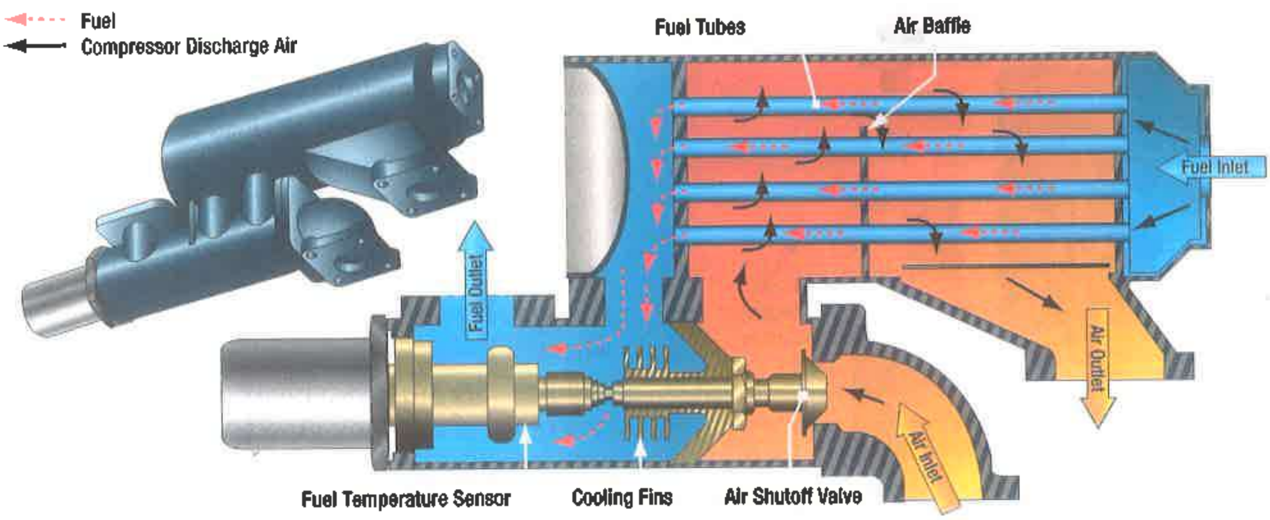


Figure 11-44. An air-fuel heat exchanger.

Figure 11-46. An electronic display of fuel parameters.

PRESSURE WARNING SIGNAL

Visual and audible warning devices are used in conjunction with gauge indications to draw the pilot's attention to certain conditions. Fuel pressure is an important parameter that merits the use of a warning signal when it falls outside the normal operating range.

Low fuel pressure warning lights can be illuminated using simple pressure sensing switches. (*Figure 11-47*) The contacts of the switch will close when fuel pressure against the diaphragm is insufficient to hold them open. This allows current to flow to the annunciator or warning light in the cockpit.

Most turbine powered aircraft utilize a low-pressure warning switch at the outlet of each fuel boost pump. The annunciator for each is positioned adjacent to the boost pump ON/OFF switch on the fuel panel in the cockpit. (*Figure 11-48*)

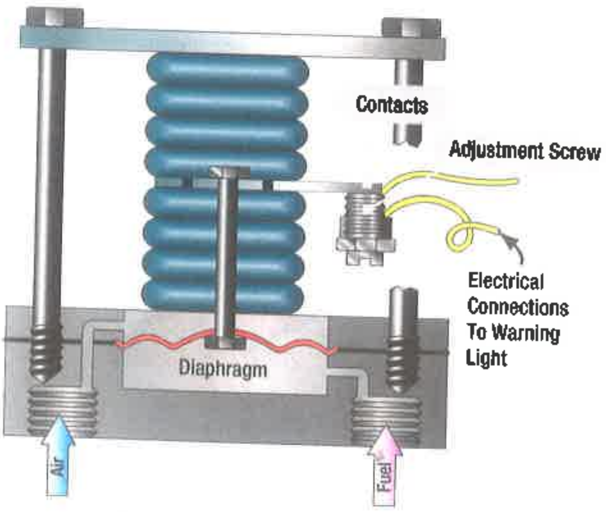


Figure 11-47. Fuel pressure warning signal.

Figure 11-48. A transport category aircraft fuel panel.

FUELING AND DEFUELING

EASA Requirement CS 29.979

Pressure Refueling and Fueling Provisions

a. Each fueling connection below the fuel level in each tank must have means to prevent the escape of hazardous quantities of fuel from that tank in case of malfunction of the entry valve.

b. For systems intended for pressure refueling, a means in addition to the normal means for limiting the tank content must be installed to prevent damage to the tank in case of failure of the normal means.

c. The rotorcraft pressure fueling system (besides fuel tanks and fuel tank vents) must withstand an ultimate load that is 2.0 times the load arising from the maximum pressure, including surge that is likely to occur during fueling. The maximum surge pressure must be established with any combination of tank valves being closed.

d. The rotorcraft defueling system (besides fuel tanks and fuel tank vents) must withstand an ultimate load that is 2.0 times the load arising from the maximum permissible defueling pressure (positive or negative) at the fueling connection.

FUELING

Always refuel an aircraft outdoors, not in a hangar where fuel vapors can build up and increase the risk and severity of an accident. Generally, there are two types of refueling process: gravity refueling and pressure refueling.

Refueling By Gravity

Gravity refueling is done from above by opening the upper fuel tank cap. The refueling nozzle is carefully inserted into the opening and the fuel is sent to the tank with a low pressure pump from the truck. Some connection valves allow the bottom tanks to be filled by gravity. When the top one is full, the bottom one is full too. When complete, the cap is attached, and subsequent tanks are opened and filled until the aircraft has the desired amount of fuel on board. Clean the area adjacent to the fill port when refueling above the wing. Make sure the fuel nozzle is also clean.

As additional precautions, open the cap only when ready to dispense the fuel. Do not insert the neck of the nozzle deeply enough to hit bottom. This could dent the tank, or the aircraft skin if it is an integral tank. Exercise caution to avoid damage to the surface of the airframe by the heavy fuel hose. All aviation fuel nozzles have static bonding wires that must be attached to the aircraft before the fuel cap is opened. (***Figure 11-49***)

Pressure Refueling

When pressure fueling, the aircraft receptacle is part of a fueling valve assembly. When the fueling nozzle is properly connected and locked, a plunger unlocks the aircraft valve so fuel can be pumped through. Normally, all tanks of each group can be fueled from a single point. (***Figure 11-50***) Ensure that the pressure developed by the refueling pump is correct for the aircraft before pumping fuel.

When fueling from a fuel truck, precautions should be taken. If the truck is not in continuous service, all sumps should be drained before moving the truck, and the fuel should be visually inspected to be sure it is bright and clean. Turbine fuel should be allowed to settle for a few hours if the fuel truck tank has recently been filled or the truck has been jostled, such as when driven over a bumpy service road at the airport.

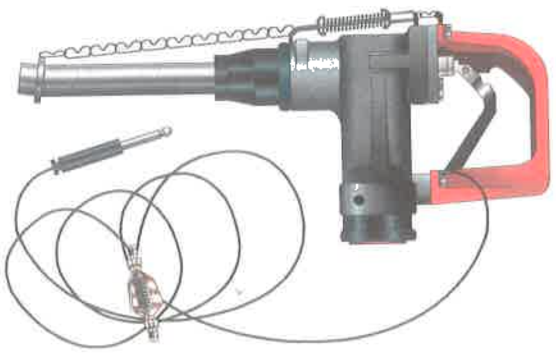


Figure 11-49. An AVGAS fueling nozzle.

Slowly maneuver the fuel truck into position for refueling. The truck should be parked parallel to the wings and in front of the fuselage if possible. Avoid backing toward the aircraft. Set the parking brake and chock the wheels. Connect a static bonding cable from the truck to the aircraft. This cable is typically stored on a reel mounted on the truck. A ladder should be used if the refuel point is not accessible while standing on the ground. Filler nozzles should be treated as the important tools that they are. They should not be dropped or dragged across the apron. Most have dust caps that should be removed only for the actual fueling process and then immediately replaced. Nozzles should be clean to avoid contamination of the fuel. They should not leak and should be repaired at the earliest sign of malfunction. Keep the fueling nozzle in constant contact with the filler neck spout when fueling. Never leave the nozzle in the fill spout unattended.

DEFUELING

Removing the fuel contained in aircraft fuel tanks is sometimes required. This can occur for maintenance, inspection, or due to contamination. Occasionally a change in flight plan may require defueling. Safety procedures for defueling are the same as those for fueling. Always defuel outside. Fire extinguishers should be on hand. Bonding cables should be attached to guard against static electricity buildup. Defueling should be performed by experienced personnel. Inexperienced personnel must be checked out before doing so without assistance. Consult the manufacturer's operations manual if in doubt.

Pressure fueled aircraft normally defuel through the pressure fueling port. The aircraft in-tank boost pumps can be used to pump the fuel out. The pump on a fuel truck can also be used to draw fuel out. Tanks can also be defueled through the tank sump drains, but the large size of the tanks usually makes this impractical.

What to do with the fuel coming out of a tank depends on a few factors. If the tank is being drained due to suspected contamination, it should not be mixed with any other fuel. It should be stored in separate containers, treated if possible or disposed of properly. The

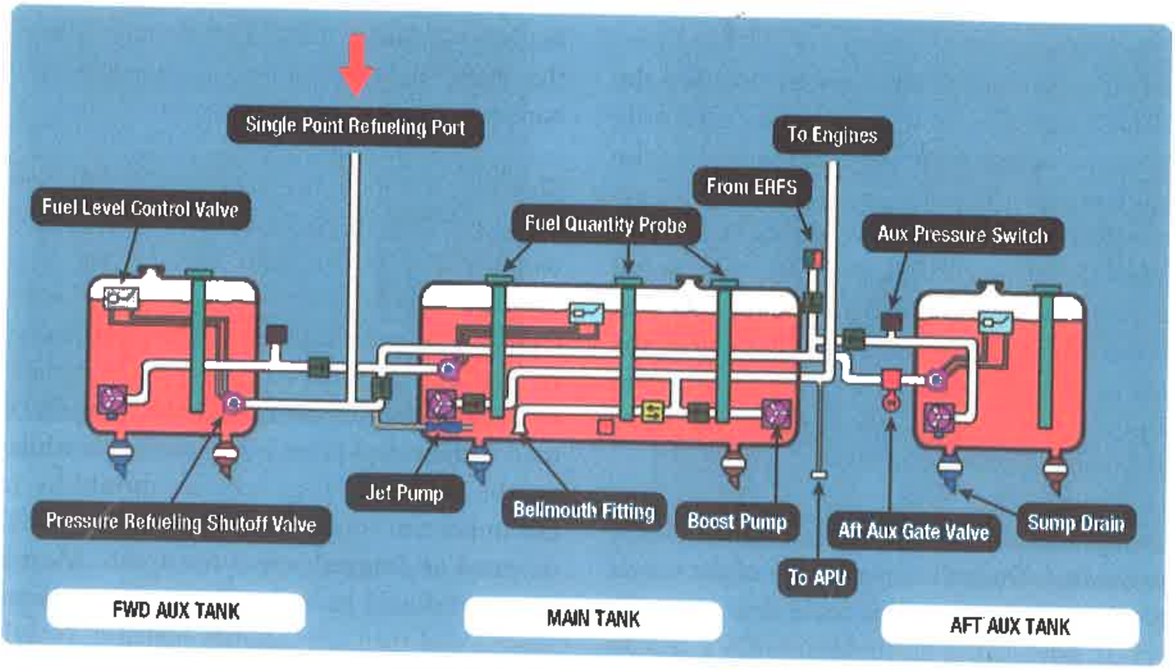


Figure 11-50. Single point refueling port.

manufacturer may have requirements for good fuel that has been defueled from an aircraft, specifying whether it can be reused and the type of storage container in which it must be stored.

Good fuel removed from an aircraft must be handled with all precautions used when handling any fuel. It must only be put into clean tanks and efforts must be made to keep it clean. It may be put back in the aircraft or another aircraft if the manufacturer allows. Large aircraft can often transfer fuel from a tank requiring maintenance to another tank to avoid the defueling process.

FIRE HAZARDS WHEN FUELING OR DEFUELING

Due to the combustible nature of AvGas and turbine fuel, the potential for fire while fueling and defueling aircraft must be addressed. Always fuel and defuel aircraft outside, not in a hangar that serves as an enclosed area for vapors to build up to a combustible level. Clothing worn by refueling personnel should not promote static electrical buildup. Synthetic fabrics such as nylon should be avoided. Cotton has proved to be safe for fuel handling attire.

The most controllable of the ingredients required for fire is the source of ignition. It is necessary to prevent ignition anywhere near the aircraft during fueling or refueling. Any open flame, such as a lit cigarette must be extinguished. Operation of any electrical devices must be avoided. Radio and radar use is prohibited. It is important to note that fuel vapors proliferate well beyond the actual fuel tank opening and a simple spark, even one caused by static electricity could be enough for ignition.

As was discussed, empty fuel tanks have an extreme potential for ignition or explosion. Although the liquid fuel has been removed, ignitable vapor can remain for a long period of time. Purging the vapor out of any empty fuel tank is an absolute necessity before any repair is initiated. Spilled fuel poses an additional fire hazard. A thin layer of fuel vaporizes quickly. Small spills should be wiped up immediately. Larger spills can be flooded with water to dissipate the fuel. Do not sweep fuel that has spilled onto the ramp.

Class B (CO_2) fire extinguishers need to be charged and nearby during the fueling and defueling. Fueling personnel must know exactly where they are and how to use them. Aim the extinguisher nozzle at the base of the flame and spray in a sweeping motion to have the agent fall over the flames to displace the oxygen and smother the fire. Dry chemical fire extinguishers rated for fuel can also be used. These leave behind a residue that requires cleanup that can be extensive and expensive. Do not use a water type extinguisher. Fuel is lighter than water and could be spread without being extinguished.

In case of an emergency, the fuel truck if used may need to be quickly driven away from the area. For this reason alone, it should be positioned correctly on the ramp relative to the aircraft.

FUEL CONTAMINATION

Continuous vigilance is required when checking fuel systems for contaminants. Daily draining of strainers and sumps is combined with periodic filter changes and inspections to ensure fuel is contaminant free.

Visual inspection of fuel should always reveal a clean, bright looking liquid. Fuel should not be opaque, which could be a sign of contamination and demands further investigation. Any suspicion of contamination must be investigated. However, visual inspection and sumping alone is not sufficient. Particles are suspended longer in jet fuel due to its viscosity. Technicians must supplement these with cautious procedures and thorough inspections to accomplish the overall goal of delivering clean fuel to the engines.

Keeping a fuel system clean begins with an awareness of the common types of contamination. Water is the most common. Solid particles, surfactants, and microorganisms are also common. However, contamination of fuel with another fuel not intended for use on a particular aircraft is possibly the worst type of contamination.

Water

Water can be dissolved into fuel or entrained. Entrained water can be detected by a cloudy appearance to the fuel. The cloudiness caused by water in the fuel tends to be more towards the bottom of the tank as the water slowly settles down. Water can enter a fuel system via condensation. Water vapor present in the space above the liquid fuel in a fuel tank condenses when the temperature changes. It normally sinks to the bottom of the tank into the sump where it can be drained before flight. However, time is required for this to happen. (***Figure 11-51***)

Figure 11-51. A sump drain tool.

The condition of the fuel and recent fueling practices are equally important. If the aircraft has been flown often and filled immediately after flight, there is less reason to suspect water contamination beyond what would be exposed during a routine sumping. An aircraft that has sat for long periods with partially full tanks is a cause of concern. Water may also be introduced into the aircraft during refueling with fuel that already contains water. Any suspected contamination from refueling of the aircraft should be investigated.

Strainers and filters are designed with upward flow exits to collect water at the bottom of the fuel bowl to be drained off. This should not be overlooked. Entrained water in small quantities usually poses no problem. Large amounts can disrupt engine operation. Settled water in tanks can cause corrosion and support microorganisms that live in the fuel/water interface. High quantities of water can also cause discrepancies in fuel quantity probe indications.

In addition to these detection methods, various field and laboratory tests can be performed to expose contamination. A common field test for water contamination is performed by adding a dye that dissolves in water but not fuel. The more water present in the fuel, the greater the dye disperses and colors the sample. Another common test kit contains a gray chemical powder that changes color to pink or purple when the contents of a fuel sample contains more than 30 Parts Per Million (PPM) of water, or 15 PPM for turbine engine fuel. (***Figure 11-52***) These levels of water are considered unsafe for operation. If greater levels are discovered, time for the water to settle out of the fuel should be given or the aircraft should be defueled and refueled with acceptable fuel.

Solid Particle Contaminants

Solid particles that do not dissolve in the fuel are common contaminants. Dirt, rust, dust, metal particles, and just about anything that can find its way into an open fuel tank is of concern. Filter elements are designed to trap these contaminants and some fall into the sump to be drained off. Pieces of debris from the inside of the system may also accumulate such as broken off sealant, pieces of filter elements, corrosion, etc.

Figure 11-52. This kit allows periodic testing for water in fuel.

Coarse sediments are those visible to the naked eye. Should they pass beyond the filters, they can clog fuel metering devices, sliding valves, and fuel nozzles. Fine sediments cannot always be seen as individual particles. They may appear as a haze in the fuel or may refract light when examining the fuel. Their presence in fuel controls and metering devices is indicated by dark shellac-like marks on sliding surfaces. The maximum amount of solid particle contamination allowable is much less in turbine engine fuel systems than in reciprocating engine fuel systems.

Preventing solid contaminant introduction into the fuel is critical. Whenever the fuel system is open, care must be taken to keep out foreign matter. Lines should be capped immediately. Fuel tank caps should not be left open for any longer than required to refuel the tanks. Clean the area adjacent to wherever the system is opened before it is opened.

It is particularly important to regularly replace filter elements and investigate any unusual solid particles that collect therein. The discovery of significant metal particles in a filter could be a sign of a failing component upstream of the filter. A laboratory analysis is possible to determine the nature and possible source of solid contaminants.

Surfactants

Surfactants are liquid contaminants that naturally occur in fuels. They can be introduced during the refining or handling processes. These surface agents usually appear as tan to dark brown liquid when they are present in large quantities, sometimes with a soapy consistency. Surfactants in small quantities are unavoidable and pose little threat.. Larger quantities do pose problems because they reduce the surface tension between water and the fuel and tend to cause water and small particles in the fuel to remain suspended rather than settling into the sumps. Surfactants also tend to collect in filter elements making them less effective.

Surfactants are usually in the fuel when it is introduced into the aircraft. Discovery of either excessive quantities of dirt and water making their way through the system or a sudsy residue in filters and sumps may indicate their presence. Most fuel providers have clay filter elements on their dispensing trucks and dispensing systems. These filters, if renewed at the proper intervals, remove most surfactants through adhesion. (*Figure 11-53*)

Fuel trucks and fuel farms may make use of laser contaminant identification technology. All fuel exiting the storage tank going into the servicing hose is passed through the analyzer. Sensors determine the difference between water and solid particle contaminants. When an excessive level of either is detected, the unit automatically shuts off flow to the nozzle. Thus, aircraft are fueled only with clean dry fuel.

Microorganisms

The presence of microorganisms in turbine engine fuels is a critical problem. There are hundreds of varieties of these life forms that live in free water at the junction of the water and fuel in a fuel tank. They form a visible

Figure 11-53. Clay filter elements.

MICROORGANISUM

Figure 11-54. This fuel-water sample.

slime that is dark brown, gray, red, or black in color. This microbial growth can multiply rapidly and can cause interference with the proper functioning of filter elements and fuel quantity indicators. Moreover, the slimy water/microbe layer in contact with the fuel tank surface provides a medium for electrolytic corrosion of the tank. (***Figure 11-54***)

The presence and level of microorganisms in a fuel tank can also be measured with a field device which detects the metabolic activity of bacteria, yeast, molds and other anaerobe microorganisms. These tests can be used to determine the amount of antimicrobial agent to be added to the fuel. A testing unit is shown in ***Figure 11-55.***

Bug test kits test fuel specifically for bacteria and fungus. While other types of microrganisms may exist, this test is quick and easy to perform. Treat a fuel sample with the product and match the color of the sample to the chart for an indication of the level of bacteria and fungus present. These are the most common types of microorganisms that grow in fuel. If levels of fungus and bacteria are acceptable, the fuel could be usable. (***Figure 11-56***) When surfactant filters are combined with contaminant identification technology and microorganism detection, chances of delivering clean fuel to the aircraft engines are good.

Figure 11-55. A capture solution.

Figure 11-56. Fuel bug test kits.

Improper Fuel

Aircraft engines operate effectively only with the proper fuel. Contamination with fuel not intended for that particular aircraft can have disastrous consequences. It is the responsibility of the technicians to put forth effort continuously to ensure that only the fuel designed for the operation of the aircraft engine(s) is put into the fuel tanks. Each fuel tank receptacle or fuel cap area is clearly marked to indicate which fuel is required. (***Figure 11-57***)

If the wrong fuel is put into an aircraft, the situation must be rectified before flight. If discovered before the fuel pump is operated and an engine is started, drain

Figure 11-57. All entry points of fuel into the aircraft.

all improperly filled tanks. Flush out the tanks and fuel lines with the correct fuel and then refill the tanks with the proper fuel.

However, if discovered after an engine has been started or attempted to be started, the procedure is more in depth. The entire fuel system, including all fuel lines, components, metering device(s) and tanks, must be drained and flushed. If the engines have been operated, a compression test should be done, and the combustion chamber and pistons should be borescope inspected. Engine oil should be drained, and all screens and filters examined for any evidence of damage. Once reassembled and the tanks have been filled with the correct fuel, a full engine run-up check should be performed before releasing the aircraft for flight.

Laboratory Testing

Before various test kits were developed for use in the field, laboratories provided complete fuel composition analysis. These services are still available. A sample is sent in a sterilized container to the lab where it can be tested for numerous factors including water, microbial growth, flash point, specific gravity, cetane index (a measure of combustibility), and more. Tests for microbes can also involve growing cultures of whatever organisms are present in the fuel. (***Figure 11-58***)

Figure 11-58. Laboratory tests of fuel samples.

Question: 11-1
Name three safety steps which must be taken every time one is working through an access panel on an integral fuel tank.

Question: 11-2
In what scenario, would a fuel bypass valve be turned on while a helicopter is in flight?

Question: 11-3
What is the primary purpose of vents in a fuel tank?

Question: 11-4
What is the main purpose of primer pumps in a helicopter fuel system?

Question: 11-5
If a fuel line fitting is found to be leaking, what is the first step for determining its cause?

Question: 11-6
If the measured pressure within a fuel line is greater at the inlet of a filter than its outlet, what is the likely reason and what occurs?

Question: 11-7
What is the purpose of a check valve within a drain valve?

Question: 11-8
What are two factors that can routinely effect the accuracy of a capacitance type fuel indicator system?

ANSWERS

Answer: 11-1
All fuel must be removed from the tank; respiratory equipment must be worn; a second safety lookout person must be present.

Answer: 11-2
In case of a fuel pump failure, so a working pump in one tank can feed both engines.

Answer: 11-3
To prevent a vacuum (or negative pressure) from being formed inside the tank as fuel is burned.

Answer: 11-4
To purge air from the fuel lines during engine start-up.

Answer: 11-5
Check that it is tightened to the proper torque.

Answer: 11-6
The filter is blocked. A filter bypass valve opens. (dirty fuel is better than no fuel).

Answer: 11-7
So that a leaky drain valve may be removed without draining the fuel in the tank.

Answer: 11-8
The fuel's temperature; variations in the dielectric constant of different turbine fuels.

HELICOPTER AERODYNAMICS, STRUCTURES AND SYSTEMS

HYDRAULIC POWER (ATA 29)

SUB-MODULE 12

PART-66 SYLLABUS LEVELS
CERTIFICATION CATEGORY – B1.3 B1.4

Sub-Module 12
HYDRAULIC POWER (ATA 29)
Knowledge Requirements

12.12 - Hydraulic Power (ATA 29) 3

System layout;
Hydraulic fluids;
Hydraulic reservoirs and accumulators;
Pressure generation: electric, mechanical, pneumatic; Emergency pressure generation;
Filters;
Pressure control;
Power distribution;
Indication and warning systems;
Interface with other systems.

12.12 - HYDRAULIC POWER

The word "hydraulic" is based on the Greek word for water, meaning the study of the physical behavior of water at rest and in motion. Today, the meaning has been expanded to the physical behavior of all liquids, including hydraulic fluid. Hydraulic systems are not new to aviation. Early aircraft had hydraulic brake systems. As aircraft became more sophisticated, newer systems with hydraulic power were developed. Hydraulic systems provide a means for the operation of aircraft components such as landing gear, flaps, flight control surfaces, brakes and more. To achieve the necessary reliability, hydraulic systems may consist of several subsystems. Each subsystem has a power generating device (pump), reservoir, accumulator, heat exchanger, filtering system, etc. System operating pressure may vary from a couple hundred pounds per square inch (psi) in small rotorcraft to 5 000 psi in large helicopters.

Hydraulic systems have many advantages as power sources for operating various aircraft units; combining the advantages of light weight, ease of installation, simplification of inspection, and minimum maintenance. Hydraulic operations are also almost 100 percent efficient, with only negligible loss due to fluid friction.

SYSTEM LAYOUT

Regardless of its function and design, every hydraulic system has a minimum number of basic components in addition to a means through which the fluid is transmitted. A basic system consists of a pump, reservoir, directional valve, check valve, pressure relief valve, selector valve, actuator, and filter. (***Figure 12-1***)

OPEN CENTER HYDRAULIC SYSTEMS

An open center system is one having fluid flow, but no pressure in the system when the actuating mechanisms are idle. The pump circulates the fluid from the reservoir, through the selector valves, and back to the reservoir. (***Figure 12-2***)

An open center system may employ any number of subsystems with a selector valve for each. Unlike closed center systems, the selector valves of open systems are always connected in series with each other. In this arrangement, fluid is always allowed free passage through each selector valve and back to the reservoir until one of the selector valves is positioned to operate a mechanism. When one of those selector valves is positioned to operate a device, fluid is directed from the pump through one of the working lines to that actuator. (***Figure 12-2B***)

With that selector valve so positioned, the flow of fluid through the valve to the reservoir is blocked. The pressure builds up in the system to overcome the resistance and moves the piston of the actuating cylinder. Fluid from the opposite end of the actuator then returns to the

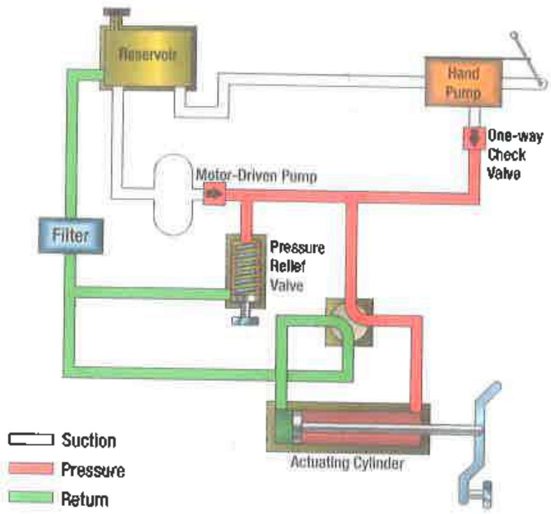


Figure 12-1. Basic hydraulic system.

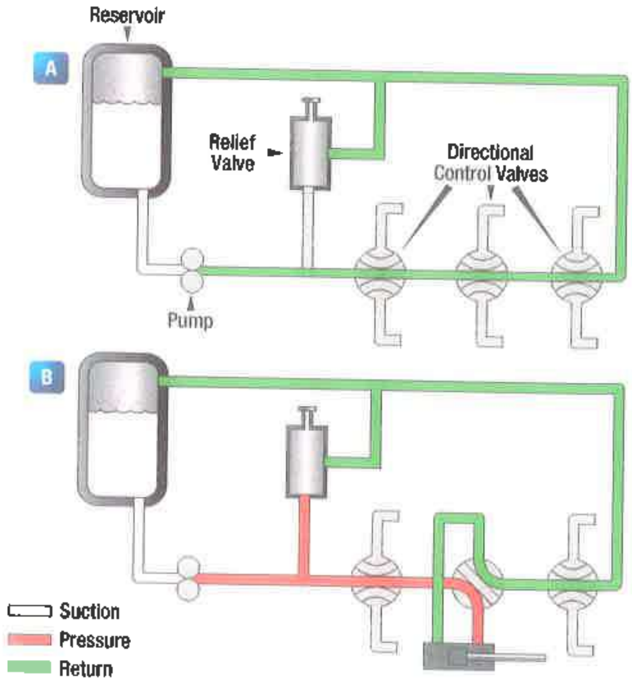


Figure 12-2. Open center hydraulic system.

selector valve and flows back to the reservoir. Operation of the system following actuation of the component depends on the type of selector valve being used.

Several types of selector valves are used in conjunction with the open center system. One type is both manually engaged and manually disengaged. First, the valve is manually moved to an operating position. When the actuating mechanism reaches the end of its stroke, the pump output continues until the relief valve relieves the pressure. The relief valve unseats and allows the fluid to flow back to the reservoir. The system pressure remains at the relief valve set pressure until the selector valve is manually returned to the neutral position. This action reopens the open center flow and allows the system pressure to drop to line resistance pressure.

The manually engaged and pressure disengaged type of selector valve is like the valve previously discussed. When the actuating mechanism reaches the end of its cycle, the pressure continues to rise to a predetermined pressure. When reached, the valve automatically returns to the neutral position.

CLOSED CENTER HYDRAULIC SYSTEMS

In the closed center system, as depicted in ***Figure 12-3***, the fluid is under pressure whenever the power pump is operating. Three actuators are arranged in parallel with actuating units B and C operating while actuating unit A is not. This system differs from the open center system in that the selector or directional control valves are arranged in parallel and not in series.

The means of controlling pump pressure varies in the closed center system. If a constant delivery pump is used, the system pressure is regulated by a pressure regulator. A relief valve acts as a backup safety device in case the regulator fails. If a variable displacement pump is used, system pressure is controlled by the pump's pressure mechanism compensator. The compensator automatically varies the volume output. When pressure approaches the normal system pressure, the compensator begins to reduce the flow output of the pump. The pump is fully compensated (near zero flow) when normal system pressure is attained. When the pump is in this fully compensated condition, its internal bypass mechanism provides fluid circulation through the pump for cooling and lubrication. A relief valve is installed in the system as a safety backup.

An advantage of the open center system over the closed center system is that the continuous pressurization of the system is eliminated. Since the pressure is built up gradually after the selector valve is moved to an operating position, there is very little shock from pressure surges. This action provides a smoother operation of the actuating mechanisms. However, the operation is slower than the closed center system, in which pressure is available the moment the selector valve is positioned. Since most aircraft applications require instantaneous operation, closed center systems are the most widely used.

EVOLUTION OF HYDRAULIC SYSTEMS

Not all helicopters use hydraulic systems, particularly the very light ones. For those, all controls are man powered. When helicopters came to increase in size, the forces needed to control the rotor head became too great for the pilot to operate on his own. For these circumstances, hydraulic actuators were inserted in the flight control mechanisms to provide the pilot assistance to overcome these higher aerodynamic loads. Hydraulic

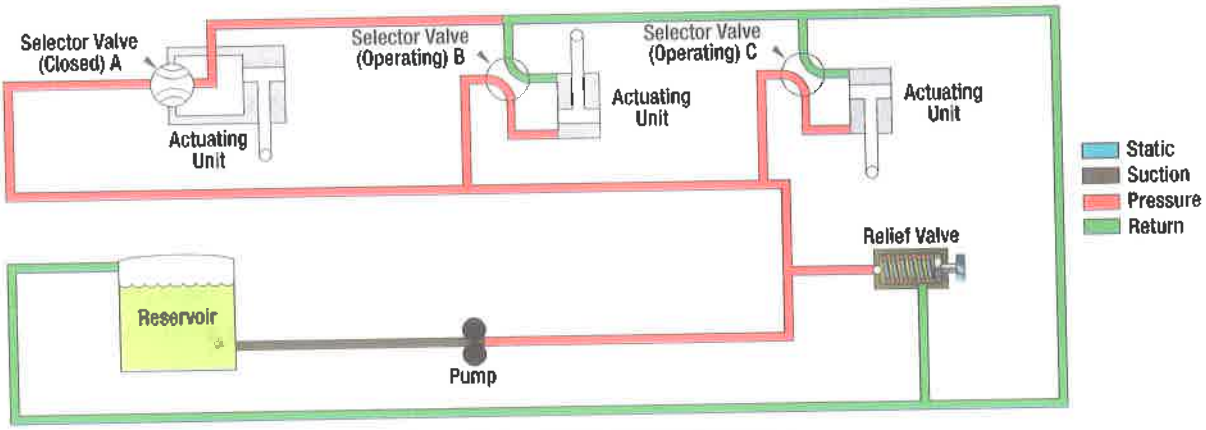


Figure 12-3. A closed-center hydraulic system.

boost systems thus increase the pilot's comfort, since the 'feel' of a hydraulically controlled helicopter is less stressful. Vibrations and other mechanical forces also become less pronounced for the pilot's controls than if they were mechanically linked to the rotor head.

Hydraulic Power Pack System

A hydraulic power pack is a compact unit that consists of an electric pump, a reservoir, valves, filters, and pressure relief valve in one assembly. (***Figure 12-4***) The advantage of the power pack is that there is no need for a centralized hydraulic supply system along with long stretches of hydraulic lines. This reduces weight. Power packs are driven by either an engine gearbox or electric motor. Integration of essential valves, filters, sensors, and transducers virtually eliminates external leakage and simplifies troubleshooting. Some power pack systems have an integrated actuator. These systems are used to control the stabilizer trim, landing gear, or flight control systems directly, thus eliminating the need for a centralized system.

Modern High Performance Systems

Modern helicopters use a power supply system and a fly-by-wire system. A fly-by-wire system places the helicopter under full time computer control and so maximizes safety by reducing pilot workload, increasing pilot situational awareness, and improving aircraft handling qualities. The Bell 525 was the first civil helicopter equipped with fly-by-wire flight controls and has a capacity to carry 16 to 20 passengers. In such a system, the pilot's inputs are electronically sent to the flight control servos which use hydraulic pressure to move the control system. Lengthy cables or push rods are then eliminated. With additional advances, some manufacturers are reducing the use of hydraulic systems in favor of electrically controlled systems.

Figure 12-4. Hydraulic power pack.

HYDRAULIC FLUIDS

Hydraulic system fluids transmit and distribute forces to various units to be actuated. Liquids can do this because they are almost incompressible. Pascal's Law states that pressure applied to any part of a confined liquid is transmitted with undiminished intensity to every other part. Thus, if several passages exist in a system, pressure can be distributed through all of them by means of the liquid. If incompressibility was the only quality required, any fluid that is not too thick could be used. However, a satisfactory fluid for a particular installation must possess several other properties. Thus manufacturers of hydraulic devices usually specify the type of fluid best suited for their equipment considering working conditions, service required, expected inside and outside temperatures, system pressures, the possibilities of corrosion, and other conditions.

HYDRAULIC FLUID PROPERTIES

Viscosity

One of the most important properties of any hydraulic fluid is its viscosity. The viscosity of a fluid is a measure of its resistance to deformation at a given rate. For liquids, it corresponds to the informal concept of "thickness". A liquid such as gasoline with a low viscosity flows easily while tar, with a high viscosity, flows slowly.

A satisfactory hydraulic fluid must have enough body to give a good seal at pumps, valves and pistons, but it must not be so thick that it resists flow, which would lead to power loss and high operating temperatures. These factors add to the load and excessive wear of parts. A fluid that is too thin also leads to rapid wear of moving parts or of parts that have heavy loads. The instrument used to measure the viscosity of a liquid is known as a viscosimeter.

Viscosity also increases as temperature decreases. The Saybolt viscosimeter (***Figure 12-5***) measures the time required, in seconds, for 60 milliliters of the tested fluid at a given temperature to pass through a standard calibrated tube. The time measured is used to express the fluid's viscosity, in Saybolt universal viscosity (SUV) or Saybolt FUROL viscosity. FUROL is an acronym for fuel and road oil.

Chemical Stability

Chemical stability is another property that is exceedingly

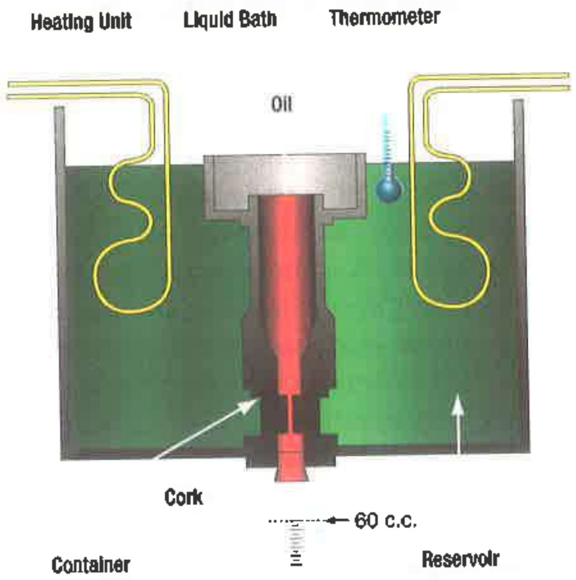


Figure 12-5. Saybolt viscosimeter.

important in selecting a hydraulic fluid. It is the fluid's ability to resist oxidation and deterioration for long periods. All fluids tend to undergo unfavorable chemical changes under severe operating conditions. Excessive temperatures have a great effect on the life of a fluid. It should be noted that the temperature of the fluid in the reservoir of a hydraulic system does not always represent a true state of operating conditions. Localized hot spots occur on bearings, gear teeth, or at the point where liquid under pressure is forced through a small orifice. Continuous passage of a liquid through these points may produce local temperatures high enough to carbonize or sludge the liquid, yet the liquid in the reservoir may not be equally affected. Fluids with a high viscosity have a greater resistance to heat than low viscosity liquids that have been derived from the same source. The average hydraulic liquid has a low viscosity, however there is a wide choice of fluids available for use within the viscosity range required by the manufacturer.

Fluids may break down if exposed to air, water, salt, or other impurities, especially if they are in constant motion or subject to heat. Some metals, such as zinc, lead, brass, and copper, have an undesirable chemical reaction on certain fluids. These chemical processes result in the formation of sludge, gums, and carbon or other deposits that clog openings, cause valves and pistons to stick or leak, and give poor lubrication to moving parts. As soon as small amounts of sludge or other deposits are formed, the rate of formation generally accelerates. As they are formed, certain changes in the physical and chemical properties of the fluid take place. The fluid usually becomes darker in color, higher in viscosity, and acids are formed.

Flash Point

Flash point is the lowest temperature at which a liquid gives off vapor in sufficient quantity to ignite momentarily or flash when a flame or a spark is applied. A high flash point is desirable for hydraulic fluids because it indicates good resistance to combustion and a low degree of evaporation at normal temperatures.

Fire Point

Fire point is the lowest temperature at which the vapor keeps burning after the ignition source is removed. It is higher than the flash point because at the flash point vapor may not be produced fast enough to sustain combustion. In general, the fire point can be assumed to be about 10°C higher than the flash point. Neither flash nor fire point depends directly on the ignition source temperature, but ignition source temperature is far higher than either the flash or fire point. Like flash point, a high fire point is required for hydraulic fluids.

TYPES OF HYDRAULIC FLUIDS

To assure proper system operation and to avoid damage to nonmetallic components of the hydraulic system, the correct fluid must be used. When adding fluid to a system, use the type specified in the aircraft manufacturer's maintenance manual or on the instruction plate affixed to the reservoir of the unit being serviced.The three principal categories of hydraulic fluids are:

- Mineral oil base fluids.
- Polyalphaolefins (Synthetic hydrocarbon base fluids).
- Phosphate Ester base fluids.

When servicing a hydraulic system, the technician must be certain to use the correct category of replacement fluid. Hydraulic fluids are not necessarily compatible. For example, mixing the fire resistant fluid MIL-H-83282 with MIL-H-5606 may render the MILH-83282 non fire resistant.

Mineral Base Fluids

Mineral oil based fluids such as MIL-H-5606 is the oldest, dating back to the 1940s. It is used in many systems, especially where the fire hazard is comparatively low. MIL-H-6083 is simply a rust inhibited version of MIL-H-5606. They are completely interchangeable.

Suppliers generally ship hydraulic components with MIL-H-6083. Mineral based hydraulic fluids are processed from petroleum. They have an odor similar to penetrating oil and are dyed red. Synthetic rubber seals are used with petroleum based fluids.

Polyalphaolefin Base Fluids

MIL-H-83282 is a fire resistant hydrogenated polyalphaolefin based fluid developed in the 1960s to overcome the flammability characteristics of MIL-H-5606. MIL-H-83282 is significantly less flammable, but has a disadvantage of high viscosity at low temperature. It is generally limited to -40°C. However, it can be used in the same system and with the same seals, gaskets, and hoses as MIL-H-5606. MIL-H-46170 is the rust inhibited version of MIL-H-83282. Small aircraft predominantly use MIL-H-5606, but some have switched to MIL-H-83282 if they can accommodate the high viscosity at low temperature. MIL-H-87257 is a development of MIL-H-83282 fluid to improve its low temperature viscosity.

Phosphate Ester Base Fluids

Phosphate Ester fluids are used in most transport aircraft and are extremely fire resistant. However, they are not fireproof and under certain conditions, they can burn. Progressive development of these fluids occurred because of the requirements of newer aircraft. Airframe manufacturers classified these new generations of hydraulic fluid types based on their performance. Today, types IV and V fluids are used.

Two distinct classes of type IV fluids exist based on their density. Type IV class 1 fluids are low density and Type IV class 2 fluids are standard density. Thus, class 1 fluids provide weight savings versus class 2. Type V fluids were developed in response to demands for a more thermally stable fluid at higher operating temperatures and will be more resistant to hydrolytic and oxidative degradation at high temperature than the type IV fluids.

The Skydrol® series of phosphate ester fluids were developed by the Douglas Aircraft Company and Monsanto to reduce the fire risk from leaking high pressure mineral oil based hydraulic fluids. Skydrol® is made of a fire resistant phosphate ester base stock, with oil additives dissolved into it to inhibit corrosion and erosion damage to servo valves. It also includes a purple or green dye for identification. Skydrol® has been approved by most airframe manufacturers including Airbus and Boeing and has been used in their products for over 40 years.

Intermixing of Fluids

Due to the difference in composition, petroleum base and phosphate ester fluids will not mix. Neither are the seals for any one fluid usable with the other fluids. Should an aircraft hydraulic system be serviced with the wrong type of fluid, immediately drain and flush the system and maintain the seals according to the manufacturer's specifications.

Compatibility With Aircraft Materials

Aircraft hydraulic systems designed around Skydrol® fluids should be virtually trouble free if properly serviced. Skydrol® does not appreciably affect common aircraft metals such as aluminum, silver, zinc, magnesium, cadmium, iron, stainless steel, bronze, chromium, as long as the fluids are kept free of contamination. Due to its phosphate ester base, thermoplastic resins including vinyl, nitrocellulose lacquers, oil based paints, and linoleums may be softened chemically by Skydrol® fluids. However, this chemical action usually requires longer than just momentary exposure. Spills that are quickly wiped up with soap and water do not harm most of these materials. Paints that are Skydrol® resistant include epoxies and polyurethanes.

Hydraulic systems require the use of special accessories that are compatible with each fluid. Appropriate seals, gaskets, and hoses must be specifically chosen for the type of fluid used. When gaskets, seals, and hoses are replaced, positive identification should be made to ensure that they are made of the appropriate material. Skydrol® type V fluid is compatible with natural fibers and with several synthetics, including nylon and polyester, which are used extensively in most aircraft. Petroleum oil hydraulic system seals of neoprene or Buna-N are not compatible with Skydrol® and must be replaced with seals of butyl rubber or ethylene propylene elastomers.

HYDRAULIC RESERVOIRS AND ACCUMULATORS

RESERVOIRS

The reservoir is a tank in which is stored an adequate supply of fluid for the system. Fluid flows from the reservoir to the pump where it is forced through the system and eventually returned to the reservoir. The reservoir not only supplies the operating needs of the system, but also replenishes fluid lost through leakage. Furthermore, the reservoir serves as an overflow basin for fluid forced out of the system by thermal expansion (the increase of fluid volume caused by temperature changes), the accumulators, and by piston and rod displacement. It also provides a place for the fluid to purge itself of air bubbles that may enter the system. Foreign matter picked up in the system may also be separated in the reservoir or as it flows through line filters.

Reservoirs are either pressurized or non-pressurized. Baffles and/or fins are incorporated in most reservoirs to keep the fluid within the reservoir from having random movement such as vortexing (swirling) and surging. These conditions can cause the fluid to foam and air to enter the pump along with the fluid.

Many reservoirs incorporate strainers in the filler neck to prevent the entry of foreign matter during servicing. These strainers are made of fine mesh screening and are usually referred to as finger strainers because of their shape. Finger strainers should never be removed or punctured as a means of speeding up the pouring of fluid into the reservoir. Reservoirs could have an internal trap to make sure that fluid goes to the pumps during negative G conditions.

Most aircraft have emergency hydraulic systems that take over if the main system fails. In many such systems, the pumps of both systems obtain fluid from a single reservoir. Under such circumstances, a supply of fluid for the emergency pump is ensured by drawing the hydraulic fluid from the bottom of the reservoir. The main system draws its fluid through a standpipe located at a higher level. With this arrangement, should the main system fluid supply become depleted, adequate fluid is left for operation of the emergency system.

Figure 12-6 illustrates that the Engine Driven Pump is not able to draw fluid anymore if the reservoir gets depleted below the standpipe. The alternating current motor driven pump still has a supply of fluid for emergency operations.

Non Pressurized Reservoirs

Most non pressurized reservoirs are constructed in a cylindrical shape. The outer housing is manufactured from a strong corrosion resistant metal. Filter elements

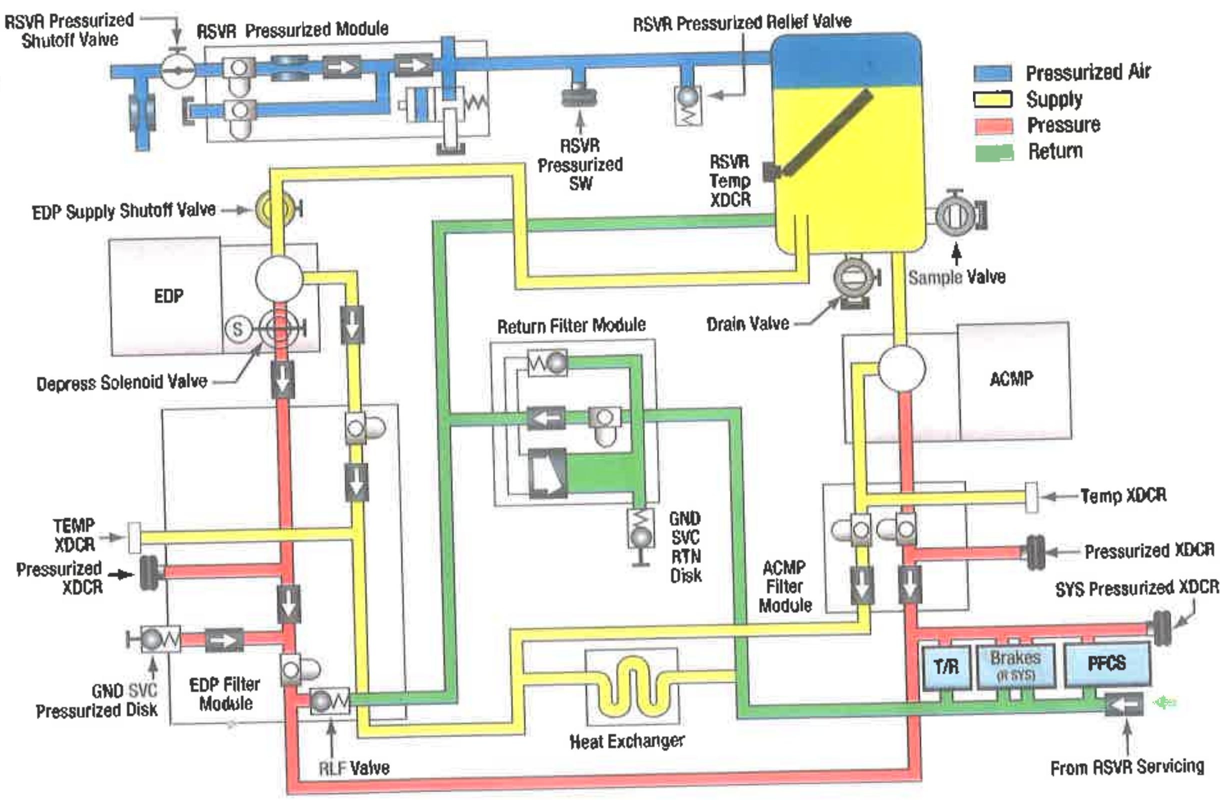


Figure 12-6. Hydraulic reservoir standpipe.

are normally installed within the reservoir to clean returning fluid. In some of the older aircraft, a filter bypass valve allows fluid to bypass the filter in the event the filter becomes clogged. Reservoirs can be serviced by pouring fluid directly into the reservoir through a filler strainer incorporated within the filler well to filter out impurities as the fluid enters the reservoir. Generally, non pressurized reservoirs use a visual gauge to indicate the fluid quantity. Gauges incorporated on or in the reservoir may be a direct reading glass tube or a float type rod that is visible through a transparent dome. In some cases, the fluid quantity may also be read in the cockpit using quantity transmitters. A typical non pressurized reservoir is shown in ***Figure 12-7.***

This reservoir consists of a welded body and cover assembly clamped together. Gaskets are incorporated to seal against leakage between assemblies. Non pressurized reservoirs are slightly pressurized due to thermal expansion of fluid and the return of fluid to the reservoir from the main system. This pressure ensures that there is a positive flow of fluids to the inlet ports of the hydraulic pumps. Most reservoirs of this type are vented directly to the atmosphere or cabin with only a check valve and filter to control the outside air source. The reservoir system includes a pressure and vacuum relief valve. The purpose of the valve is to maintain a differential pressure range between the reservoir and cabin. A manual air bleed valve is installed on top to vent the reservoir. The valve is connected to the reservoir vent line to allow depressurization of the reservoir. The valve is actuated prior to servicing the reservoir to prevent fluid from being blown out of the filler as the cap is being removed. The manual bleed valve also needs to be actuated if hydraulic components need to be replaced.

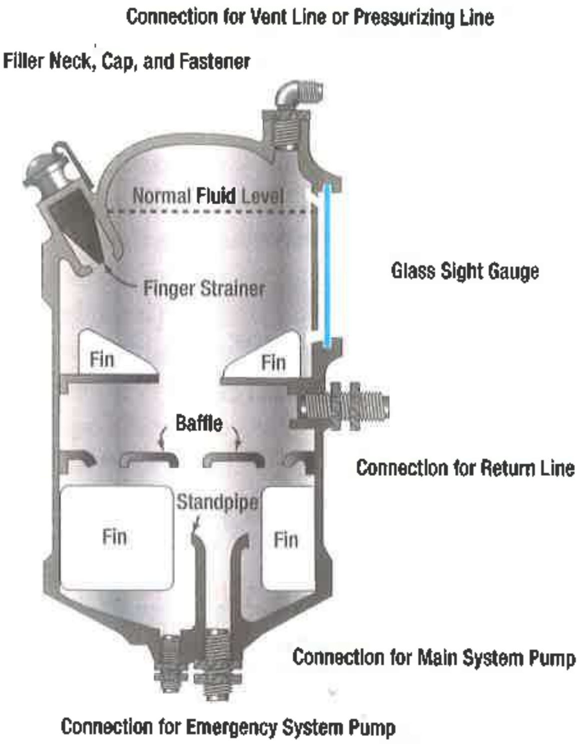


Figure 12-7. Non-pressurized reservoir.

Pressurized Reservoirs

Pressurizing assures a positive flow of fluid to the pump at high altitudes when low atmospheric pressures are encountered. On some aircraft, the reservoir is pressurized by bleed air taken from the compressor section of the engine. On others, the reservoir may be pressurized by the hydraulic system pressure.

Air Pressurized Reservoirs

- Air pressurized reservoirs are used in many aircraft. (***Figure 12-8 and Figure 12-9***) Pressurization is required because the reservoirs are often located near the main gear box or other non pressurized areas. Engine bleed air is used to pressurize the reservoir. The reservoirs are typically cylindrical in shape. The following components are installed on a typical reservoir:
  - Reservoir pressure relief valve - prevents over pressurization of the reservoir. Valve opens at a preset value.

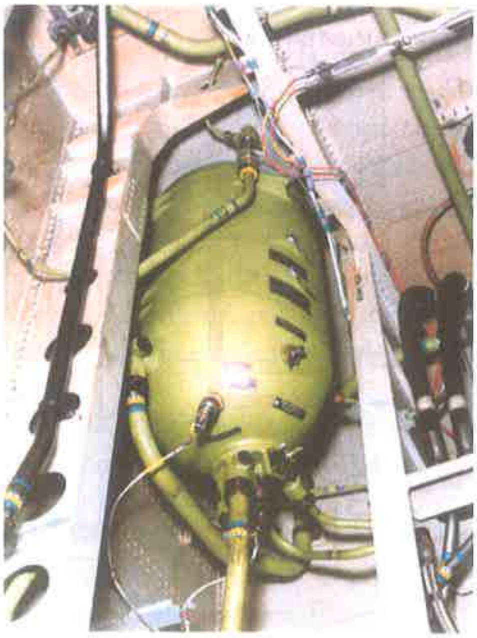


Figure 12-8. Air-pressurized reservoir.

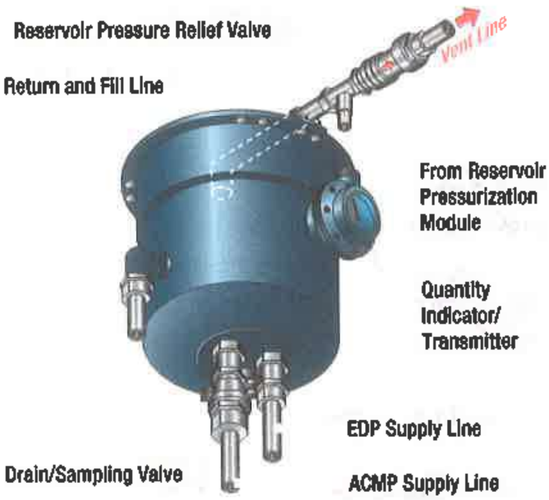


Figure 12-9. Components of an air-pressurized reservoir.

- Sight glasses (low and overfull) - provides visual indication for flight crews and maintenance personnel that the reservoir needs to be serviced.
- Reservoir sample valve - used to draw a sample of hydraulic fluid for testing.
- Reservoir drain valve - used to drain the fluids out of the reservoir for maintenance operation.
- Reservoir temperature transducer - provides hydraulic fluid temperature information for the flight deck. (***Figure 12-10***)
- Reservoir quantity transmitter - transmits fluid quantity to the flight deck so that the flight crew can monitor fluid quantity during flight. (***Figure 12-10***)

A reservoir pressurization module is installed close to the reservoir to supply bleed air. (***Figure 12-11***) The module typically consists of the following parts:

- Filters (2)
- Check Valves (2)
- Test Port
- Manual Bleed Valve
- Gauge Port

A manual bleeder valve is incorporated into the module. During hydraulic system maintenance, it is necessary to relieve reservoir air pressure to assist in the installation and removal of components, lines, etc. This type of valve is small in size and has a push button installed in the outer case. When the button is pushed, pressurized air from the reservoir flows through the valve to an overboard vent until the pressure is depleted or the button is released. When the button is released, the internal spring causes the poppet to return to its seat. As some hydraulic fluid can escape from the manual valve when the button is depressed, use caution by putting a rag around the valve to catch hydraulic fluid spray. Hydraulic fluid spray can cause injuries.

Fluid Pressurized Reservoirs

Some reservoirs are pressurized by hydraulic system pressure. Regulated hydraulic pump output pressure is applied to a movable piston inside the reservoir. This small piston is attached to and moves a larger piston against the reservoir fluid. The reduced force of the small piston when applied by the larger piston is adequate to provide head pressure for high altitude operation. The


Figure 12-10. Temperature and quantity sensors.

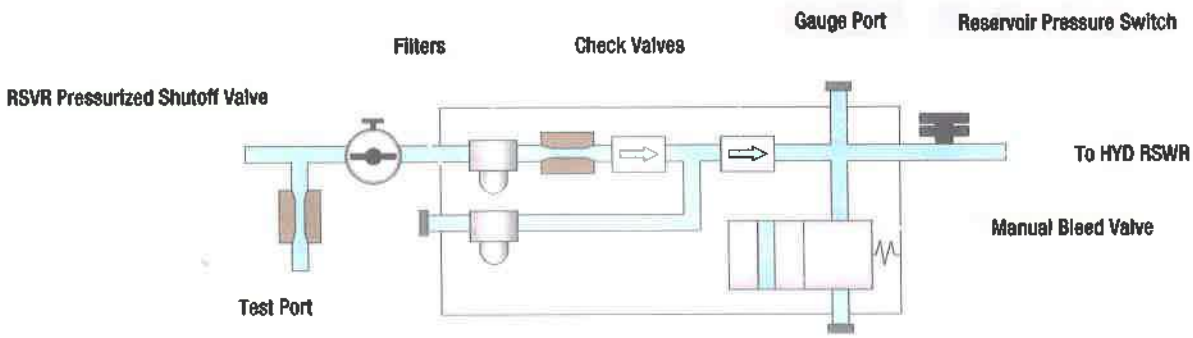


Figure 12-11. Reservoir pressurization module.

small piston protrudes from the body of the reservoir. The amount exposed is used as a reservoir fluid quantity indicator. ***Figure 12-12*** illustrates the concept behind the fluid pressurized reservoir.

The reservoir has five ports: pump suction, return, pressurizing, overboard drain, and bleed ports. Fluid is supplied to the pump through the suction port and returns to the reservoir from the return port. Pressure from the pump enters the pressurizing cylinder in the top of the reservoir through the pressurizing port. The overboard drain port drains the reservoir, when necessary, such as while performing maintenance. Bleed ports also aid in servicing the reservoir. When servicing this type of reservoir, place a container under the bleed drain port. The fluid should then be pumped into the reservoir until air free fluid flows through the bleed drain port. The reservoir fluid level is indicated by the markings on the part of the pressurizing cylinder that moves through the reservoir dust cover assembly.

There are three fluid level markings indicated on the cover: full at zero system pressure (FULL ZERO PRESS), full when the system is pressurized (FULL SYS PRESS), and REFILL. When the system is unpressurized and the pointer on the reservoir lies between the two full marks, a marginal reservoir fluid level is indicated. When the system is pressurized and the pointer lies between REFILL and FULL SYS PRESS, a marginal reservoir fluid level is also indicated.

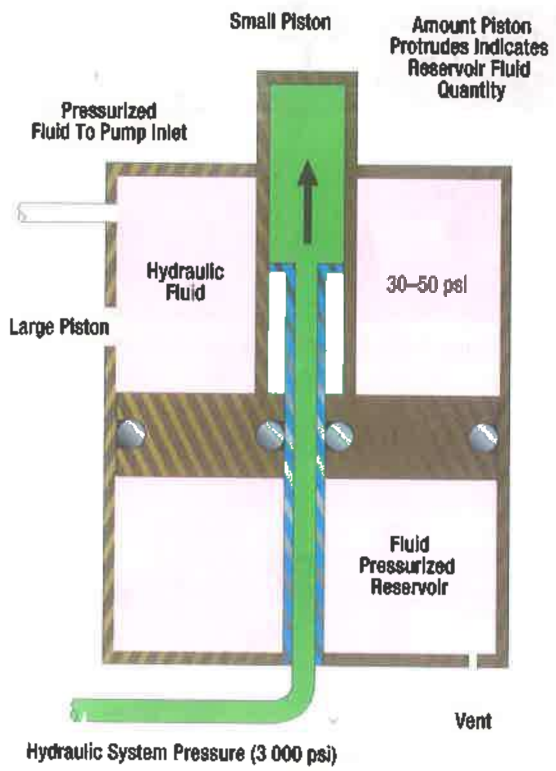


Figure 12-12. Operating principle.

Reservoir Servicing

Non pressurized reservoirs can be serviced by pouring fluid directly into the reservoir through a filler strainer assembly incorporated within the filler well to strain out impurities as the fluid enters the reservoir. Many reservoirs also have a quick disconnect service port at the bottom of the reservoir. A hydraulic filler unit can be connected to the service port to add fluid to the reservoir. This method reduces the chances of contamination of the reservoir. Aircraft that use pressurized reservoirs often have a central filling station in the ground service bay to service all reservoirs from a single point. (***Figure 12-13***)

ACCUMULATORS

The accumulator is a steel sphere divided into two chambers by a synthetic rubber diaphragm. The upper chamber contains fluid at system pressure, while the lower chamber is charged with nitrogen or air. Cylindrical types are also used in high pressure hydraulic systems. Many aircraft have several accumulators in the system including a main system accumulator and an emergency system accumulator. There may also be auxiliary accumulators located in various sub-systems.

The functions of an accumulator are:

1. Dampen pressure surges in the hydraulic system caused by actuation of a unit and the effort of the pump to maintain pressure at a preset level.
2. Aid or supplement the power pump when several units are operating at once by supplying extra power from its accumulated, or stored, power.
3. Store power for the limited operation of a hydraulic unit when the pump is not operating.
4. Supply fluid under pressure to compensate for small leaks that would cause the system to cycle continuously by action of the pressure switches continually kicking in.

Types of Accumulators

There are two general types of accumulators used in aircraft hydraulic systems: spherical and cylindrical.

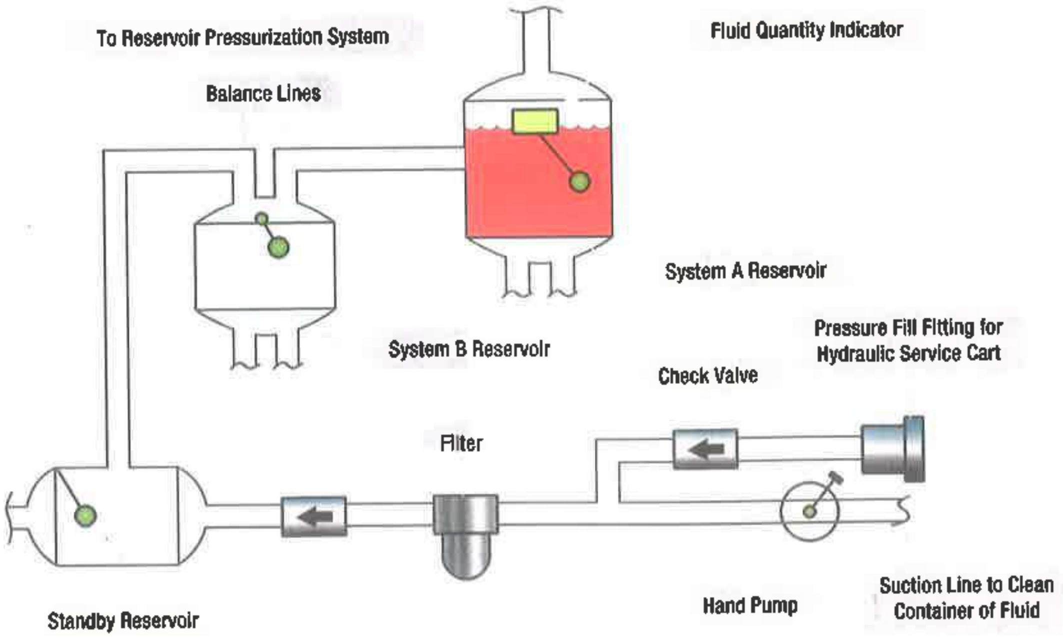


Figure 12-13. A hydraulic ground service station.

Spherical

The spherical type accumulator is constructed in two halves that are threaded, or welded together. Two threaded openings exist. The top port accepts fittings to connect to the pressurized system to the accumulator. The bottom port is fitted with a gas servicing valve. A synthetic rubber diaphragm, or bladder, is installed in the sphere to create two chambers. Pressurized hydraulic fluid occupies the upper chamber and nitrogen or air charges the lower chamber. A screen at the fluid pressure port keeps the diaphragm from extruding through the port when the lower chamber is charged and fluid pressure is zero. A rigid button or disc may also be attached to the diaphragm or bladder for this purpose. The bladder is installed through a large opening in the bottom of the sphere and secured with a threaded retainer plug. The gas servicing valve mounts into the retainer plug. (***Figure 12-14***)

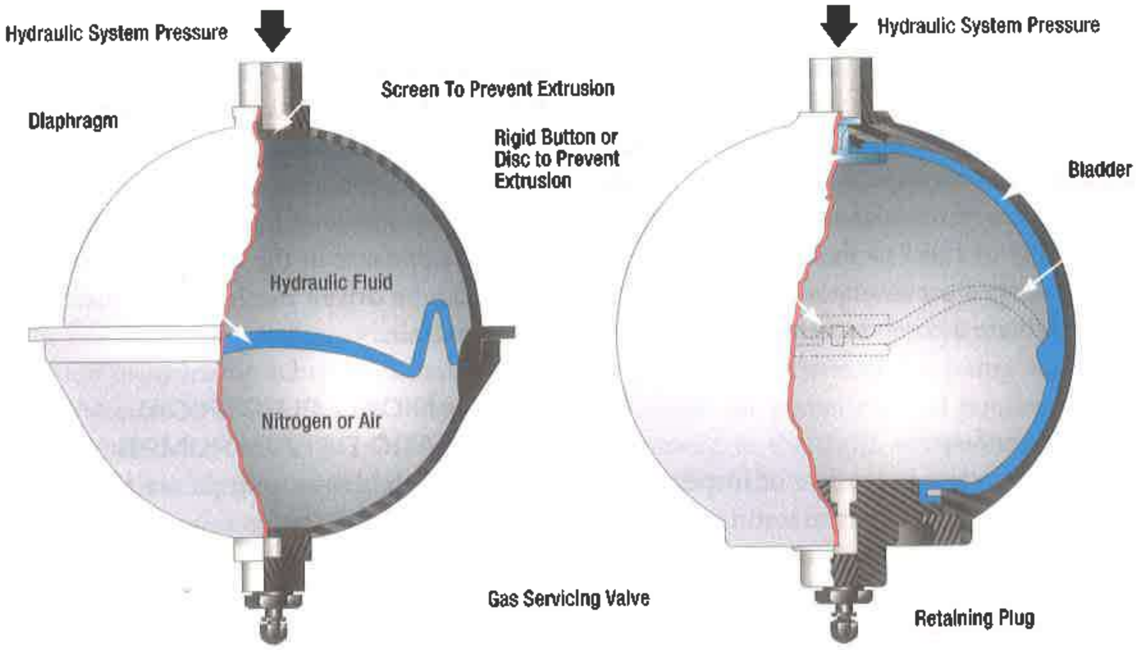


Figure 12-14. A spherical accumulator.

Cylindrical

Cylindrical accumulators consist of a cylinder and piston assembly. End caps are attached to both ends of the cylinder. The internal piston separates the fluid and air/nitrogen chambers. The end caps and piston are sealed with gaskets and packing to prevent external leakage or internal leakage between the chambers. In one end cap, a hydraulic fitting is used to attach the fluid chamber to the hydraulic system. In the other end cap, a filler valve is used to perform the same function as that valve in the spherical accumulator. (***Figure 12-15***)

In operation, the compressed air chamber is charged to a predetermined pressure that is somewhat lower than the system operating pressure. This initial charge is referred to as the accumulator preload. As an example, let us assume that the cylindrical accumulator is designed for a preload of 1 300 psi in a 3 000 psi system. When the initial charge of 1 300 psi is introduced into the unit, hydraulic system pressure is zero. As air pressure is applied through a gas servicing valve, it moves the piston toward the opposite end until it bottoms. If the air behind the piston has a pressure of 1 300 psi, the hydraulic system pump has to create a pressure within the system greater than 1 300 psi before the hydraulic fluid can actuate the piston.

- At 1 301 psi the piston starts to move within the cylinder, compressing the air as it moves.
- At 2 000 psi, it has backed up several inches.
- At 3 000 psi, the piston has backed up to its normal operating position, compressing the air until it occupies a space less than one half the length of the cylinder.

When actuation of the hydraulic unit lowers the system pressure, the compressed air expands against the piston, forcing fluid from the accumulator. This supplies an instantaneous supply of fluid to the hydraulic system component. The charged accumulator may also supply fluid pressure to actuate a component(s) briefly in case of pump failure.

Maintenance of Accumulators

Maintenance of accumulators consists of inspections, minor repairs, replacement of parts, and testing. As there is an element of danger in maintaining accumulators, precautions must be strictly observed to prevent injury and damage. Before disassembling any accumulator, ensure that all preload air (or nitrogen) pressure has been discharged. Failure to release the preload could result in serious injury. Before making this check, be certain you know the type of high pressure air valve used. When you know that all air pressure has been removed, you can take the unit apart.

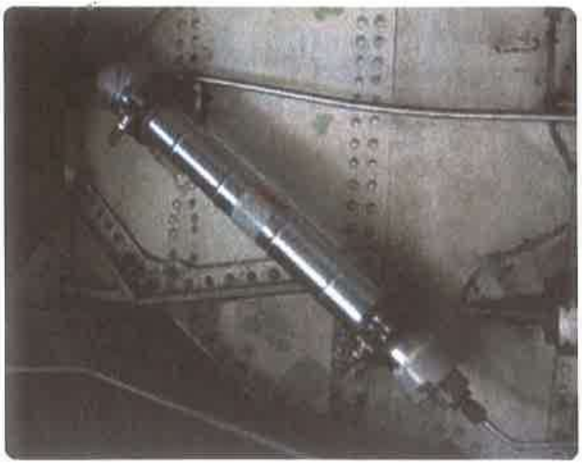


Figure 12-15. Cylindrical accumulator.

HEAT EXCHANGERS

Some larger helicopters use heat exchangers in their hydraulic power supply system to cool the hydraulic fluid from the hydraulic pumps. This extends the service life of the fluid and the pumps. They are typically located in the fuel tanks of the aircraft. The heat exchangers use aluminum finned tubes to transfer heat from the fluid to the fuel. The fuel in the tanks that contain the heat exchangers must be maintained at a specific level to ensure adequate cooling of the fluid. (***Figure 12-16***)

PRESSURE GENERATION: ELECTRIC, MECHANICAL, AND PNEUMATIC

All aircraft hydraulic systems have at least one power driven pump and may include a hand pump in case the power driven pump becomes inoperative. The pump is the source of fluid flow, which when restricted generates pressure in the hydraulic system. A hydraulic pump can be driven mechanically, electrically or with pneumatic air.

MECHANICAL, ELECTRICAL, AND PNEUMATIC DRIVEN PUMPS

Mechanically driven pumps are the primary source of pressure generation on most aircraft. Typically, the pump is mounted on the accessory gearbox of the main engine and is rotated by a shaft. When the engine is operating, the pump supplies ample fluid flow to generate pressure within the hydraulic system.

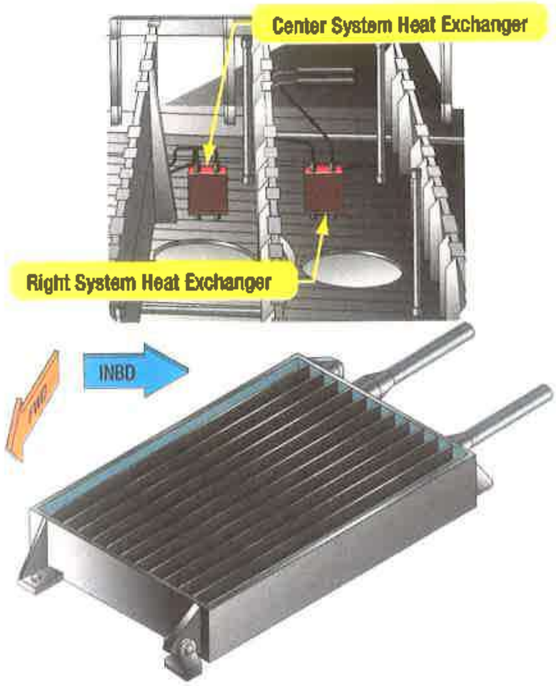


Figure 12-16. Hydraulic heat exchanger.

Electrical motor driven pumps also exist. Often these are the same as the mechanically driven pumps, but the drive shaft is turned by an electrical motor. As such, electrically driven pumps can be mounted away from the engine(s). Electrically driven pumps are installed for use in emergencies or during ground operation when engines are not running.

Pneumatically driven pumps also exist on some aircraft. Typically, these are used as demand pumps to supplement the primary pumps and are driven by air from the pneumatic system.

Modern large helicopters use a combination of engine driven, electrically powered, pneumatic pumps, and power transfer units as fully redundant aircraft hydraulic systems. (***Figure 12-17 and Figure 12-18***)

HAND PUMPS

The hydraulic hand pump is used in some older aircraft for the operation of hydraulic subsystems and in a few newer aircraft systems as a backup unit. Hand pumps are generally installed for testing purposes, as well as for use in emergencies. Hand pumps are also installed to service the reservoirs from a single refilling station. The single refilling station reduces the chances for the introduction of fluid contamination.

Figure 12-17. Engine-driven pump.

Figure 12-18. Electrically driven pump.

Single Action Hand Pumps

A single action hand pump draws fluid into the pump on one stroke and pumps that fluid out on the next stroke. It is rarely used in aircraft due to this inefficiency.

Double Action Hand Pumps

Double action hand pumps produce fluid flow and pressure on each stroke of the handle. (***Figure 12-19***) The double action pump consists essentially of a housing that has a cylinder bore and two ports, a piston, two spring loaded check valves, and an operating handle. An O-ring on the piston seals against leakage between the two chambers of the cylinder bore. An O-ring in a groove in the end of the pump seals against leakage between the piston rod and housing. When the piston is moved to the right, the pressure in the chamber to the left of the piston is lowered. The inlet check valve opens, and hydraulic fluid is drawn into the chamber. At the same time, the rightward movement of the piston forces the ball check valve against its seat. Fluid in the chamber to the right of the piston is forced out and into the hydraulic system. When the piston is moved to the

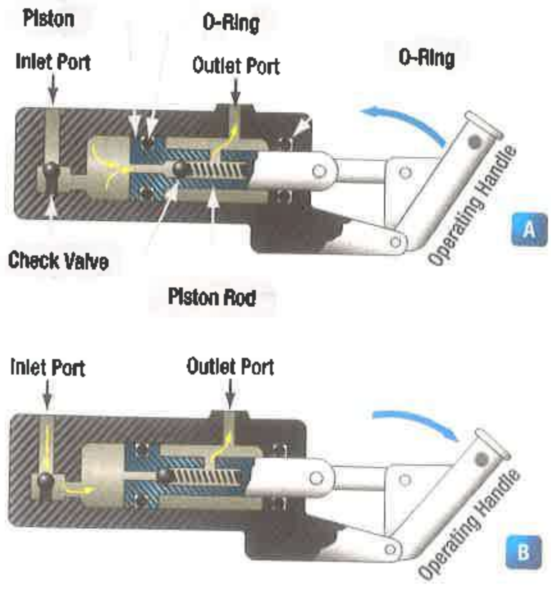


Figure 12-19. Double action hand pump.

left, the inlet port ball check valve seats. Pressure in the chamber left of the piston rises, forcing the piston ball check valve off its seat. Fluid flows from the left chamber through the piston to the right chamber. The volume in the chamber right of the piston is smaller than that of the left chamber due to the displacement created by the piston rod. As the fluid from the left chamber flows into the smaller right chamber, the excess volume of fluid is forced out of the outlet port to the hydraulic system.

Rotary Action Hand Pumps
A rotary hand pump may also be employed. It produces continuous output while the handle is in motion. ***Figure 12-20*** shows a rotary hand pump in a hydraulic system.

CLASSIFICATION OF PUMPS
All pumps may be classified as either positive or non-positive displacement. Most pumps used in hydraulic systems are positive displacement. A non-positive displacement pump produces a continuous flow. However, because it does not provide a positive internal seal against slippage, its output varies considerably as pressure varies. Centrifugal and propeller pumps are examples of non-positive pumps. If the output port of a non-positive pump was blocked off, the pressure would rise, and output would decrease to zero. Although the pumping element would continue moving, flow would stop because of slippage inside the pump.


Figure 12-20. Rotary hand pump.

In a positive displacement pump, slippage is negligible compared to the pump volumetric output. If the output port were plugged, pressure would increase instantaneously to the point that the pump pressure relief valve opens.

Constant Displacement Pumps
A constant displacement pump, regardless of rotations per minute, forces a fixed or unvarying quantity of fluid through the outlet port during each revolution of the pump. Constant displacement pumps are sometimes called constant volume or constant delivery pumps. They deliver a fixed quantity of fluid per revolution, regardless of the pressure demands. Since the constant delivery pump provides a fixed quantity of fluid during each revolution, the quantity of fluid delivered per minute depends upon pump rotations per minute. When a constant displacement pump is used in a hydraulic system in which the pressure must be kept at a constant value, a pressure regulator is required.

Gear Type Power Pump
A gear type power pump is a constant displacement pump. It consists of two meshed gears that revolve in a housing. (***Figure 12-21***) The driving gear is driven by the aircraft engine or some other power unit. The driven gear meshes with and is driven by the driving gear. Clearance between the teeth as they mesh and between the teeth and the housing is very small. The inlet port

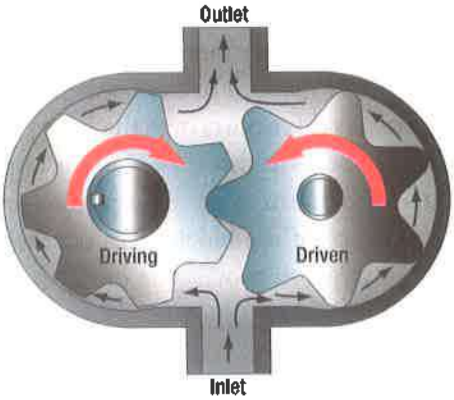


Figure 12-21. Gear-type power pump.

of the pump is connected to the reservoir and the outlet port is connected to the pressure line. When the driving gear turns, it turns the driven gear. Fluid is captured by the teeth as they pass the inlet and travels around the housing and exits at the outlet.

Gerotor Pump

A gerotor power pump consists essentially of a housing containing an eccentric shaped stationary liner, an internal rotor having seven wide teeth of short height, a spur driving gear having six narrow teeth, and a pump cover that contains two crescent shaped openings. (***Figure 12-22***)

One opening extends into an inlet port and the other extends into an outlet port. During the operation of the pump, the gears turn clockwise together. As the pockets between the gears on the left side of the pump move from a lowermost position toward a topmost position, the pockets increase in size, resulting in the production of a partial vacuum within these pockets. Since the pockets enlarge while over the inlet port crescent, fluid is drawn into them. As these same pockets (now full of fluid) rotate over to the right side of the pump, moving from the topmost position toward the lowermost position, they decrease in size. This results in the fluid being expelled from the pockets through the outlet port crescent.

Piston Pumps

Piston pumps can be constant or variable displacement. All have flanged bases for the purpose of mounting the pumps on the accessory drive cases of the engine. A pump drive shaft, which turns the mechanism, extends through the pump housing slightly beyond the mounting base. Torque from the driving unit is transmitted to the pump drive shaft by a drive coupling. The drive coupling is a short shaft with a set of male splines on both ends. The splines on one end engage with female splines in a driving gear. The splines on the other end engage with female splines in the pump drive shaft. (***Figure 12-23***)

Pump drive couplings are designed to serve as safety devices. The shear section of the drive coupling, located midway between the two sets of splines, is smaller in diameter than the splines. If the pump becomes unusually hard to turn or becomes jammed, this section shears, preventing damage to the pump or driving unit. (***Figure 12-24***)

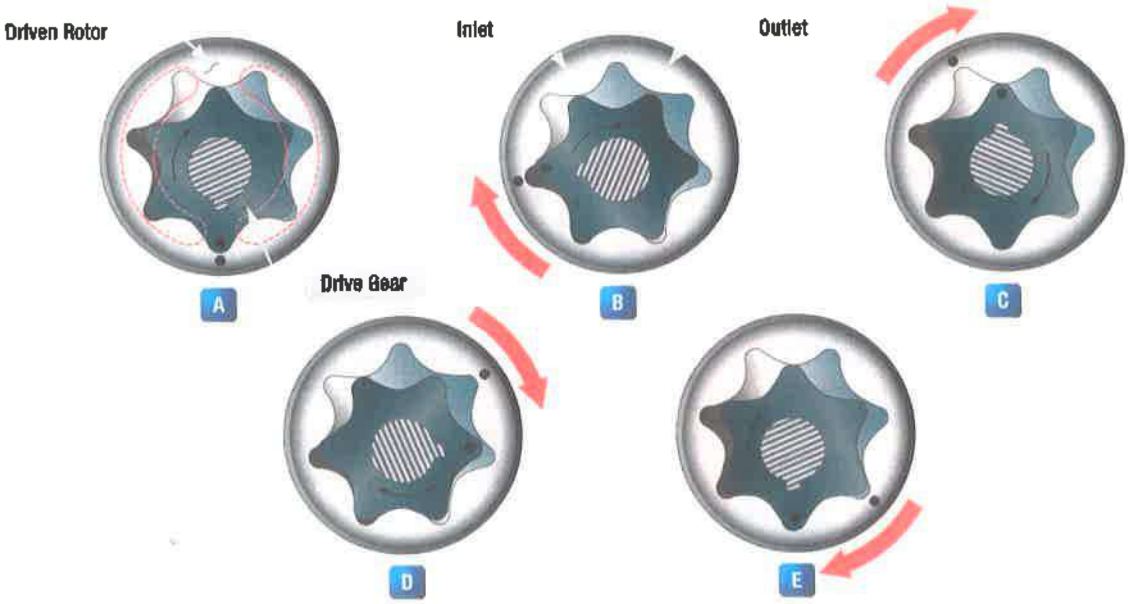


Figure 12-22. Gerotor pump.

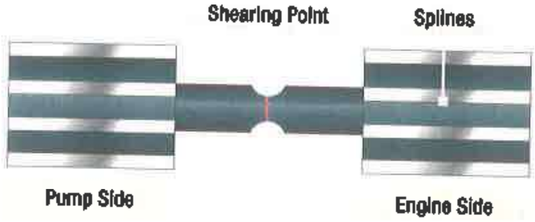


Figure 12-23. Hydraulic pump shear shaft.

The basic pump is a cylinder block, a piston for each bore, and a valve plate with inlet and outlet slots. The valve plate slots let fluid into and out of the bores as the pump operates. The cylinder bores lie parallel to and symmetrically around the pump axis. All aircraft axial piston pumps have an odd number of pistons.

Bent Axis Piston Pump

A typical constant displacement axial type pump is shown in ***Figure 12-25***. Its angular housing causes a corresponding angle to exist between the cylinder block and the drive shaft plate to which the pistons are attached. It is this angular configuration of the pump that causes the pistons to stroke as the pump shaft is turned. When the pump operates, all parts within the pump, except the outer races of the bearings that support the drive shaft, the cylinder bearing pin on which the cylinder block turns, and the oil seal turn together as a rotating group. At one point of rotation, a minimum distance exists between the top of the cylinder block and the upper face of the drive shaft plate. Because of the angled housing at a point of rotation 180° away, the distance between the top of the cylinder block and the upper face of the drive shaft plate is at a maximum. At any given moment of operation, three of the pistons move away from the top face of the cylinder block, producing a partial vacuum in the bores in which these pistons operate. As this occurs over the inlet port, fluid is drawn into these bores at this time. On the opposite side of the cylinder block, three different pistons are moving toward the top face of the block. This occurs while the rotating group is passing over the outlet port causing fluid to be expelled from the pump. The continuous and rapid action of the pistons is overlapping in nature and results in practically no output pulsing.

Inline Piston Pump

The simplest type of axial piston pump is the swash plate design in which a cylinder block is turned by the drive shaft. Pistons fitted to bores in the cylinder block are connected through piston shoes and a retracting ring so that the shoes bear against an angled swash plate. As the block turns, the piston shoes follow the swash plate causing the pistons to reciprocate. The ports are arranged in the valve plate so that the pistons pass the inlet as they are pulled out and pass the outlet as they are forced back in. In these pumps, displacement is determined by the size and number of pistons, as well as their stroke length, which varies with the swash plate angle.

Vane Pump

The vane type power pump is also a constant displacement pump. It consists of a housing containing four vanes (blades), a hollow steel rotor with slots for the vanes, and a coupling to turn the rotor. (***Figure 12-26***)

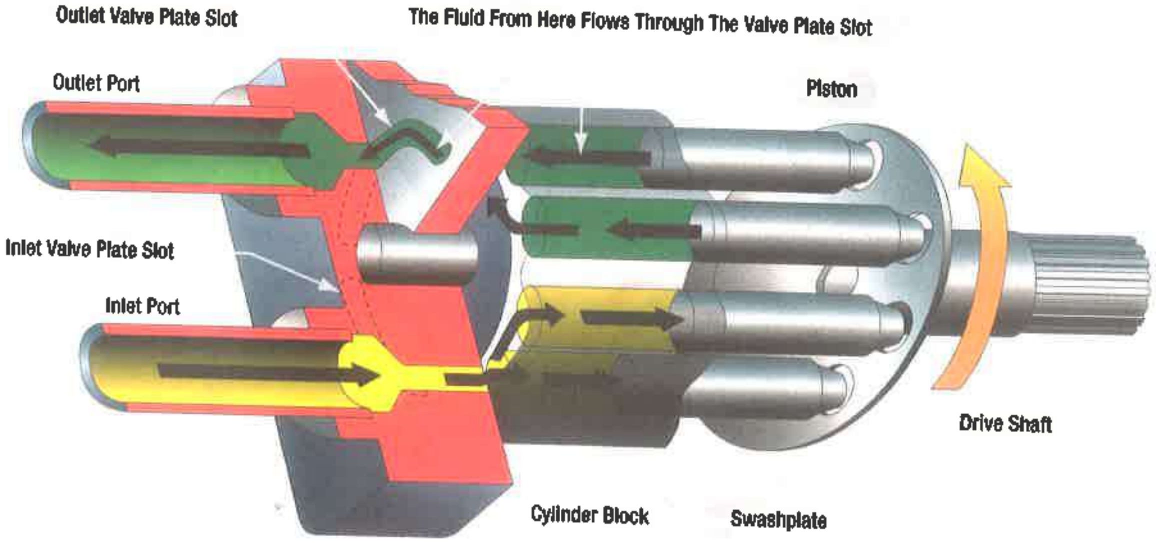


Figure 12-24. Axial inline piston pump.

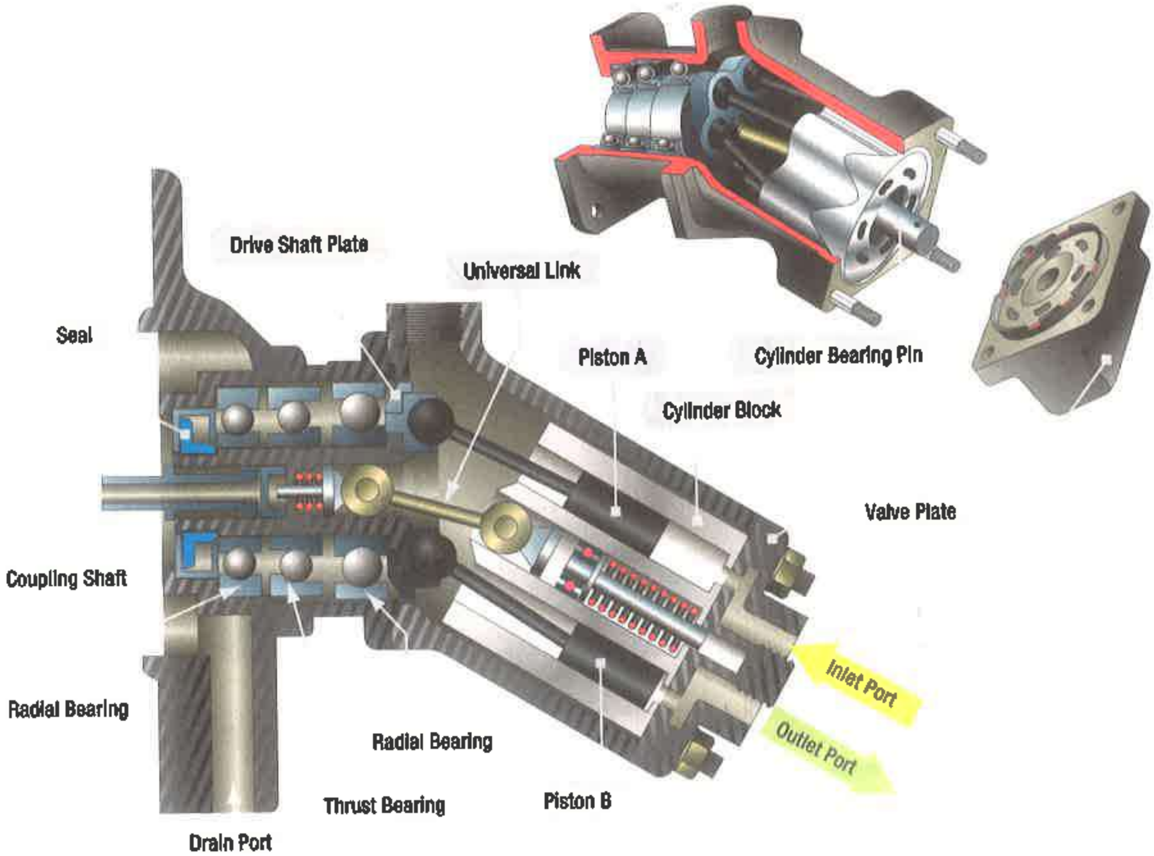


Figure 12-25. Bent axis piston pump.

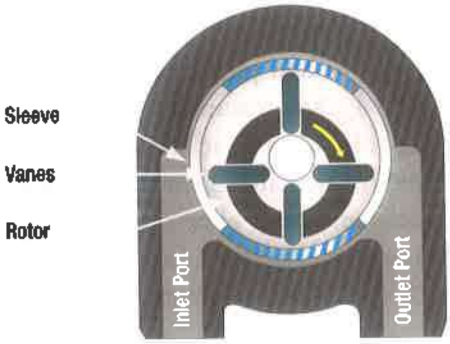


Figure 12-26. Vane-type power pump.

The rotor is positioned off center within the sleeve. The vanes, which are mounted in the slots in the rotor together with the rotor, divide the bore of the sleeve into four sections. As the rotor turns, each section passes one point where its volume is at a minimum and another point where its volume is at a maximum. The volume gradually increases from minimum to maximum during the first half of a revolution and gradually decreases from maximum to minimum during the second half of the revolution. As the volume of a given section increases, that section is connected to the pump inlet port through a slot in the sleeve. Since a partial vacuum is produced by the increase in volume of the section, fluid is drawn into the section through the pump inlet port and the slot in the sleeve. As the rotor turns through the second half of the revolution and the volume of the given section is decreasing, fluid is displaced out of the section through the slot in the sleeve aligned with the outlet port, and out of the pump.

Variable Displacement Pumps

A variable displacement pump has a fluid output that is varied to meet the pressure demands of the system. The pump output is changed automatically by a compensator within the pump. The following paragraph discusses a two stage Vickers variable displacement pump. (***Figure 12-27***)

BASIC PUMPING OPERATION

The aircraft engine rotates the pump drive shaft, cylinder block, and pistons via a gearbox. Pumping action is generated by piston shoes that are restrained and slide on the shoe bearing plate in the yoke assembly. Because the yoke is at an angle to the drive shaft, the rotary motion of the shaft is converted to piston reciprocating motion. As the piston withdraws from the cylinder block, inlet pressure forces fluid through a porting in the valve plate

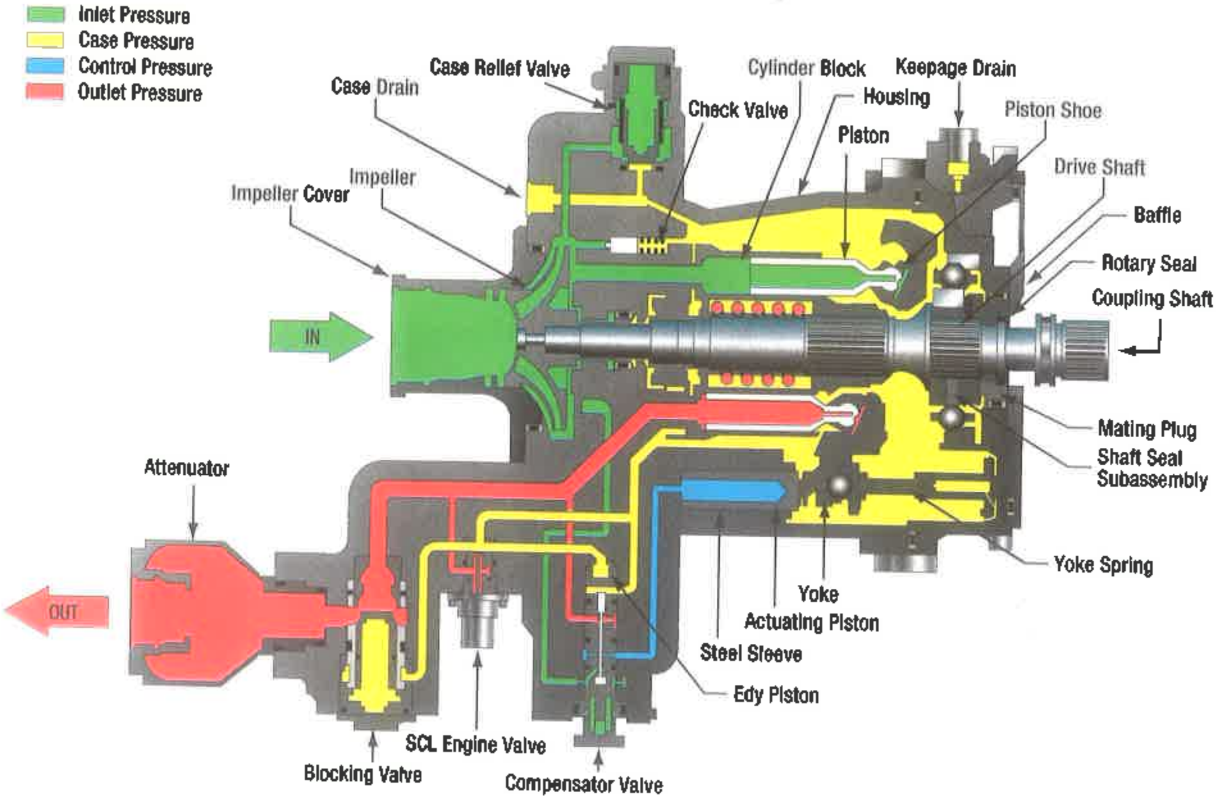


Figure 12-27. Variable displacement pump.

into the cylinder bore. The piston shoes are restrained in the yoke by a piston shoe retaining plate and a shoe plate during the intake stroke. As the drive shaft continues to turn the cylinder block, the piston shoe continues following the yoke bearing surface. This begins to return the piston into its bore (toward the valve block). The fluid contained in the bore is pre-compressed, then expelled through the outlet port. Discharge pressure holds the piston shoe against the yoke bearing surface during the discharge stroke and provides the shoe pressure through an orifice in the piston and shoe subassembly. With each revolution of the drive shaft and cylinder block, each piston goes through the pumping cycle described above, completing one intake and one discharge stroke. High pressure fluid is ported out through the valve plate, past the blocking valve, to the pump outlet.

The blocking valve is designed to remain open during normal pump operation. Internal leakage keeps the pump housing filled with fluid for lubrication of rotating parts and cooling. The leakage is returned to the system through a case drain port. The case relief valve protects the pump against excessive case pressure, relieving it to the pump inlet.

Normal Pumping Mode

A pressure compensator is a spool valve that is held in the closed position by an adjustable spring load. (*Figure 12-28*) When pump outlet pressure (system pressure) exceeds the pressure setting (2 850 psi for full flow), the spool moves to admit fluid from the pump outlet against the actuator piston. In *Figure 12-28*, the pressure compensator is shown at cracking pressure, meaning the outlet pressure is just high enough to move the spool to begin admitting fluid to the piston.

The yoke is supported inside the pump housing on two bearings. At pump outlet pressures below 2 850 psi, the yoke is held at its maximum angle relative to the drive shaft centerline by the force of the yoke return spring. Decreasing flow demand causes outlet pressure to become high enough to open the compensator valve and admit fluid to the actuator piston. This control pressure overcomes the yoke return spring force and strokes the pump yoke to a reduced angle. The reduced angle of the yoke results in a shorter stroke for the pistons and reduced displacement. (*Figure 12-29*)

The lower displacement results in a corresponding reduction in pump flow. The pump delivers only that flow required to maintain the desired pressure in the

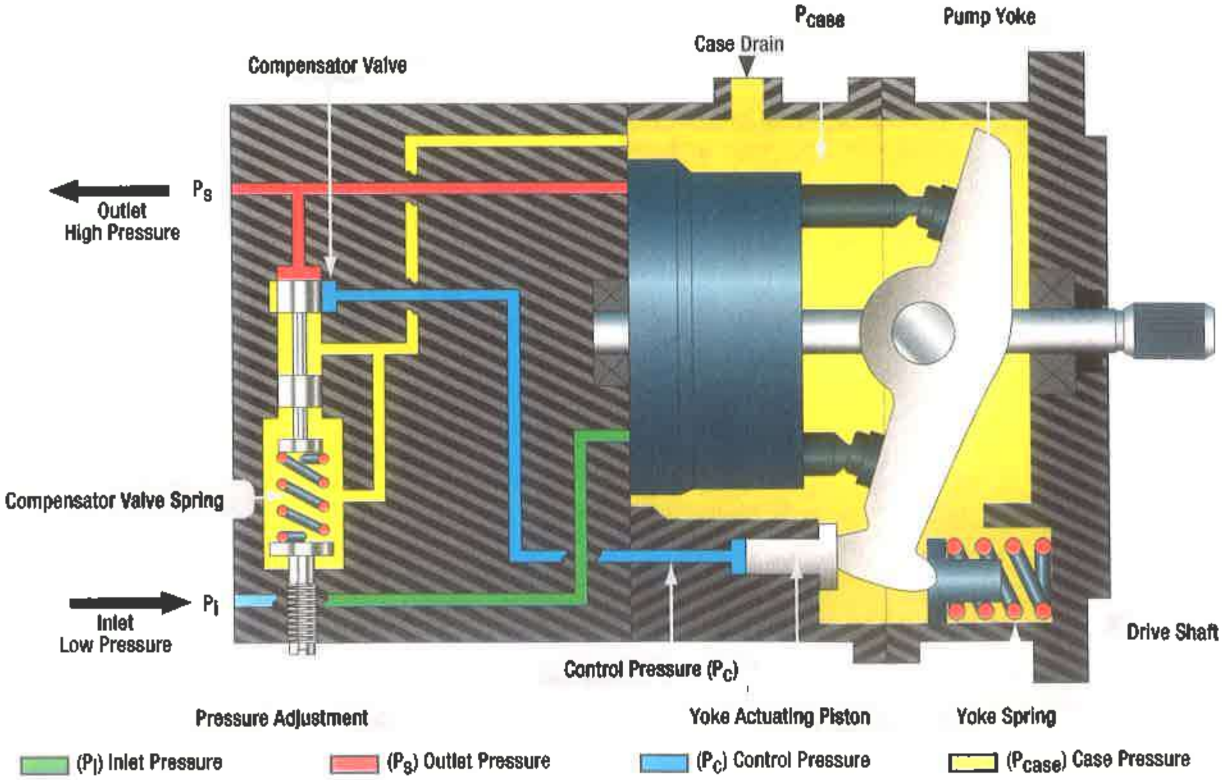


Figure 12-28. Normal pumping mode.

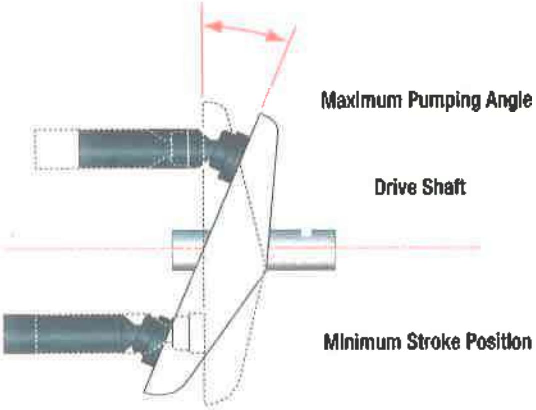


Figure 12-29. Yoke angle.

system. When there is no demand for flow from the system, the yoke angle decreases to nearly zero degrees of stroke. In this mode, the unit pumps only its own internal leakage. Thus, at pump outlet pressures above 2 850 psi, displacement decreases as outlet pressure rises. At system pressures below this level, no fluid is admitted through the pressure compensator valve to the actuator piston and the pump remains at full displacement, delivering full flow. Pressure is then determined by the system demand. The unit maintains zero flow at a system pressure of 3 025 psi.

Depressurized Mode

When the solenoid valve is energized, the solenoid valve moves up against the spring force and the outlet fluid is ported to the control piston on the top of the compensator (depressurizing piston). (***Figure 12-30***)

The high pressure fluid pushes the compensator spool beyond its normal metering position. This removes the compensator valve from the circuit and connects the actuator piston directly to the pump outlet. Outlet fluid is also ported to the blocking valve spring chamber, which equalizes pressure on both sides of its plunger.

The blocking valve closes due to the force of the blocking valve spring and isolates the pump from the external hydraulic system. The pump strokes itself to zero delivery at an outlet pressure that is equal to the pressure required on the actuator piston to reduce the yoke angle to nearly zero, approximately 1 100 psi. This depressurization and blocking feature can be used to reduce the load on the engine during startup and, in a multiple pump system, to isolate one pump at a time and check for proper system pressure output.

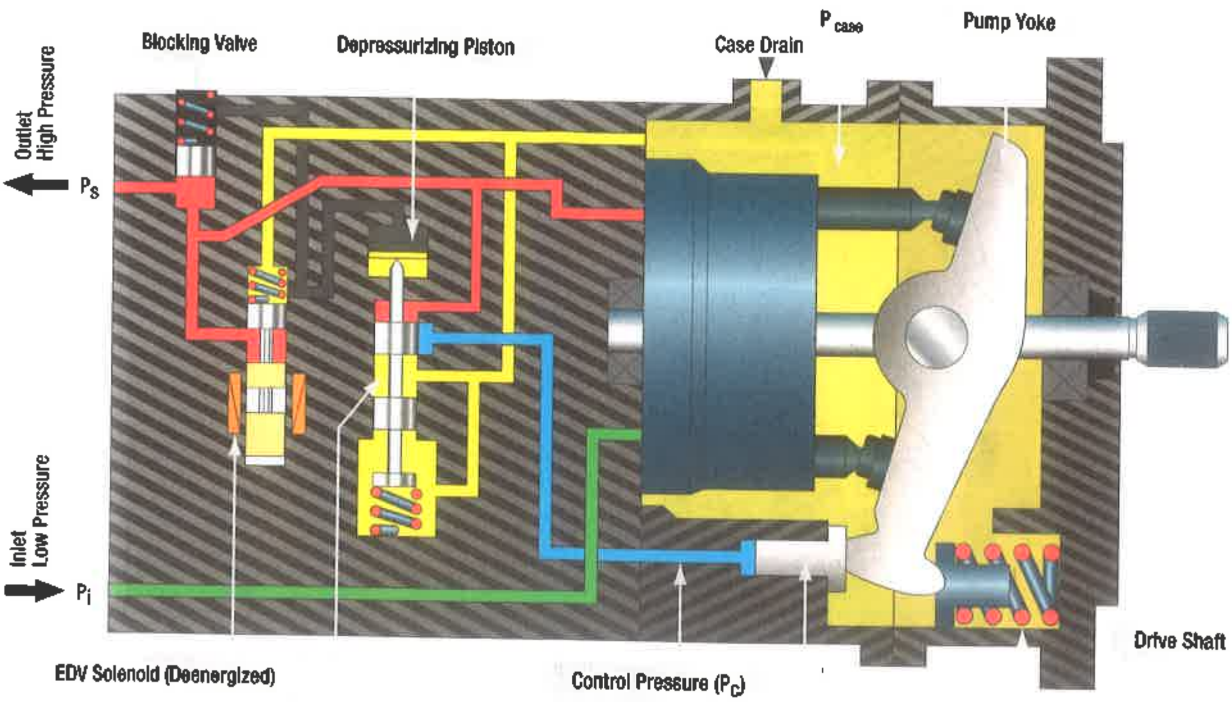


Figure 12-30. Depressurized mode.

HYDRAULIC SEALS

Seals are used to prevent fluid from passing a certain point, and to keep air and dirt out of the system in which they are used. The increased use of hydraulics and pneumatics in aircraft systems has created a need for packing and gaskets of varying characteristics and designed to meet the many variations of operating speeds and temperatures to which they are subjected.

No one type of seal is satisfactory for all installations. Some of the reasons for this are:

- Pressure at which the system operates.
- The type of fluid used in the system.
- The metal finish and the clearance between adjacent parts.
- The type of motion (rotary or reciprocating), if any.

Most seals are made from synthetic materials that are compatible with the hydraulic fluid used. Seals used for MIL-H-5606 hydraulic fluid are not compatible with Skydrol® and servicing the hydraulic system with the wrong fluid could result in leaks and system malfunctions. Seals for systems that use MIL-H-5606 are made of neoprene or Buna-N. Seals for Skydrol® are made from butyl rubber or ethylene-propylene elastomers.

Seals are divided into three main classes: packings, gaskets, and wipers. A seal may consist of more than one component, such as an O-ring and a backup ring, or possibly an O-ring and two backup rings. Hydraulic seals used internally on a sliding or moving assembly are normally called packings. (***Figure 12-31***) Hydraulic seals used between non-moving fittings and bosses are normally called gaskets.

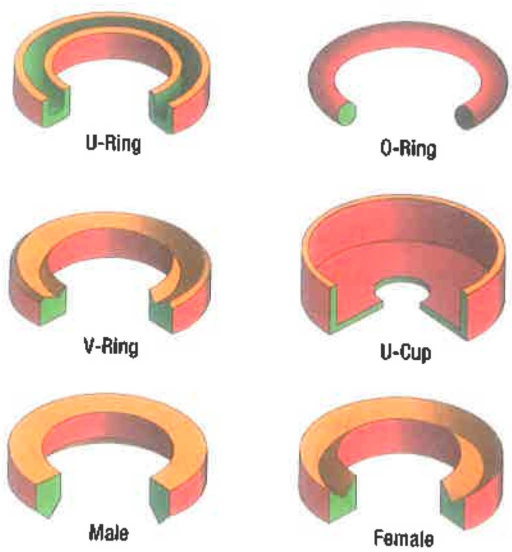


Figure 12-31. Packings.

PACKINGS

V-Rings

V-ring packings (AN6225) are one way seals and are always installed with the open end of the V facing the pressure. Each must have a male and female adapter to hold it in the proper position after installation. It is also necessary to torque the seal retainer to the value specified by the manufacturer of the component being serviced or the seal may not give satisfactory service.

U-Ring

U-ring packings (AN6226) and U-cup packings are used in brake assemblies and brake master cylinders. The U-ring and U-cup seals pressure in only one direction. Therefore, the lip of the packing must face toward the pressure. U-ring packings are primarily low pressure packings to be used with pressures less than 1 000 psi.

O-Rings

Most packings and gaskets used in aircraft are manufactured in the form of O-rings. An O-ring is circular in shape, and its cross section is small in relation to its diameter. The cross section is round and has been molded and trimmed to extremely close tolerances. O-rings seal effectively in both directions. This sealing is done by distortion of its elastic compound. Advances in aircraft design have made new O-ring composition necessary to meet changing conditions.

O-Ring manufacturers provide color coding on some O-rings, but this is not a reliable or complete means of identification. The color coding system does not identify sizes, but only system fluid or vapor compatibility. Color codes on O-rings that are compatible with MIL-H-5606 fluid always contain blue but may also contain red or other colors. Packings and gaskets suitable for use with Skydrol® fluid are always coded with a green stripe, but may also have a blue, gray, red, green, or yellow dot as a part of the color code. Color codes on O-rings that are compatible with hydrocarbon fluid always contain red, but never contain blue. A colored stripe around the circumference indicates that the O-ring is a boss gasket seal. The color of the stripe indicates fluid compatibility: red for fuel, blue for hydraulic fluid. The coding on some rings is not permanent. On others, it may be omitted due to manufacturing difficulties or interference with operation. Furthermore, the color coding system provides no means to establish the age of the O-ring or its temperature limitations. Because of the difficulties with color coding, O-rings are available in individual hermetically sealed envelopes labeled with all pertinent data. When selecting an O-ring for installation, the basic part number on the sealed envelope provides the most reliable compound identification.

Backup O-rings (MS28782) made of Teflon do not deteriorate with age, are unaffected by system fluid or vapor, and can tolerate temperature extremes exceeding those encountered in high pressure hydraulic systems. Their dash numbers indicate not only their size but also relate directly to the dash number of the O-ring for which they are dimensionally suited. They are procurable under a number of basic part numbers and are interchangeable. Any Teflon backup ring may be used to replace any other Teflon backup ring if it is of proper overall dimension to support the applicable O-ring. Backup rings are not color coded or otherwise marked and must be identified from package labels. The inspection of backup rings should include a check to ensure that surfaces are free from irregularities, that the edges are clean cut and sharp, and that scarf cuts are parallel. When checking Teflon spiral backup rings, make sure that the coils do not separate more than 0.6 cm when unrestrained. Be certain that backup rings are installed downstream of the O-ring. (***Figure 12-32***)

When removing or installing O-rings, avoid using pointed or sharp edged tools that might cause scratching or marring of hydraulic component surfaces or cause damage to the O-rings. Special tools for the installation of O-rings are available. While using the removal and the installation tools, contact with cylinder walls, piston heads, and related precision components is not desirable. After the removal of all O-rings, the parts that receive new O-rings have to be cleaned and inspected to make sure that they are free from contamination. Each replacement O-ring should be removed from its sealed package and inspected for defects such as blemishes, abrasions, cuts, or punctures. Although an O-ring may appear perfect at first glance, slight surface flaws may exist. These flaws are often capable of preventing satisfactory performance under the variable operating pressures of aircraft systems. Such flaws are difficult to detect. One aircraft manufacturer recommends using a 4x power magnifying glass with adequate lighting to inspect each ring before it is installed. By rolling the ring on an inspection cone or dowel, the inner diameter

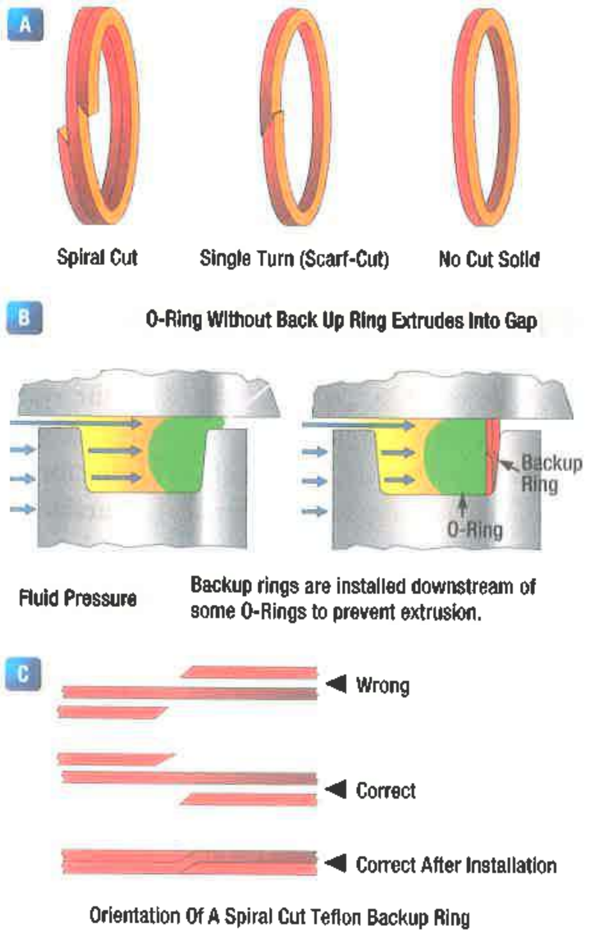


Figure 12-32. Backup O-rings installed downstream.

surface can also be checked for small cracks, particles of foreign material, or other irregularities that cause leakage or shorten the life of the O-ring. The slight stretching of the ring when it is rolled inside out helps to reveal some defects not otherwise visible.

After inspection and prior to installation, immerse the O-ring in clean hydraulic fluid. During the installation, avoid rolling and twisting the O-ring to maneuver it into place. If possible, keep the position of the O-ring mold line constant. When the O-ring installation requires spanning or inserting through sharply threaded areas, ridges, slots, or edges, use protective measures such as O-ring entering sleeves, as shown in ***Figure 12-33A***. After the O-ring is placed in the cavity provided, gently roll the O-ring with your fingers to remove any twist that might have occurred during installation. (***Figure 12-34***)

Gaskets

Gaskets are used as static (stationary) seals between two flat surfaces. Some of the more common gasket materials are asbestos, copper, cork, and rubber.

Asbestos sheeting is used wherever heat resistance is needed. It is used extensively for exhaust system gaskets. Most asbestos exhaust gaskets have a thin sheet of copper edging to prolong their life. Asbestos based gaskets were used until public awareness of asbestos related diseases motivated manufacturers to find substitutes for the toxic mineral. Most gasket manufacturers now use heat resistant substitutes for asbestos.

A solid copper washer can be used where it is essential to have a non-compressible, yet semi-soft gasket.

Cork gaskets can be used as an oil seal between the engine crankcase and accessories, and where a gasket is required that can occupy an uneven or varying space caused by a rough surface or expansion and contraction.

Rubber sheeting can be used where there is a need for a compressible gasket. It should not be used in any place where it may encounter gasoline or oil because the rubber deteriorates very rapidly when exposed to these substances.

Gaskets are used in fluid systems around the end caps of actuating cylinders, valves, and other units. The gasket generally used for this purpose is in the shape of an O-ring, similar to O-ring packings.

Wipers

Wipers are used to clean and lubricate the exposed portions of piston shafts. They prevent dirt from entering the system and help protect the piston shaft against scoring. Wipers may be either metallic or felt. They are sometimes used together with a felt wiper installed behind a metallic wiper.

EMERGENCY PRESSURE GENERATION

Generation of hydraulic pressure in emergency situations varies. Large aircraft with multiple hydraulic systems are designed to ensure hydraulic pressure to critical components even in the event of a complete system failure or loss of engines. Generally, electrically driven pumps are used when engine driven pumps fail. Should there be no working engines, not only would the engine driven pump be inoperative but electrical generator output would cease as well. This may leave only the aircraft batteries to provide electrical power to the electrically driven pumps.

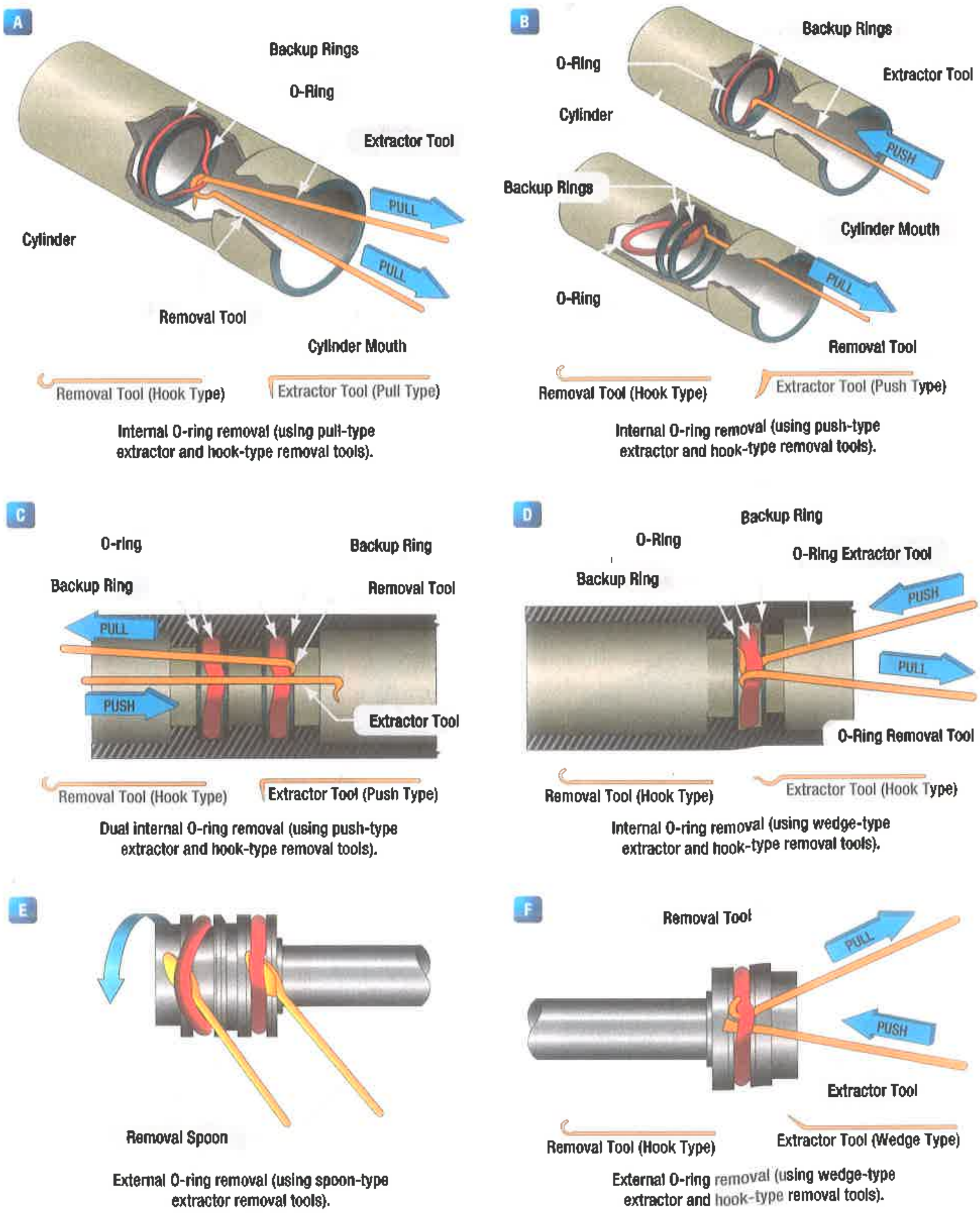


Figure 12-33. O-ring installation techniques.

HYDRAULIC MOTORS

Just as a rotating shaft drives a hydraulic pump to move fluid, fluid forced through the pump can rotate the attached shaft. This is the principle behind a hydraulic motor. Hydraulic fluid forced through the pump rotates the shaft of the pump, which as a result makes the pump a motor. The motion of the shaft is then used to drive something to which it is attached. Piston type motors are the most used in hydraulic systems. (***Figure 12-35***) They are basically the same as hydraulic pumps except they are used to convert hydraulic energy into mechanical energy.

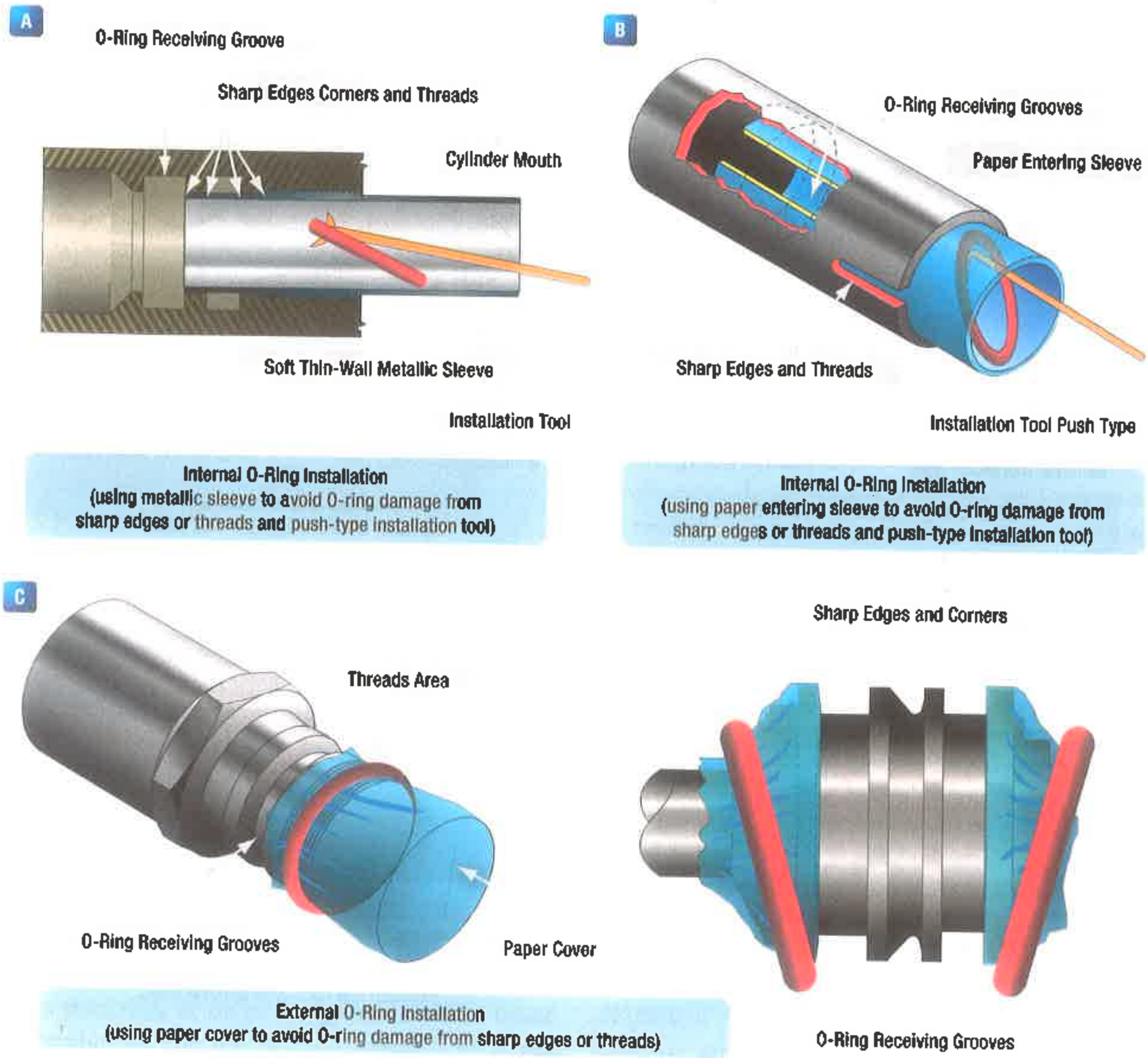


Figure 12-34. More O-ring installation techniques.

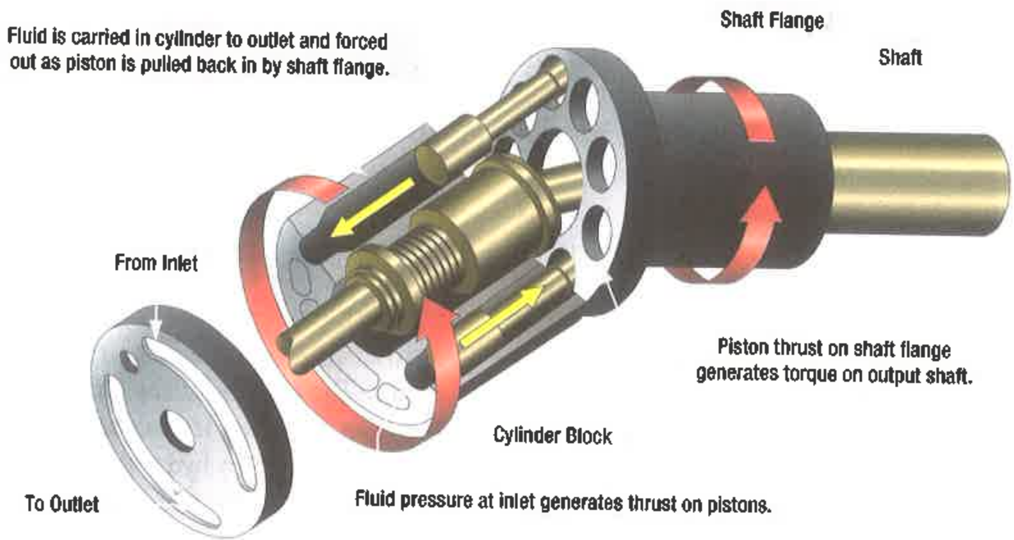


Figure 12-35. Bent axis piston motor.

The most used hydraulic motor is the fixed displacement bent axis type. This type of motor is used for the activation of trailing edge flaps, leading edge slats, and stabilizer trim. Some equipment uses a variable displacement piston motor where very wide speed ranges are desired. Although some piston type hydraulic motors are controlled by directional control valves, they are often used in combination with variable displacement pumps.

This pump/motor combination is used to provide a transfer of power between a driving element and a driven element. Some applications for which hydraulic transmissions may be used are speed reducers, variable speed drives, constant speed or constant torque drives, and torque converters. Some advantages of hydraulic transmission of power over mechanical transmission of power are:

- Quick, easy speed adjustment over a wide range while the power source is operating at a constant (most efficient) speed.
- Rapid, smooth acceleration or deceleration.
- Control over maximum torque and power.
- Cushioning effect to reduce shock loads.
- Smoother reversal of motion.

POWER TRANSFER UNITS (PTUs)

Hydraulic motors are also used in Power Transfer Units (PTUs). In a PTU, two units, a hydraulic pump and hydraulic motor are connected via a single drive shaft so that power can be transferred between two hydraulic systems. Depending on the direction of transfer, each unit works as either a motor or a pump. The pressurized hydraulic system forces fluid through the motor which turns the shaft of the pump that moves fluid through the second hydraulic system. Thus, power is transferred from one system to the other. While the PTU transfers power, it does not transfer any fluid from one system to the other. (*Figure 12-36*)

Different types of PTUs are in use. Some can only transfer power in one direction while others can transfer power both ways. Some PTUs have a fixed displacement, while others use a variable displacement pump. Regardless, the application of PTUs in aircraft allow component operation in a hydraulic system in which the pump has failed. The system with a working pump drives the motor of the PTU so that the pump shaft rotates in the system with the failed pump. Activation can be manual or automatic depending on the aircraft. In an automatically activated system, a pressure switch is used to detect pump failure which opens a valve to divert fluid from the healthy system to the PTU.

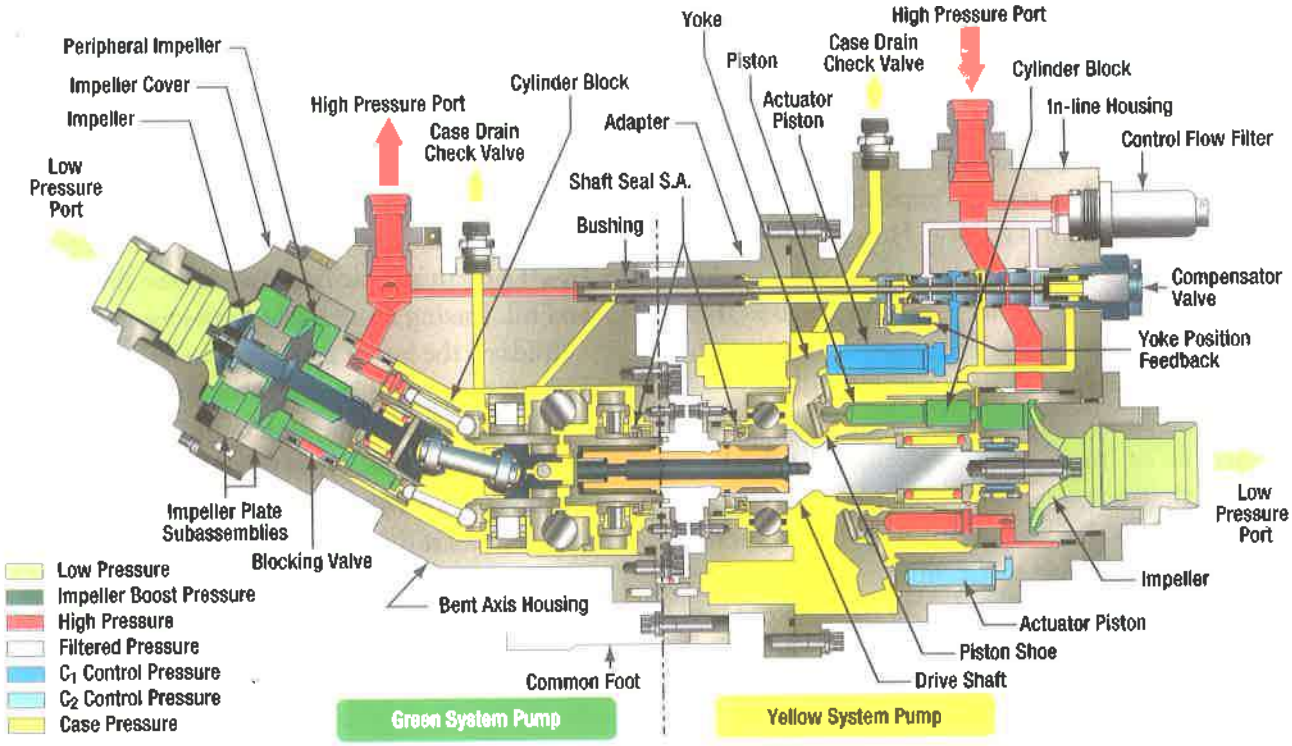


Figure 12-36. Power transfer unit.

HYDRAULIC MOTOR DRIVEN GENERATORS (HMDG)

In case of an electrical failure, a hydraulic motor driven generator can be employed. An HMDG provides an alternative source of electrical power. The servo controlled variable displacement HMDG is an AC generator driven by the hydraulic motor portion of the unit. The generator part is typically designed to maintain the desired system output frequency of 400 Hz. Thus, an aircraft with an HMDG can maintain electrical power should a primary generator fail.

SYSTEM CONTAMINATION AND FILTERS

HYDRAULIC FLUID CONTAMINATION

Experience has shown that trouble in a hydraulic system is inevitable whenever the fluid is allowed to become contaminated. The nature of the trouble, whether a simple malfunction or the complete destruction of a component, depends to some extent on the type of contaminant. Two general contaminants are:

- Abrasives, including such particles as core sand, weld spatter, machining chips, and rust.
- Non-abrasives, including those resulting from oil oxidation and soft particles worn or shredded from seals and other organic components.

Contamination Check

Whenever it is suspected that a hydraulic system has become contaminated or has been operated at temperatures exceeding the maximum specified, a check of the system should be made. The filters in hydraulic systems are designed to remove most foreign particles that are visible to the naked eye. However hydraulic fluid that appears clean to the naked eye may still be contaminated to the point that it is unfit for use. Thus, visual inspection of the fluid does not determine the total amount of contamination in the system.

Large particles of impurities in the hydraulic fluid are indications that one or more components are being subjected to excessive wear. Isolating the defective component requires a systematic process of elimination. To determine which component is defective, fluid samples should be taken from the reservoir and from various other locations in the system. Samples should be taken in accordance with the applicable manufacturer's instructions for each sub-system. Some hydraulic systems are equipped with permanently installed bleed valves for taking samples, whereas on others, lines must be disconnected to provide a sample.

Hydraulic Sampling Schedule

- Routine sampling - each system should be sampled at least once a year, or every 3 000 flight hours, or when the airframe manufacturer suggests.
- Unscheduled maintenance - when malfunctions may have a fluid related cause, samples should be taken.
- Suspicion of contamination - if contamination is suspected, fluids should be drained and replaced.

Sampling Procedure Example:

- Pressurize and operate the hydraulic system for 10-15 minutes. During this period, operate various flight controls to activate valves and thoroughly mix the hydraulic fluid.
- Shut down and depressurize the system.
- Before taking samples, always be sure to wear the proper protective equipment including at the minimum safety glasses and gloves.
- Wipe off the sampling port or tube with a lint free cloth. Do not use shop towels or paper products that could produce lint. Generally speaking, the human eye can see particles down to about 40 microns in size. Since we are concerned with particles down to five microns, it is easy to contaminate a sample without knowing it.
- Place a waste container under the reservoir drain valve and open the valve so that a steady, but not forceful, stream is running.
- Allow approximately one pint (250 ml) of fluid to drain. This purges any settled particles from the sampling port.
- Insert a clean sample bottle under the fluid stream and fill, leaving an air space at the top.
- Withdraw the bottle and close the drain valve.
- Fill out the sample identification label supplied in the sample kit, making sure to include customer name, aircraft type, aircraft tail number, hydraulic system sampled, and date sampled. Indicate on the sample label if this is a routine sample or if it is being taken due to a suspected problem.
- Service system reservoirs to replace the fluid that was removed.
- Submit samples for analysis to the laboratory.

Contamination Control

Filters provide adequate control of contamination during all normal hydraulic system operations. Control of the size and amount of contamination entering the system from any other source is the responsibility of the people who service and maintain the equipment. Therefore, precautions should be taken to minimize contamination during maintenance, repair, and service operations. If the system becomes contaminated, the filter element should be removed and cleaned or replaced.

As an aid in controlling contamination, the following maintenance and servicing procedures should always be followed:

- Maintain all tools and the work area in a clean, dirt free condition.
- A suitable container should always be provided to receive the hydraulic fluid that is spilled during component removal or disassembly procedures.
- Before disconnecting hydraulic lines or fittings, clean the affected area with a dry-cleaning solvent.
- All hydraulic lines and fittings should be capped or plugged immediately after disconnecting.
- Before assembly of any hydraulic components, wash all parts in an approved dry-cleaning solvent.
- After cleaning the parts in the dry-cleaning solution, dry the parts thoroughly and lubricate them with the recommended preservative or hydraulic fluid before assembly. Use only clean, lint free cloths to wipe or dry the component parts.
- All seals and gaskets should be replaced during the reassembly procedure, using only those recommended by the manufacturer.
- Parts should be cautiously connected to avoid stripping metal slivers from threaded areas. All fittings and lines should be installed and torqued in accordance with applicable instructions.

HYDRAULIC SYSTEM FLUSHING

When inspection of hydraulic filters or hydraulic fluid indicates that the fluid is contaminated, flushing the system may be necessary. This must be done according to the manufacturer's instructions; however, a typical procedure for flushing is as follows:

- Connect a hydraulic test stand to the inlet and outlet test ports of the system. Verify that the ground unit fluid is clean and contains the same fluid as the aircraft.
- Change the system filters.
- Pump clean, filtered fluid through the system, and operate all subsystems until no signs of contamination are found during inspection of the filters. Dispose of contaminated fluid and filter.
- Disconnect the test stand and cap the ports.
- Ensure that the reservoir is filled to the full line or service level. It is very important to check if the fluid in the hydraulic test stand is clean before the flushing operation starts.

FILTERS

A filter is a straining device used to clean the hydraulic fluid, preventing foreign particles and contaminating substances from remaining in the system. (***Figure 12-37***) If such objectionable material were not removed the entire hydraulic system of the aircraft could fail through the breakdown or malfunctioning of a single unit of the system.

The hydraulic fluid holds in suspension tiny particles of metal that are deposited during the normal wear of selector valves, pumps, and other components. Such minute particles of metal may damage the units through which they pass if they are not removed by a filter. Since tolerances within the hydraulic system components are quite small, it is apparent that the reliability and efficiency of the entire system depends upon adequate filtering. Filters may be located within the reservoir, in the pressure line, in the return line, or in any other location the designer of the system decides that they are needed to safeguard against impurities. Modern designs often use a filter module that contains several filters and other components. (***Figure 12-38***)

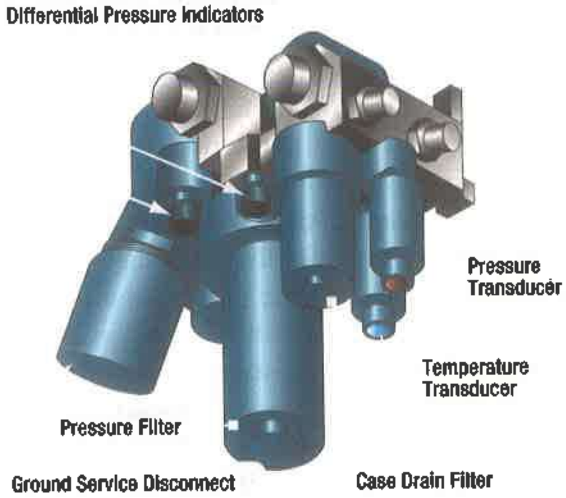


Figure 12-37. Filter module components.

There are many styles of filters. Their position in the aircraft and their design requirements determine their shape and size. Most filters in modern aircraft are the inline type. Inline filter assemblies are comprised of the head assembly, bowl, and element. The head assembly is secured to the aircraft structure and connecting lines. The bowl is the housing that holds the element to the filter head and is removed when element removal is required. The element may be a micron, porous metal, or magnetic type. The micron element is normally thrown away when removed. The porous metal and magnetic filter elements are designed to be cleaned by various methods and replaced in the system.

In the absence of specific replacement instructions, a recommended service life of filter elements is:

Pressure Filters - 3 000 hours
Return Filters - 1 500 hours
Case Drain Filters - 600 hours

MICRON FILTERS

A typical micron type filter assembly utilizes an element made of specially treated paper that is formed in vertical convolutions (wrinkles). An internal spring holds the elements in shape. The micron element is designed to prevent the passage of solids greater than 10 microns (0.000 394 inch) in size. (*Figure 12-39*)

In the event a filter element becomes clogged, the spring-loaded relief valve in the filter head bypasses the fluid after a differential pressure of 50 psi has been built up. Hydraulic fluid enters the filter through the inlet port in the filter body and flows around the element inside the bowl. Filtering takes place as the fluid passes through the element into the hollow core, leaving the foreign material on the outside of the element.

MAINTENANCE OF FILTERS

Maintenance of filters is relatively easy. It mainly involves cleaning the filter assembly and cleaning or replacing the element. Micron filters should have the element replaced periodically according to applicable instructions. Since reservoir filters are of the micron type, they must also be periodically changed or cleaned. For filters using other than micron elements, cleaning is usually all that is necessary. However, it should be inspected very closely to ensure it is undamaged. The methods and materials used in cleaning all filters are too numerous to be included in this text. Consult the manufacturer's instructions for this information. When replacing filter elements, be sure that there is no pressure on the filter bowl. Protective clothing and a face shield must be used to prevent fluid from contacting the eyes. Replace the element with one of the proper rating. After the element has been replaced, the system must be pressure tested to ensure that the sealing element in the filter is intact. In the event of a major component failure, such as a pump, consideration must be given to replacing the entire filter as well as the failed component.

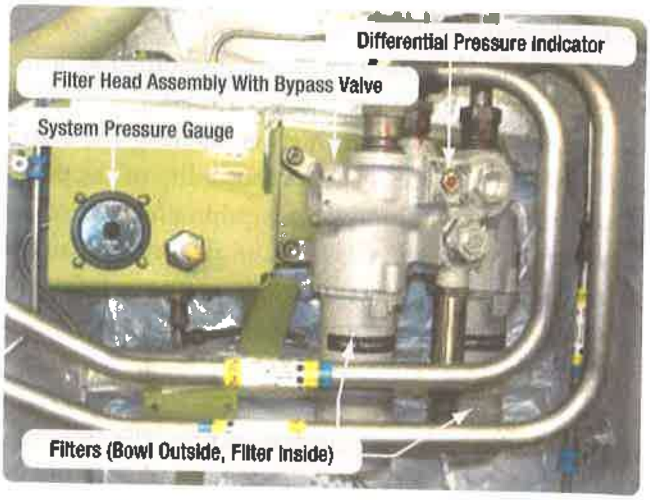


Figure 12-38. A transport category filter module.

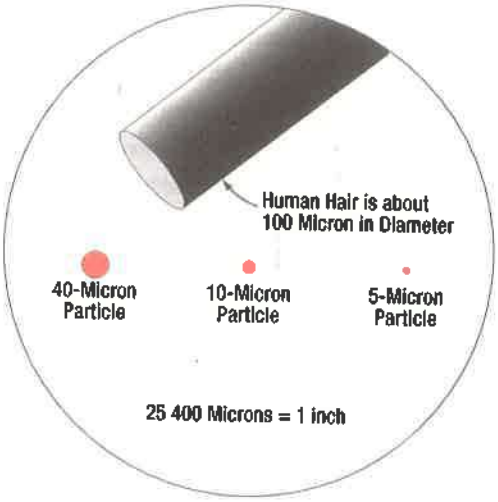


Figure 12-39. Size comparison in microns.

FILTER DIFFERENTIAL PRESSURE INDICATORS

The extent to which a filter element is loaded can be determined by measuring the drop in hydraulic pressure across the element under rated flow conditions. This drop, or differential pressure, provides a convenient means of monitoring the condition of installed filter elements and is the operating principle used in the differential pressure or loaded filter indicators found on many filter assemblies.

Differential pressure indicating devices have many configurations, including electrical switches, continuous reading visual gauges, and visual indicators with memory. Memory indicators usually take the form of magnetic or mechanically latched buttons or pins that extend when the differential pressure exceeds that allowed for a serviceable element. (***Figure 12-40***)

When this increased pressure reaches a specific value, inlet pressure forces the spring loaded magnetic piston downward, breaking the magnetic attachment between the indicator button and the magnetic piston. This allows the red indicator to pop out, signifying that the element must be cleaned. The button or pin, once extended, remains in that position until manually reset and provides a permanent (until reset) warning of a loaded element. This feature is particularly useful where it is impossible for an operator to continuously monitor the indicator, such as in a remote location on the aircraft. Some button indicators have a thermal lockout device incorporated in their design that prevents operation of the indicator below a certain temperature. The lockout prevents the higher differential pressure generated at cold temperatures by high fluid viscosity from causing a false indication of a loaded filter element.

Differential pressure indicators are a component part of the filter assembly and are normally tested and overhauled as part of the complete assembly. With some model filter assemblies, it is possible to replace the indicator itself without removal of the filter assembly if it is suspected of being inoperative or out of calibration. It is important that the external surfaces of button type indicators be kept free of dirt or paint to ensure free movement of the button.

Indications of excessive differential pressure, regardless of the type of indicator, should never be disregarded. All such indications must be verified and action taken to replace the loaded element. Failure to replace a stuck element can result in system starvation, filter element collapse, or the loss of filtration when bypass assemblies are used. Verification of loaded filter indications is particularly important with button type indicators as they may have been falsely triggered by mechanical shock, vibration, or a cold start of the system. Verification is usually obtained by manually resetting the indicator and operating the system to create a maximum flow demand ensuring that the fluid is at near normal operating temperatures.

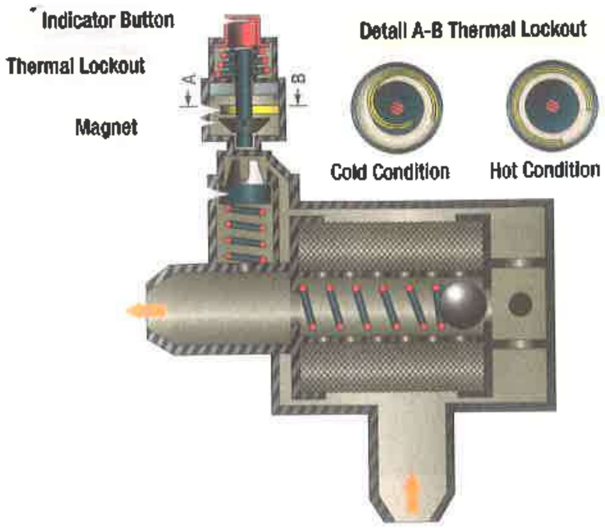


Figure 12-40. Filter bypass valve.

PRESSURE CONTROL

The safe and efficient operation of fluid power systems, system components, and related equipment requires a means of controlling pressure. There are many types of automatic pressure control valves designed for this purpose. Some provide a release of pressure that exceeds maximums; some only reduce the pressure for a lower pressure subsystem; and others keep the pressure in a system within an operating range.

PRESSURE RELIEF VALVES

Hydraulic pressure must be regulated to perform desired tasks. A pressure relief valve is used to limit the amount of pressure being exerted on a confined fluid. This is necessary to prevent failure of components or a rupture of hydraulic lines. The pressure relief valve is, in effect, a system safety valve.

The design of relief valves incorporates adjustable spring loaded valves. They are installed in such a manner as to discharge fluid from the pressure line into a reservoir return line when the pressure exceeds that for which the valve is adjusted. Various designs of pressure relief valves are in use, but in general, all employ a spring loaded device operated by hydraulic pressure and spring tension. (***Figure 12-41***)

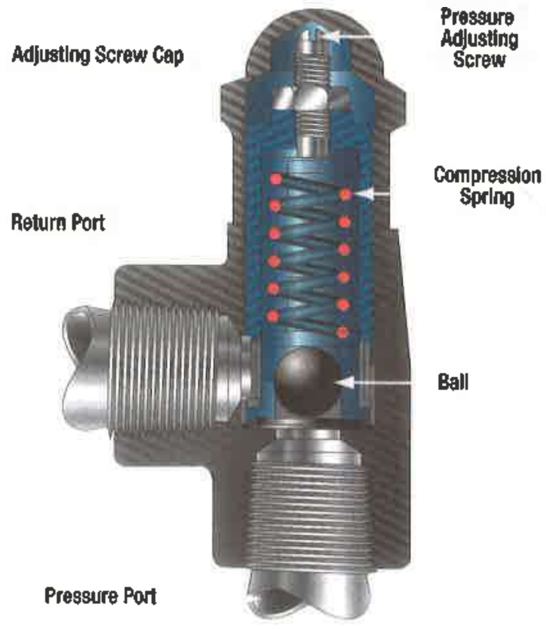


Figure 12-41. Pressure relief valve.

Pressure relief valves are adjusted by increasing or decreasing the tension on the spring to determine the pressure required to open the valve. They may be classified by type of construction or uses in the system. The most common types of valves are:

- Ball Type - in valves with a ball type device, the ball rests on a contoured seat. Pressure acting on the bottom of the ball pushes it off its seat, allowing the fluid to bypass.
- Sleeve Type - in valves with a sleeve type device, the ball remains stationary and a sleeve type seat is moved up by the fluid pressure. This allows the fluid to bypass between the ball and the sliding sleeve seat.
- Poppet Type - in relief valves with a poppet type device, a cone shaped poppet may have several design configurations, however it is basically a cone and seat machined at matched angles to prevent leakage. As the pressure rises to a predetermined setting, the poppet is lifted off its seat. This allows the fluid to pass through the opening created and out the return port.

Pressure relief valves cannot be used as pressure regulators in large hydraulic systems that depend on engine driven pumps as the primary source of pressure because the pump is constantly under load and the energy expended in holding the valve off its seat is changed into heat. This heat is transferred to the fluid and in turn, to the packing rings, causing them to deteriorate rapidly. They may however, be used as regulators in small, low pressure systems or when the pump is electrically driven and only used intermittently.

Pressure relief valves may be used as:

1. System relief valves - the most common use of the pressure relief valve is as a safety device against the possible failure of a pump, compensator or other regulating device. All hydraulic systems that have hydraulic pumps incorporate pressure relief valves.
2. Thermal relief valve - the thermal relief valve is used to relieve excessive pressures resulting from thermal expansion of the fluid. They are used where a check valve or selector valve prevents pressure from being relieved through the main system relief valve. Thermal relief valves are usually smaller than system relief valves.

PRESSURE REGULATORS

A pressure regulator is used in hydraulic systems that are pressurized by constant delivery type pumps. One purpose of the pressure regulator is to manage the output of the pump to maintain system operating pressure within a predetermined range. The other purpose is to permit the pump to turn without resistance (unloading the pump) at times when pressure in the system is within normal operating range. The pressure regulator is placed in the system so that the pump output can get into the system pressure circuit only by passing through the regulator. The combination of a constant delivery pump and a pressure regulator is virtually the equivalent of a compensator controlled, variable delivery type pump. (***Figure 12-42***)

PRESSURE REDUCERS

Pressure reducers are used in hydraulic systems where it is necessary to lower the normal operating pressure by a specified amount. Reducing valves provide a steady pressure into a system that operates at a lower pressure than the supply system. A reducing valve can normally be set for any desired downstream pressure within the design limits of the valve. Once the valve is set, the reduced pressure is maintained regardless of changes in supply pressure (as long as the supply pressure is at least as high as the reduced pressure desired) and regardless of the system load, if the load does not exceed the capacity of the reducer. (***Figure 12-43***)

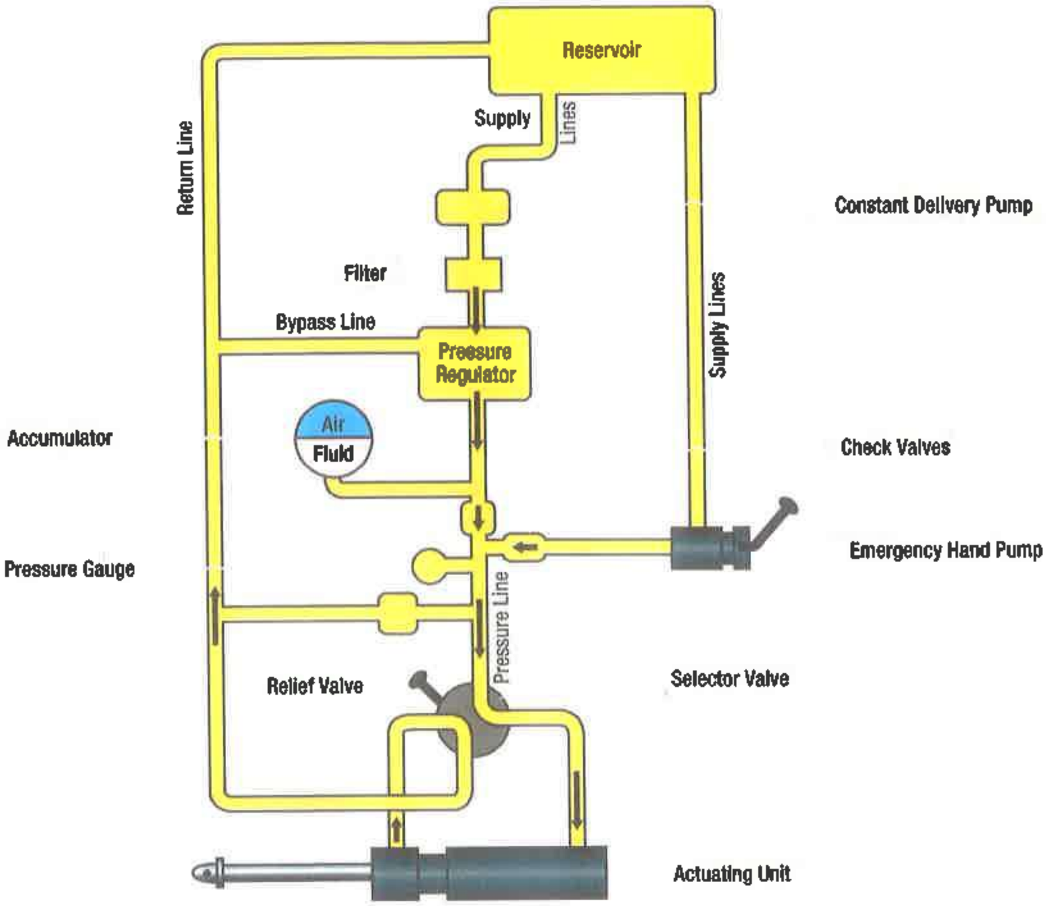


Figure 12-42. The location of a pressure regulator.

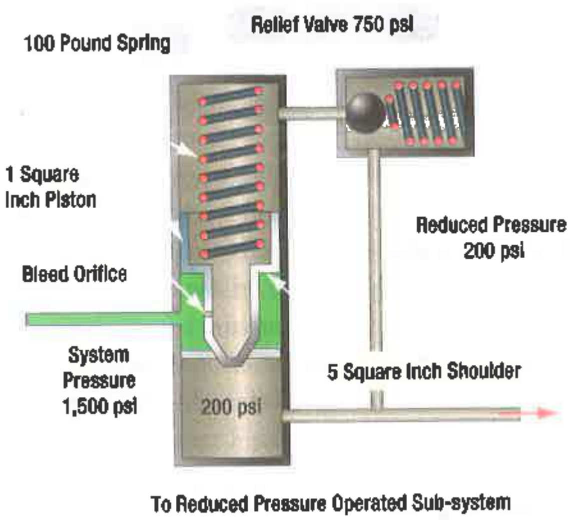


Figure 12-43. Operating mechanism.

POWER DISTRIBUTION

Power distribution in a hydraulic system is controlled by a variety of flow control valves. These valves control the speed and/or direction of fluid flow in the system. They provide for the operation of various components when desired and the speed at which the component operates. Examples of flow control valves include selector valves, check valves, sequence valves, priority valves, shuttle valves, quick disconnect valves, hydraulic fuses and shutoff valves.

SHUTOFF VALVES

Shutoff valves are used to shut off the flow of fluid to a particular system, subsystem, or component. In general, shutoff valves are electrically powered. They are used to distribute hydraulic power to various components in the system. (***Figure 12-44***)

Figure 12-44. Shutoff valves.

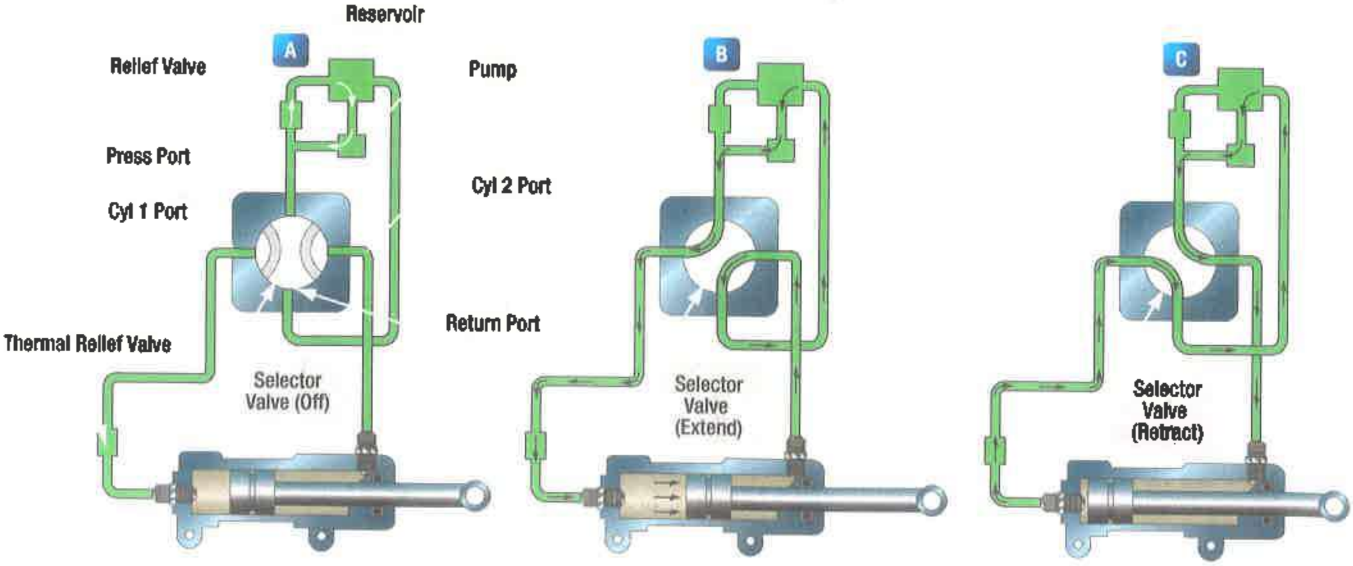


Figure 12-45. Operation of a closed-center four-way selector valve.

A shutoff valve may be used to create a priority in a hydraulic system. In this case, the valve is controlled by a pressure switch. The switch is adjusted so that if pressure drops below the set point, the valve closes. All components upstream of the shutoff valve are prioritized in that they receive system pressure even when full pressure is not available.

SELECTOR VALVES

Selector valves control the direction of movement of a hydraulic actuating cylinder or similar device. It provides for the simultaneous flow of hydraulic fluid both into and out of the unit. Hydraulic system pressure can be routed with the selector valve to operate the unit in any direction, and/or create a corresponding return path to the reservoir. There are two main types of selector valves: open center and closed center. An open center valve allows a continuous flow of hydraulic fluid even when the selector is not in the position to actuate a unit. A closed center selector valve blocks the flow of fluid when it is in the neutral or off position. (***Figure 12-45A***)

Selector valves may be poppet type, spool type, piston type, rotary type, or plug type. (***Figure 12-46***) Regardless, each selector valve has a unique number of ports determined by the requirements of the system in which it is used. Closed centered selector valves with four ports (4 way valves) are most common in aircraft systems. ***Figure 12-45*** illustrates how this valve connects to the pressure and return lines of a hydraulic system, as well as to the two ports on a common actuator.

Figure 12-46. A poppet-type four-way selector valve.

Most selector valves are mechanically controlled by a lever or electrically controlled by solenoid or servo. (***Figure 12-47***) The four ports on a four-way selector valve always have the same function. One port receives pressurized fluid from the system hydraulic pump. A second port always returns fluid to the reservoir. The third and fourth ports are used to connect the selector valve to the actuating unit.

There are two ports on the actuating unit. When the selector valve is positioned to connect pressure to one port on the actuator, the other actuator port is simultaneously connected to the reservoir return line through the selector valve. (***Figure 12-45B***) Thus, the unit operates in a certain direction. When the selector valve is positioned to connect the pressure of the other port on the actuating unit, the original port

Figure 12-47. Four-way servo control valve.

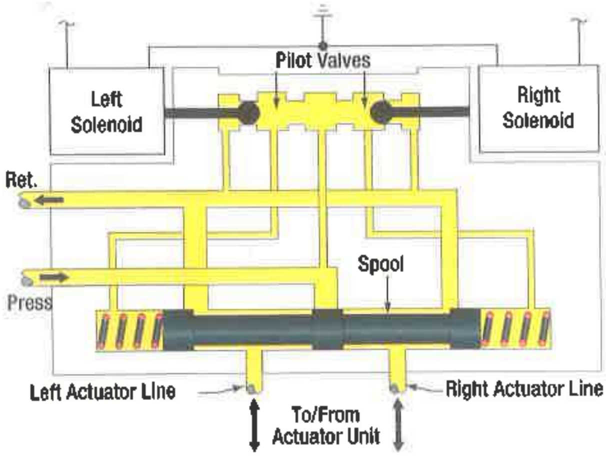


Figure 12-48. Servo control valve solenoids not energized.

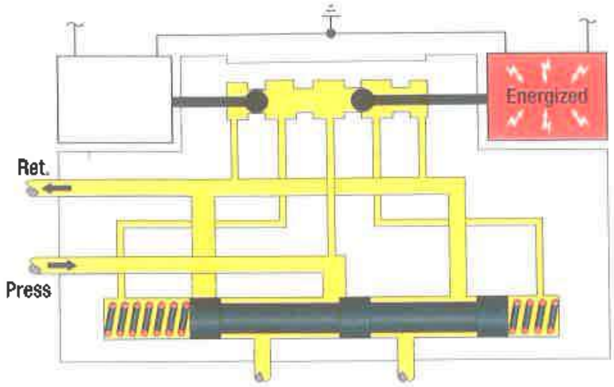


Figure 12-49. Servo control valve right solenoid energized.

is simultaneously connected to the return line through the selector valve and the unit operates in the opposite direction. (***Figure 12-45C***)

Figure 12-48 illustrates the internal flow paths of a solenoid operated selector valve. The closed center valve is shown in the neutral or off position. Neither solenoid is energized. The pressure port routes fluid to the center lobe on the spool which blocks the flow. Fluid pressure flows through the pilot valves and applies equal pressure on both ends of the spool. The actuator lines are connected around the spool to the return line. When selected via a switch in the cockpit, the right solenoid is energized. The right pilot valve plug shifts left, which blocks pressurized fluid from reaching the right end of the main spool. The spool slides to the right due to greater pressure applied on the left end of the spool. The center lobe of the spool no longer blocks the system pressurized fluid which flows to the actuator through the left actuator line. At the same time, return flow is blocked from the main spool left chamber so the actuator (not shown) moves in the selected direction. Return fluid from the moving actuator flows through the right actuator line past the spool and into the return line. (***Figure 12-49***)

The actuator contacts a limit switch when the desired motion is complete. The switch causes the right solenoid to deenergize and the right pilot valve reopens. Pressurized fluid can once again flow through the pilot valve and into the main spool right end chamber. There, the spring and fluid pressure shifts the spool back to the left into the neutral or off position shown in ***Figure 12-48***. To make the actuator move in the opposite direction, the cockpit switch is moved in the opposite direction. All motion inside the selector valve is the same as described above but in the opposite direction. The left solenoid is energized. Pressure is applied to the actuator through the right port and return fluid from the left actuator line is connected to the return port through the motion of the spool to the left.

CHECK VALVES

A check valve allows fluid to flow unimpeded in one direction but prevents flow in the opposite direction. A check valve may be an independent component situated in-line somewhere in the system or it may be built into a component. A typical check valve consists of a spring loaded ball and seat inside a housing. The spring compresses to allow fluid flow in the designed direction. When flow stops, the spring pushes the ball against the seat which prevents fluid from flowing in the opposite direction through the valve. An arrow on the outside of the housing indicated the direction in which fluid flow is permitted. (***Figure 12-50***) A check valve may also be constructed with a spring loaded flapper or coned shape piston instead of a ball.

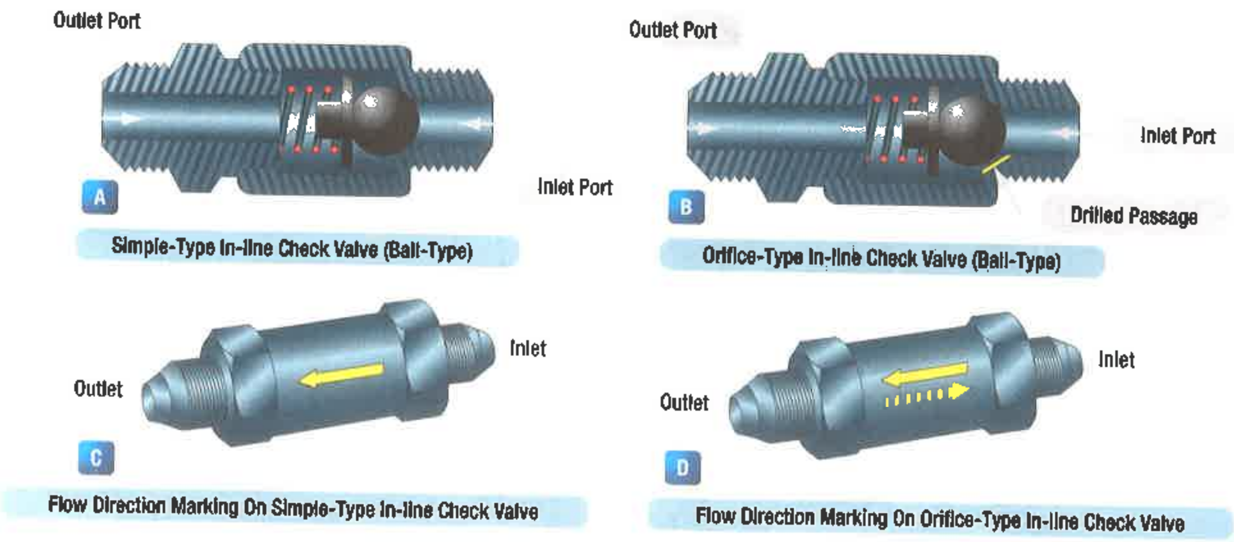


Figure 12-50. An in-line check valve and orifice type.

Orifice Type Check Valve

Some check valves allow full flow in one direction but only restricted flow in the opposite direction. These are known as orifice check valves or damping valves. The valve contains the same spring, ball, and seat combination as a normal check valve, but the seat area has a calibrated orifice machined into it. Thus, fluid flow is unrestricted in one direction while the ball is pushed off of its seat. However, when fluid back flows, the spring forces the ball against the seat which limits fluid flow to the amount that can pass through the orifice. The reduced flow in this opposite direction slows the motion, or dampens the actuator associated with the check valve. (***Figure 12-50***) An orifice check valve may be included in a hydraulic landing gear actuator system. When the gear is raised, the check valve allows full fluid flow to lift the heavy gear at maximum speed. When lowering the gear, the orifice in the valve prevents the gear from violently dropping by restricting fluid flow out of the actuating cylinder.

SEQUENCE VALVES

Sequence valves control the sequence of operation between two branches in a circuit. They enable one unit to automatically set another unit into motion. For example, in a landing gear actuating system, the landing gear doors must open before the landing gear starts to extend. Conversely, the landing gear must be completely retracted before the doors close. A sequence valve installed in each landing gear actuating line performs this function. A sequence valve is somewhat like a relief valve except that, after the set pressure has been reached, the sequence valve diverts the fluid to a second actuator to do work in another part of the system. There are various types of sequence valves. Some are controlled by pressure, some mechanically, and some are controlled by electric switches.

Pressure Controlled Sequence Valve

The operation of a pressure controlled sequence valve is illustrated in ***Figure 12-48***. The opening pressure is obtained by adjusting the tension of the spring that normally holds the piston in the closed position. (Note that the top part of the piston has a larger diameter than the lower part.) Fluid enters the valve through the inlet port, flows around the lower part of the piston and exits the outlet port, where it flows to the primary (first) unit to be operated. (***Figure 12-51A***)

When the primary actuating unit completes its operation, pressure in the line to the actuating unit increases sufficiently to overcome the force of the spring, and the piston rises. The valve is then in the open position. (***Figure 12-51B***) The fluid entering the valve takes the path of least resistance and flows to the secondary unit. A drain passage is provided to allow any fluid leaking past the piston to flow from the top of the valve and is usually connected to the main return line.

Mechanically Operated Sequence Valve

The mechanically operated sequence valve is operated by a plunger that extends through the body of the valve. (***Figure 12-52***) The valve is mounted so that the plunger is operated by the primary unit. A check valve, either a ball or poppet, is installed between the fluid ports in the body. It can be unseated by either the plunger or fluid

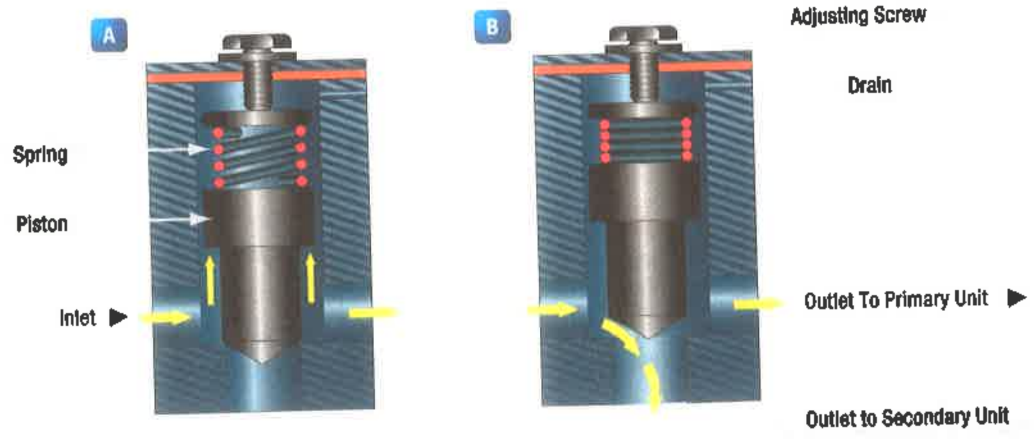


Figure 12-51. A pressure-controlled sequence valve.

pressure. Port A and the actuator of the primary unit are connected by a common line. Port B is connected by a line to the actuator of the secondary unit.

When fluid under pressure flows to the primary unit, it also flows into the sequence valve through port A to the seated check valve in the sequence valve. To operate the secondary unit, the fluid must flow through the sequence valve. The valve is located so that the primary unit moves the plunger as it completes its operation. The plunger unseats the check valve and allows the fluid to flow through the valve, out port B, and to the secondary unit.

PRIORITY VALVES

A priority valve gives priority to the critical hydraulic subsystems over noncritical systems when system pressure is low. For instance, if the pressure of the priority valve is set for 2 200 psi, all systems receive pressure when the pressure is above 2 200 psi. If the pressure drops below 2 200 psi, the priority valve closes, and no fluid flows to the noncritical systems. (*Figure 12-53*) Some hydraulic designs use pressure switches and electric shutoff valves to assure that the critical systems have priority over noncritical systems if system pressure is low.

SHUTTLE VALVES

In certain fluid power systems, the supply of fluid must be from more than one source to meet system requirements. In some systems, an emergency system is provided as a source of pressure in the event of normal system failure. The emergency system actuates only essential components. The purpose of the shuttle valve is to isolate the normal system from an alternate or emergency system. It is small and simple; yet a very important component. (*Figure 12-54*)

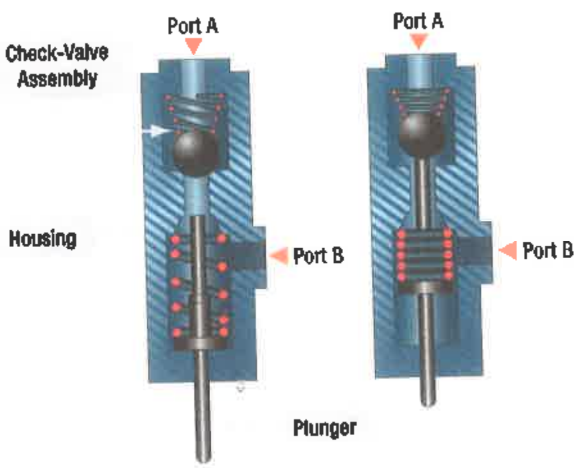


Figure 12-52. Mechanically operated sequence valve.

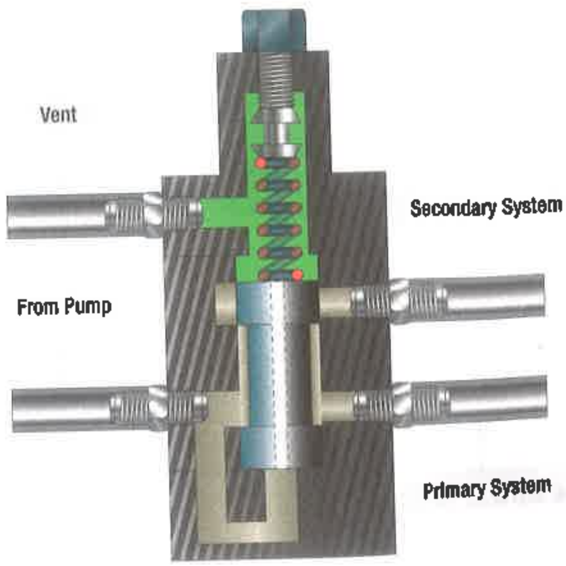


Figure 12-53. Priority valve.

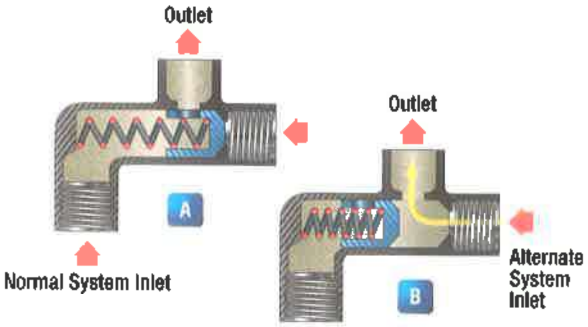


Figure 12-54. A spring-loaded piston-type shuttle valve.

Its housing contains three ports - normal system inlet, alternate or emergency system inlet, and outlet. A shuttle valve used to operate more than one actuating unit may contain additional outlet ports. Enclosed in the housing is a sliding part called the shuttle. Its purpose is to seal off one of the inlet ports. When a shuttle valve is in the normal operation position, fluid has a free flow from the normal system inlet port, through the valve, and out the outlet port to the actuating unit. The shuttle is seated against the alternate system inlet port and held there by normal system pressure and by a shuttle valve spring. The shuttle remains in this position until the alternate system is activated. This action directs fluid under pressure from the alternate system to the shuttle valve and forces the shuttle from the alternate inlet port to the normal inlet port. Fluid from the alternate system then has a free flow to the outlet port but is prevented from entering the normal system by the shuttle which seals off the normal system port.

The shuttle may be one of four types:

1. Sliding plunge.
2. Spring loaded piston.
3. Spring loaded ball.
4. Spring loaded poppet.

In shuttle valves that are designed with a spring, the shuttle is normally held against the alternate system inlet port by the spring.

QUICK DISCONNECT VALVES

Quick disconnect valves are installed in hydraulic lines to prevent loss of fluid when units are removed. Such valves are installed in the pressure and suction lines of the system immediately upstream and downstream of the power pump. In addition to pump removal, a power pump can be disconnected from the system and a hydraulic test stand connected in its place.

These valve units consist of two interconnecting sections coupled together by a nut when installed in the system. Each valve section has a piston and poppet assembly. These are spring loaded to the closed position when the unit is disconnected. (*Figure 12-55*)

HYDRAULIC FUSES

A hydraulic fuse is a safety device. Fuses may be installed at strategic locations throughout a hydraulic system. They detect a sudden increase in flow, such as a burst downstream, and shut off the fluid flow. By closing, a fuse preserves hydraulic fluid for the rest of the system. One type of fuse, referred to as the automatic resetting type is designed to allow a certain volume of fluid per minute to pass through it. If the volume passing through the fuse becomes excessive, the fuse closes and shuts off the flow. When the pressure is removed it automatically resets itself to the open position. Fuses are usually cylindrical in shape, with an inlet and outlet port at opposite ends. (*Figure 12-56*)

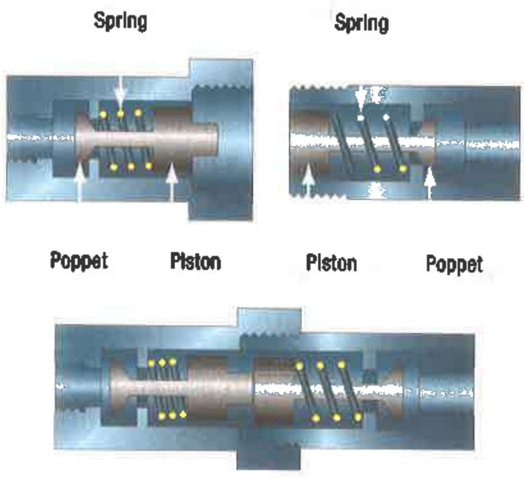


Figure 12-55. A hydraulic quick-disconnect valve.

Figure 12-56. Hydraulic fuse.

HYDRAULIC ACTUATORS

A hydraulic actuating cylinder transforms energy in the form of fluid pressure into a mechanical force to perform work. It is used to impart linear motion to some movable object or mechanism. Actuating cylinders consists of a cylinder housing, one or more pistons and piston rods, and some seals. The housing contains a polished bore in which the piston operates, and one or more ports through which fluid enters and leaves the bore. The piston and rod form an assembly. The piston moves forward and backward within the bore, and an attached piston rod moves into and out of the cylinder housing through an opening in one end. Seals are used to prevent leakage between the piston and the cylinder bore and between the piston rod and the end of the cylinder. Both the cylinder housing and the piston rod have provisions for mounting and for attachment to mechanism that is to be moved.

Actuating cylinders are of two types: single action and double action. The single action (single port) type can produce powered movement in one direction only. The double action (two ports) type can produce powered movement in two directions.

Linear Actuators

A single action actuating cylinder is illustrated in ***Figure 12-57A***. Fluid under pressure enters the port at the left and pushes against the face of the piston, forcing the piston to the right. As the piston moves, air is forced out of the spring chamber through the vent hole, compressing the spring. When pressure on the fluid is released to the point it exerts less force than from the compressed spring, the spring pushes the piston toward the left. As the piston moves to the left, fluid is forced out of the fluid port. At the same time, the moving piston pulls air into the spring chamber through the vent hole. A three way valve is normally used for controlling the operation of a single action actuating cylinder. A double action cylinder is illustrated in ***Figure 12-57*** and is usually controlled by a four way selector valve.

Figure 12-58 shows an actuating cylinder interconnected with a selector valve. When the selector valve is placed in the on or extend position, fluid is admitted under pressure to the left hand chamber of the actuating cylinder. This results in the piston being forced toward the right. As the piston moves toward the right, it pushes return fluid out of the right hand chamber and through the selector valve to the reservoir. When the selector valve is placed in its retract position, as in ***Figure 12-58***, fluid pressure enters the right chamber, forcing the piston toward the left. As the piston moves toward the left, it pushes return fluid out of the left chamber and through the selector valve to the reservoir. Besides having the ability to move a load into position, a double acting cylinder also can hold a load. This capability exists because when the selector valve used to control operation of the actuating cylinder is placed in the off position, fluid is trapped in the chambers on both sides of the actuating piston. Internal locking actuators also are used in some applications.

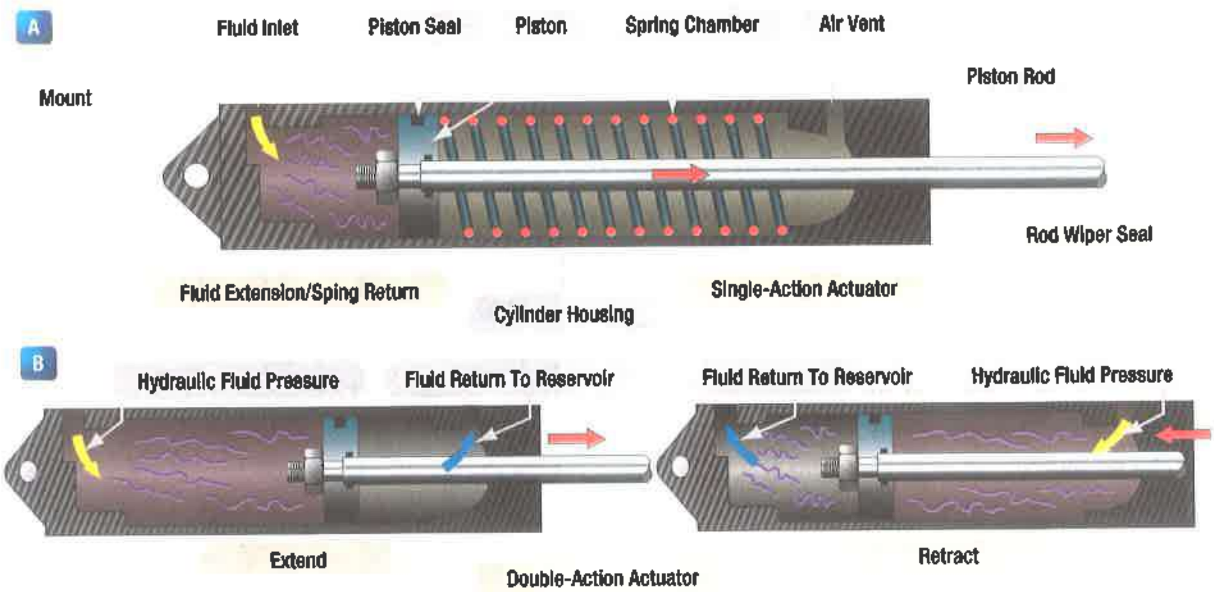


Figure 12-57. Linear actuator.

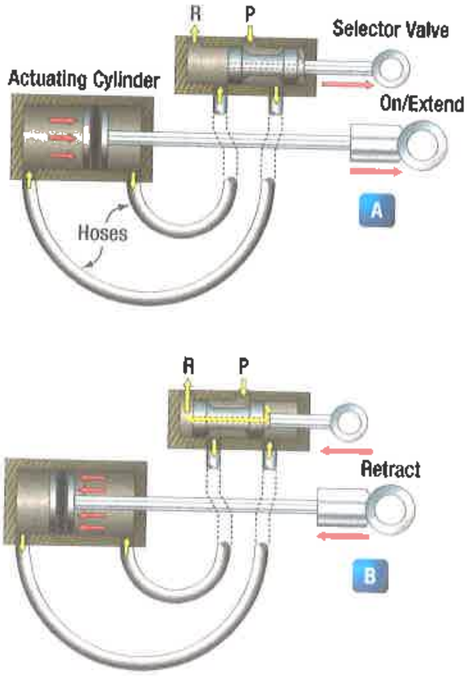


Figure 12-58. Linear actuator operation.

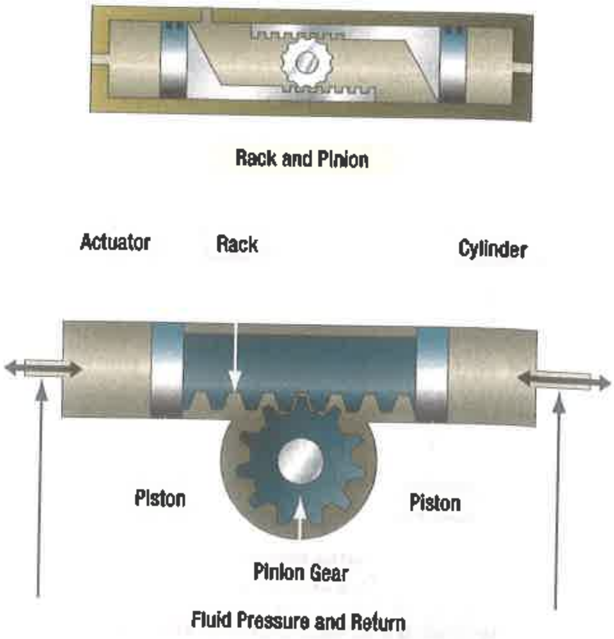


Figure 12-59. Rack and pinion gear.

Rotary Actuators

Rotary actuators can mount directly to the part without taking up the long stroke lengths required by cylinders. Rotary actuators are not limited to the 90° pivot arc typical of cylinders. They can achieve arc lengths of 180°, 360°, or even 720°, depending on the configuration. An often used type of rotary actuator is the rack and pinion type used for many nose wheel steering mechanisms. In a rack-and-pinion actuator, a long piston with one side machined into a rack engages a pinion to turn the output shaft. (*Figure 12-59*) One side of the piston receives fluid pressure while the other side is connected to the return. When the piston moves, it rotates the pinion.

INDICATION AND WARNING SYSTEMS

There are just a few hydraulic system indications on the flight deck. Fluid pressure and temperature are the primary parameters monitored as well as fluid quantity. Reservoir pressurization may also be monitored. Electro-hydraulic transducers are mounted in the system in key locations so that pressures and temperatures can be displayed on a gauge or LCD screen. A separate transmitter and indication is often used for brake pressure.

For servicing and maintenance, direct reading indicators are installed so that technicians can observe system status while on the ramp. System pressure sensors are often located at the hydraulic pressure filter modules. Low pressure warning switches are located downstream of the pump outlet and may also be at the module. A hydraulic panel on the flight deck incorporates pump switches and temperature and pressure indications in older aircraft. Warnings and indications are displayed on status screens away from the switches on glass cockpit aircraft.

Figure 12-60 illustrates a typical hydraulic control panel on the flight deck of an older Boeing aircraft. *Figure 12-61* illustrates system status, synoptic and maintenance page displays for the hydraulic system on a glass cockpit aircraft.

Figure 12-60. Hydraulic panel example.

Hydraulic fluid quantity is monitored at the reservoir by means of a float gauge, sight glass, or other sensing mechanism which sends a signal to the flight deck for gauge or LCD display. A low quantity warning switch may be included.

Hydraulic system fluid temperature indication is usually limited to an overheat annunciator for each pump or system. Temperature switches, often located in the return line as the fluid enters the reservoir trip when a preset temperature is reached. A signal is sent to the flight deck for annunciation. Temperature sensors for hydraulic systems with electrically driven pumps serve as motor temperature monitoring devices as well. Motor driven pumps are more likely to get hot than engine driven pumps. The hot motor transfers some of its heat to the fluid as it circulates.

Hydraulic system warnings include low pressure annunciations for each system. Typically, a lamp will illuminate, flash, or change color on the flight deck when a pressure sensor sends an electronic signal that a low pressure condition exists.

Many indicators display a low pressure warning when the hydraulic pumps are off which goes away when the pumps are switched on and operate normally.

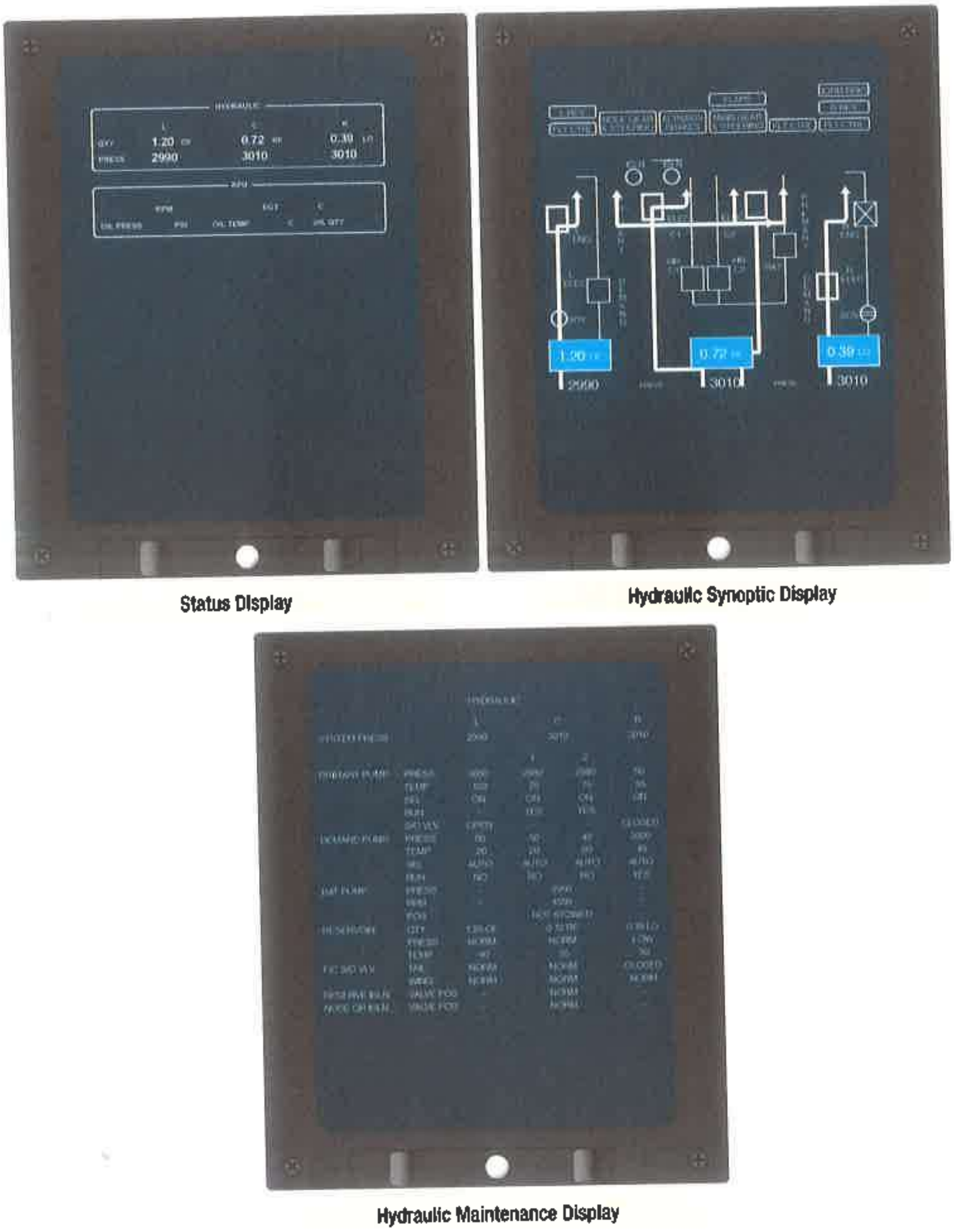


Figure 12-61. Hydraulic system information.

INTERFACE WITH OTHER SYSTEMS

Many aircraft systems use hydraulic power such as landing gear extension and retraction, flight controls, and auto pilot. In most cases, the operational logic for these advanced systems are controlled by computers. To integrate the mechanical power of the hydraulic system, hydraulic system parameters and status conditions must be input into the controlling computer. In the absence of any malfunction, the computer controller activates the correct hydraulic system components when needed. Vital systems control logic can also dictate operation in alternate modes should the hydraulic system parameters be out of the normal operating range. For example, if the hydraulic pump used for normal operations is not maintaining acceptable system pressure, logic circuits reconfigure the operational mode from 'normal' to an alternate mode that utilizes the backup hydraulic pump. Hydraulic system parameters that are captured in analog format are converted to digital format for use in the control system logic.

Question: 12-1
What is the primary difference between an open and closed center hydraulic system?

Question: 12-2
What is the importance of viscosity in hydraulic fluids?

Question: 12-3
Which type of hydraulic fluid is both the most fire resistant and the most thermally stable at high temperatures?

Question: 12-4
What is the primary method of cooling hydraulic fluid in large helicopters?

Question: 12-5
What determines the pressure output of a constant displacement hydraulic pump?

Question: 12-6
How are O-rings identified as to their type and size?

Question: 12-7
What is the primary purpose of a Hydraulic motor Driven Generator?

Question: 12-8
What occurs when a hydraulic fluid filter becomes clogged?

ANSWERS

Answer: 12-1
An open center system is pressurized only when the actuating mechanisms are functioning.

Answer: 12-2
That the fluid is non-viscous enough so as not to impede flow, yet viscous enough to resist leakage.

Answer: 12-3
Phosphate Ester Type V fluids.

Answer: 12-4
A heat exchanger located within a fuel tank.

Answer: 12-5
Its operating speed in rotations per minute.

Answer: 12-6
By the labeling on its package.

Answer: 12-7
To provide backup electrical power in the case of an electrical system failure.

Answer: 12-8
A downstream sensor senses low pressure and opens a valve to bypass the filter.

HELICOPTER AERODYNAMICS, STRUCTURES AND SYSTEMS

ICE AND RAIN PROTECTION (ATA 30)

SUB-MODULE 13

PART-66 SYLLABUS LEVELS	
CERTIFICATION CATEGORY →	B1.3 B1.4

Sub-Module 13

ICE AND RAIN PROTECTION (ATA 30)

Knowledge Requirements

12.13 - Ice and Rain Protection (ATA 30)	
Ice formation, classification and detection;	3
Anti-icing and de-icing systems: electrical, hot air and chemical;	
Rain repellent and removal;	
Probe and drain heating;	
Wiper system.	

12.13 - ICE AND RAIN PROTECTION (ATA 30)

Rain, snow and ice are longtime enemies of transportation. Flying has added a new dimension, particularly with respect to ice. Under certain atmospheric conditions, ice can build rapidly on airfoils and air inlets. On days when there is visible moisture in the air, ice can form on blade leading edge surfaces when freezing temperatures start. Water droplets in the air can be supercooled to below freezing without turning into ice unless they are disturbed in some manner. This unusual occurrence is partly due to the surface tension of the water droplet not allowing the droplet to expand and freeze. However, when aircraft surfaces disturb these droplets, they immediately turn to ice on those surfaces. (*Figure 13-1*)

To perform as designed; airfoils, including rotor blades and engine inlets must be completely smooth and free of any irregularities or contamination in the form of ice, snow, or frost. Even a small amount of surface contamination can reduce lift and raise the stall speed. Accidents have occurred due to airfoil surface roughness caused by frost. The additional weight caused by ice accumulation is also problematic.

ICE FORMATION, CLASSIFICATION AND DETECTION

TYPES OF AIRCRAFT ICE

There are three types of ice encountered during flight: clear, rime, and gleam ice.

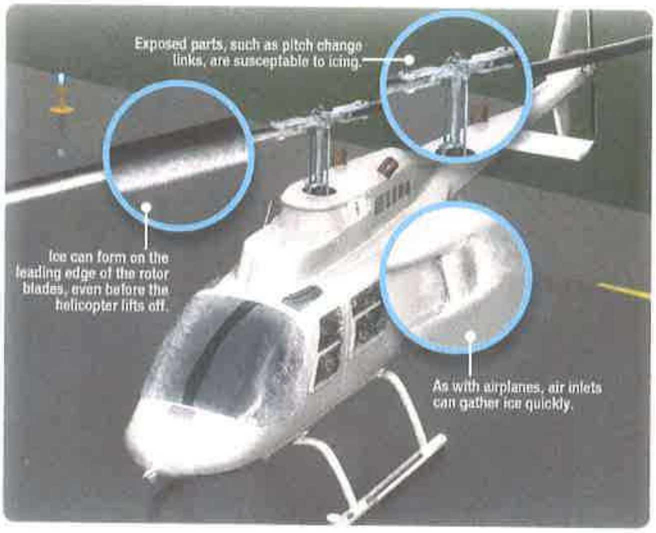


Figure 13-1. Ice formation can effect various parts of a helicopter structure.

Clear Ice

Clear ice forms when water drops flow out over the aircraft surface, gradually freezing as a smooth sheet of solid ice. Formation occurs when droplets are large, such as in rain or in cumulus clouds. The water remains a liquid for a period of time and runs back over the surface until it has lost enough heat to freeze. Clear ice is hard, heavy, and tenacious. Its removal by de-icing equipment is especially difficult. (*Figure 13-2*)

Rime Ice

Rime ice forms when water drops are small, such as those in stratified clouds or light drizzle. The liquid portion remaining after initial impact freezes rapidly before the drop has time to spread over the surface. The small frozen droplets trap air giving the ice a white appearance. Rime ice is lighter in weight than clear ice, however its weight is of little significance. The irregular shape and rough surface of rime ice decreases the effectiveness and efficiency of the aerodynamic airfoils. This reduces lift and increases drag. Rime ice is brittle and more easily removed than clear ice. (*Figure 13-3*)

Mixed clear and rime icing can form rapidly when water drops vary in size or when liquid drops intermingle with

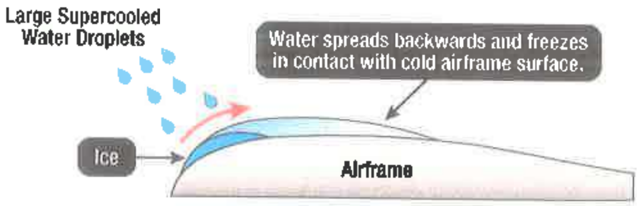


Figure 13-2. Formation of clear ice.

Figure 13-3. Rime ice buildup on an aircraft structure.

snow or ice particles. Ice particles become embedded in clear ice, building a very rough accumulation sometimes in a mushroom shape on leading edges. Ice may be expected to form whenever there is visible moisture in the air and the temperature is near or below freezing. An exception is carburetor icing, which can occur during warm weather with no visible moisture present.

Gleam Ice

Gleam ice will form when complete freezing of the water particles on impact takes longer than in the case of rime ice. Such ice will form when the water particles are large and the air temperature very low. The remaining water will freeze rapidly to trap some air, giving the ice an opaque appearance. The delay in freezing of the residual water gives it time to flow back. The ice that forms will extend further back over the leading edge and the surface will not be as rough as rime ice.

Hoarfrost

Hoarfrost appears as a thin uniform deposit of ice with a fine white crystalline texture. It forms in clear air on surfaces which are below the freezing point, with water vapor being converted directly into ice crystals, forming a white feathery coating. Active frost is similar to hoarfrost, but continuously reforms. Active frost can only be removed with de/anti-icing fluids. All types of frost need to be removed prior to flight, then will not reform.

ICING EFFECTS

Ice or frost forming on aircraft creates hazards detrimental to flight. The resulting malformation of the airfoil when ice adheres to it decreases the amount of lift the airfoil can produce. The additional weight and unequal formation of the ice may also cause unbalancing of the aircraft, making it hard to control. Enough ice to cause an unsafe flight condition can form in a very short period, thus some method of ice prevention or removal is necessary.

Ice buildup decreases main and tail rotor efficiency because of the change in blade shape. It causes destructive vibration and hampers true instrument readings. Control surfaces become unbalanced or frozen. Fixed slots are filled, and movable slots jammed. Radio reception is hampered, and engine performance is affected. Ice, snow, and slush have a direct impact on the safety of flight. Not only because of degraded lift, reduced take off performance and/or maneuverability of the aircraft, but when chunks break off, they can also cause engine failures and structural damage.

Icing in flight predominantly forms on leading edges and protrusions such as pitot probes or antenna.

In this sub-module, ice prevention (anti-ice) and ice elimination (de-ice) using electric, pneumatic, and chemical systems are discussed. The ice and rain protection systems used on helicopters involve the following components:

- Blades Of Main And Tail Rotors
- Horizontal And Vertical Stabilizer Leading Edges
- Engine Cowl Leading Edges
- Air Data Probes
- Flight Deck Windows
- Antenna

ICE DETECTION

While some ice can be detected visually, most modern aircraft have one or more ice sensors and enunciators that warn the flight crew of icing conditions. In some aircraft, multiple ice detectors are used. In others, ice detection systems automatically turn on certain anti-ice systems when icing is detected. Ice detectors are mounted on the sides of the forward fuselage to receive impact air as the aircraft moves forward. Two independent detectors and detector systems are a normal configuration for redundancy reasons, to be sure the information is received from one of the two.

In all cases when working around ice detector probes, note that the probes get extremely hot when activated and can cause serious burns. Also note that electrostatic handling precautions should be exercised when handling ice detectors.

Pressure Ice Detectors

A pressure ice detector consists of an elliptical shaped tube. Mounted on the base is a sensitive pressure switch, actuated by a diaphragm. The detector is mounted in the air flow. In the leading edge of the tube small holes are drilled and connected to the lower side of a diaphragm. A large hole is connected to the top of the diaphragm. The total area of the small holes in the leading edge exceeds that of the large hole.

If ice builds up on the leading edge, the small holes are blocked faster than the large hole. Therefore, the dynamic air pressure on the upper side of the diaphragm overcomes the pressure on the lower side and the pressure switch will be actuated. When there is no ice buildup on the detector, the higher pressure on the underside of the diaphragm keeps the pressure switch open.

If the pressure switch closes due to ice accumulation the ice warning relay will be energized and the warning light and the heater within the detector are activated. After approximately 20 seconds the ice is melted and the pressure switch reopens, ready for a new ice warning cycle. This cycling will continue until such time that the icing conditions no longer exist. (***Figure 13-4***)

Vibrating Ice Detectors

The typical vibrating anti-ice detector contains a probe which is vibrated at an established rate. Inside the probe housing are circuit cards and a microprocessor. The probe is electrically connected to the ice protection control unit and the aircraft data buses.

Should ice collect on the ice detector probe, the established frequency of vibration decreases. A slight lowering of the frequency causes the integral electric probe heater to come ON. The heater quickly melts the ice in 5-7 seconds and shuts OFF. If ice reforms on the probe, the heater will cycle ON again, melt the ice and turn OFF again. The control logic inside the detector unit monitors the heater cycles. When the probe heat cycles two or more times, the detector automatically sends an icing signal, and the rotor blade anti-ice system is automatically turned ON. An annunciation is also made to alert the crew. Note that the aircraft must be in the air with the anti-ice switch on the flight deck set to AUTO.

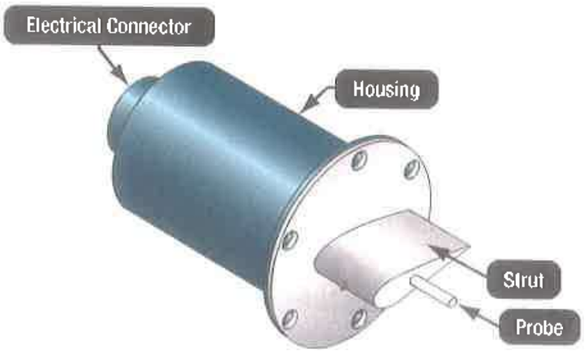


Figure 13-4. A pressure ice detector.

The example shown in ***Figure 13-5*** uses an ultrasonic, axially vibrating probe to detect the presence of icing conditions. The sensing probe is a nickel alloy tube that has a natural resonant frequency of 40 kHz and is mounted in the strut at its mid-point with one inch exposed to the air stream.

Hot Rod Ice Detector

A hot rod detector consists of an oblong base in aluminum alloy on which is mounted a detector mast made of an aerodynamic section of steel tube, inclined backwards at about 30° from the vertical, mounted on the side of the fuselage, so that it can be seen from the flight compartment windows. The mast houses a heating element. The heating element is normally off. When icing conditions are met, ice accumulates on the leading edge of the mast. This can then be observed by the flight crew. During nighttime operations, the built-in spotlight can be turned on to illuminate the mast. By manually selecting a switch at the heating element, the ice formed is dispersed for later observation. (***Figure 13-6***)

Serrated Rotor Ice Detector Head

A serrated rotor detector incorporates an integrated drive shaft coupled to a small AC motor via a reducer which rotates alongside a fixed blade knife. The motor

Figure 13-5. A Goodrich 0871LH1 freezing rain sensor and ice detector.

Figure 13-6. A Hot Rod type ice detector.

housing is connected via a spring loaded rocker bar to a set of microswitches. The motor and gearbox assembly is mounted on a static lug fixed to the motor housing. It and a microswitch assembly are surrounded by a cylindrical housing. The detector is mounted through the side of the fuselage so that the interior housing is subjected to ambient conditions with the exterior being pressure tight from the aircraft cabin.

The serrated rotor is continuously driven by the electric motor so that its periphery rotates within 0.050 mm of the leading edge of the bladed knife. The torque required to drive the rotor in non-icing conditions is low since it only must overcome the friction of the bearings. If ice builds up on the rotor so that the space between it and the knife edge is filled, a substantial increase in torque is required causing a toggle bar to move against its spring loaded assembly. This activates a microswitch to trigger a warning signal. Once the icing conditions cease, the knife cutter will no longer shave the ice, the torque load will reduce, the motor's torque will return, and the switch returned to its normal position. (***Figure 13-7***)

ANTI-ICING AND DE-ICING SYSTEMS: ELECTRICAL, HOT AIR AND CHEMICAL

Ice control systems are designed for anti-icing or for de-icing. Anti-icing equipment is turned on before entering icing conditions and is designed to prevent ice from forming. A surface may be anti-iced by keeping it dry, by heating to a temperature that evaporates water on contact, or by heating the surface just enough to prevent freezing. De-icing equipment is designed to remove ice after it begins to accumulate. Ice may be controlled on aircraft structures by the methods described in ***Figure 13-8.***

Figure 13-7. A serrated rotor ice detector head.

LOCATION OF ICE	METHOD OF CONTROL	
	Anti-Ice	De-Ice
Leading Edge Of The Blade	Thermal Pneumatic	
	Thermal Electric	
	Chemical	Chemical
Engine Inlets	Thermal Pneumatic	
	Thermal Electric	
Leading Edge of Horizontal and Vertical Stabilizers	Thermal Pneumatic	
	Thermal Electric	
	Chemical	
Pitot Tube, Static Ports, Air Data Sensors, Water Drain, Tanks and Lines	Thermal Electric	
Windshield/Flight Deck Windows	Thermal Pneumatic	Thermal Pneumatic (Primarily for De-fogging)
	Thermal Electric	

Figure 13-8. Methods of ice control.

Where anti-icing methods are used, the protection system is switched on prior to encountering icing conditions and remains on so that no ice is allowed to form on the surface. Typical anti-icing systems are used for engine air intakes, air intakes, air data sensors (pitot tubes etc.), cockpit windows and windshields, and water outlets.

With de-icing methods, the protection system is automatically switched ON and OFF at regular intervals. During the OFF period, a certain amount of ice is allowed to accumulate which will not seriously affect the aerodynamic shape of the surface. The ice is then removed by operating the system for a short time.

There are four main methods used for ice protection:

1. Thermal; by channeling hot air along the internal surfaces of the leading edges. This method is used mostly for the tail and for engine inlets.
2. Electric; by installing heaters on the leading edges of the surfaces. This method is used for rotating parts like the main and tail rotor.
3. Inflatable boots: by distorting the airfoil shape and forcing ice to break off. This method is sometimes used on the vertical and horizontal stabilizer.
4. Chemical: by reducing the freezing point of water so that ice will not adhere. This method is used throughout various helicopter structures.

THERMAL ANTI-ICE SYSTEMS

In thermal systems, the leading edges of the stabilizers or inlets are provided with a second inner skin positioned to form a small gap. Heated air is routed through this gap, providing enough heat in the outer skin of the leading edge to melt ice already formed and prevent additional ice from forming. The air is released into the atmosphere through outlets in the skin surfaces and in some cases, in the ends of the stabilizers. (***Figure 13-9***)

There are several methods by which heated air can be supplied, including purging air from a turbine engine compressor, heating dynamic air by passing it through a heat exchanger located in a system of engine exhaust gas, and by dynamic air combustion heating.

In a compressor bleed system, hot air is taken directly from an engine compressor stage and passes through the main line after mixing with a supply of cold air in a mixing chamber. In some systems, safety shut-off valves are provided to ensure that sufficient air flow for all defrost requirements are provided within acceptable duct pressures and structural limits.

In the heat exchanger method, exhaust gases can be diverted to pass between tubes through which outside air enters the main supply ducts. The supply of exhaust gases is generally regulated by a thermostatic damper installed in the pipe between the exhaust unit and the heat exchanger.

In a combustion heating system, dynamic air passes through a cylindrical envelope enclosing a sealed chamber in which a fuel/air mixture is burned and routed to contact the walls of the chamber. Air for combustion is derived from a separate air intake and is supplied to the chamber by means of a fan.

Thermal Engine Anti-Ice (EAI)

It is extremely important that ice cannot build up on the engine inlet cowl. Should ice form and then break off, it is ingested by the engine and could cause engine damage. Therefore, engine anti-ice is automatically turned on when the ice detection system begins to cycle. The EAI operates similarly to the stabilizer anti-ice system. Bleed air supplied from a high stage compressor bleed port is ducted to the leading edge of the engine inlet cowls. It exits the cowl through overboard vents. A pneumatically actuated EAI valve controls the flow of the warm bleed air to the inlet cowl. The valve is supplied for control pressure from an EAI controller. It regulates activation pressure to the EAI valve. (***Figure 13-10***) Signals from the ice detection system are delivered to the EAI logic control card along with duct pressure information from sensors downstream of the valve. The logic circuits control the operation of the EAI controller which positions the EAI valve. An air-to-air heat exchanger cools the bleed air used by the controller, the pressure regulating and shutoff valve and the high pressure and fan air controller.

EAI control is automatic when the anti-ice switch is set to AUTO. To operate the EAI on the ground, ON must be selected. Built In Test Equipment (BITE) is also active with the switch position ON or AUTO on the ground and in the air. EAI indications and warnings are displayed on the flight deck EICAS displays.

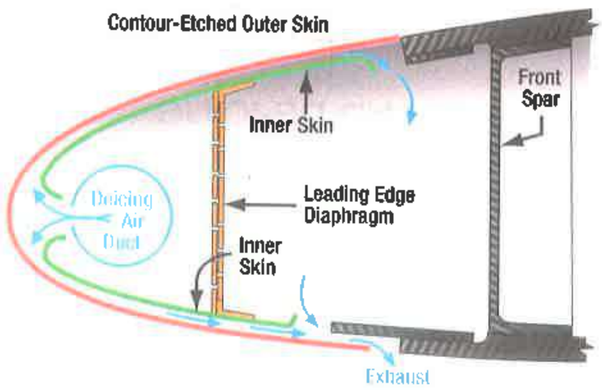


Figure 13-9. Thermal anti-ice systems.

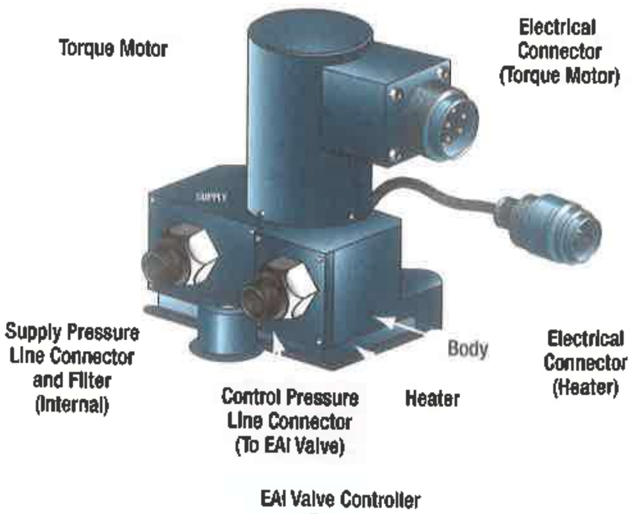


Figure 13-10. EAI valve controller.

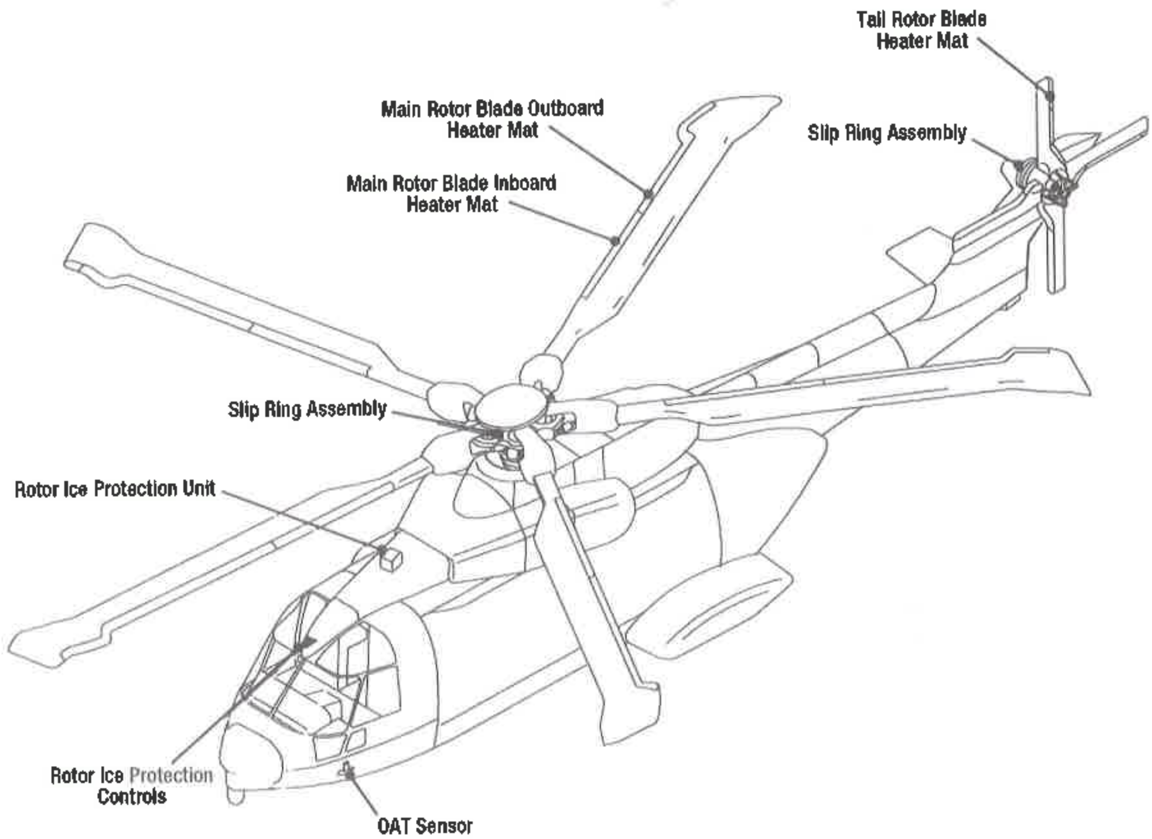


Figure 13-11. RIPS basic external components.

ELECTRICAL ANTI-ICE SYSTEMS

Due to its rotation, it is difficult to route hot air to provide anti-icing to the main and tail rotor blades and various mast components.

For the anti-icing and de-icing of these critical components, a Rotor Ice Protection System (RIPS) functions by electrically heating mats which cover approximately 20% of the leading edge chord along the entire length of the blade. The heat production is obtained from any available electrical source. Anti-frost mats are continuously supplied with electricity while the de-frost mats are heated intermittently. ***Figure 13-11*** shows the basic external components of a RIPS system. ***Figure 13-12*** shows some the electronic components making up a typical RIPS system. Electrical anti-ice systems should not be used on the ground, to avoid overheating and possible injury, except for short functional tests. (***Figure 13-13***)

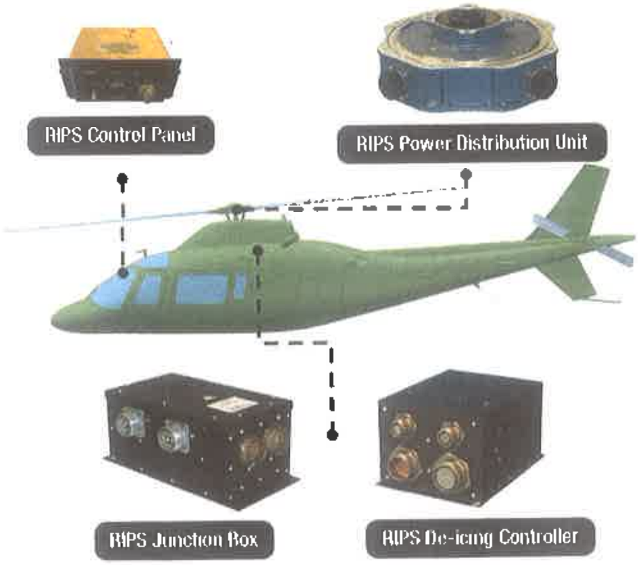


Figure 13-12. RIPS internal components.

MECHANICAL DE-ICING SYSTEMS

The accumulation of frost or ice considerably reduces the aerodynamic properties of the horizontal and vertical stabilizers by adhering to the leading edges of the profiles. On some helicopters, this accumulation is prevented by inflatable boots on the leading edges (and slats, if fitted) of the horizontal stabilizer. The boots are inflated for 6 seconds every 5 minutes by engine or APU compressor bleed air pressure which distorts the profiles and breaks the ice that has formed on them. In-between cycles, the air in the boot depressurizes to restore the leading edge profile. (***Figure 13-14***)

Figure 13-13. EAIS switches.

Figure 13-14. A pneumatic deicing boot on an airfoil, similar to a helicopter stabilizer.

CHEMICAL ANTI-ICE SYSTEMS

Chemical anti-icing is used in some aircraft to anti-ice the leading edges of the blades, stabilizers, and windshields. Ice protection is based upon the freezing point depressant concept. An antifreeze solution is mixed with the supercooled water in the cloud, depresses its freezing point, and allows the mixture to flow off the aircraft without freezing.

The system is designed to anti-ice, but it is also capable of de-icing an aircraft as well. When ice has accumulated on the leading edges, the antifreeze solution chemically breaks down the bond between the ice and airframe. This allows aerodynamic forces to carry the ice away.

Ground Chemical De-Icing

When aircraft surfaces are contaminated by frozen moisture, they must be de-iced prior to dispatch. When freezing precipitation exists and there is a risk of recontamination of the surface before takeoff, aircraft surfaces must additionally be anti-iced.

Ground de-icing of an aircraft is performed with spray equipment and often a boom truck to facilitate access. There are many formulas for de-icing fluid. An ethylene glycol or propylene glycol-based liquid is typical. It is mixed with hot water and sprayed on the aircraft being careful to avoid spraying into critical areas such as engine and APU inlets, probes and ports, air conditioning inlets and exits, and fuel tank vents.

De-Icing Fluids

- Type I de-ice fluid is an effective agent depending on the ratio of water to glycol, the temperature of the fluid when applied, as well as the ambient conditions. A thin layer of Type I fluid remains on the aircraft after the application which then acts as an anti-icing agent. However, as Type I fluid has low viscosity, and its anti-ice capacity lasts only a few minutes.
- Type II fluid is commonly used on large turbine powered aircraft. Type II fluid is a propylene glycol based fluid with molecular polymers added as a thickener. As such, when applied to de-ice the aircraft, Type II becomes a thicker coat adhering to the surfaces and protecting against new ice, snow, or from frost forming. As the aircraft airspeed increases during flight, the force of the air against the fluid layer decreases its viscosity and it is blown off.
- Type IV fluid is like Type II in that it has significant additives and leaves a thixotropic coating once applied. The anti-icing capabilities are greater for Type IV fluid which provide longer holdover times.

The coloring of these fluids is standardized. In general, straight glycol is colorless, Type I fluids are orange, Type II fluids are white/pale yellow, and Type IV fluids are green.

Holdover Time (HOT) is the estimated time that de-icing/anti-icing fluid prevents the formation of frost or ice and the accumulation of snow on the critical surfaces of an aircraft. HOT begins when the final application of de-icing and anti-icing fluid commences and expires when the de-icing and anti-icing fluid loses its effectiveness.

Guidelines for holdover times anticipated for SAE type IV fluid mixtures as function of weather conditions and OAT.
CAUTION: This table is for use in departure planning only, and it should be used in conjunction with pretakeoff check procedures.

OAT °C	OAT °F	SAE type IV fluid concentration neat fluid water (vol. %/vol.%)	Frost*	Freezing Fog	Snow◊	Freezing drizzle***	Light free rain	Rain on cold soaked wing	Other‡
			Approximate holdover times under various weather conditions (hours:minutes)						
Above 0	Above 32	100/ 0	18:00	1:05–2:15	0:35–1:05	0:40–1:10	0:25–0:40	0:10–0:50	CAUTION: No Holdover Time Guidelines Exist
		72/25	6:00	1:05–1:45	0:30–1:05	0:35–0:50	0:15–0:30	0:05–0:35	
		50/50	4:00	0:15–0:35	0:05–0:20	0:10–0:20	0:05–0:10	CAUTION: Clear Ice May Require Louch For Confirmation	
0 through −3	32 through 27	100/ 0	12:00	1:05–2:15	0:30–0:55	0:40–1:10	0:15–0:40		
		75/25	5:00	1:05–2:15	0:25–0:50	0:35–0:50	0:15–0:30		
		50/50	3:00	1:15–0:35	0:05–0:15	0:10–0:20	0:05–0:15		
below −3 through −14	below 27 through 7	100/ 0	12:00	0:20–0:50	0:20–0:40	**0:20–0:45	**0:10–0:25		
		75/25	5:00	0:25–0:50	0:15–0:25	**0:15–0:30	**0:10–0:20		
below −14 through −25	below 7 through −13	100/ 0	12:00	0:15–0:40	0:15–0:30				
below −25	below −13	100/ 0	SAE type IV fluid may be used below −25°C(−13 °F) if the freezing point of the fluid is at least 7°C(13°F) below the OAT and the aerodynamic acceptance criteria are met. Consider use of SAE type I when SAE type IV fluid cannot be used.						

°C = Degrees Celsius
°F = Degrees Fahrenheit
OAT= Outside Air Temperature
VOL = Volume

The responsiblity for the application of these data remains with the user:
* During conditions that apply to aircraft protection for ACTIVE FROST.
** No holdover time guidelines exist for this condition below −10°C (14°F).
*** Use light freezing rain holdover times if positive identification of freezing drizzle is not possible.
‡ Snow pellets, ice pellets, heavy snow, moderate and heavy freezing rain, hail.
◊ Snow includes snow grains.

CAUTIONS:
- The time of protection will be shortened in heavy weather conditions: heavy precipitation rates or high moisture contents.
- High wind velocity or jet blast may reduce holdover time below the lowest time stated in the range.
- Holdover time may be reduced when aircraft skin temperature is lower than OAT.

Figure 13-15. FAA Type IV de-ice holdover time guidelines.

Figure 13-15 shows a holdover timetable for Type IV fluid. HOT guidelines for other fluids are available and must be used when comparing options for different fluid use and options considering weather and traffic conditions.

De-Icing Procedures

All surfaces that have an aerodynamic, sensing, movement, or measuring function must be clean. These surfaces cannot necessarily be cleaned and protected in the same conventional de-icing/anti-icing manner. Some areas require only a cleaning operation, while others need protection against freezing. The procedure of de-icing may also vary according to aircraft limitations. The use of hot air may be required when de-icing (e.g., landing gear or blades). Some critical elements and procedures that are common for most aircraft are:

1. De-icing/anti-icing fluids must not be sprayed directly on wiring harnesses and electrical components (e.g., receptacles, junction boxes), onto brakes, wheels, exhausts, or thrust reverser's.
2. De-icing/anti-icing fluid must not be directed into the orifices of pitot heads, static ports, or directly onto airstream direction detectors probes/angle of attack airflow sensors.
3. All reasonable precautions must be taken to minimize fluid entry into engines, other intakes/ outlets, and control surface cavities.
4. Fluids must not be directed onto flight deck or cabin windows as this can cause crazing of acrylics or penetration of the window seals.
5. Any forward area from which fluid can blow back onto windscreens during taxi or subsequent takeoff shall be free of residues prior to departure.
6. All traces of the fluid on flight deck windows should be removed prior to departure, particular attention being paid to windows fitted with wipers.
7. Landing gear and wheel bays must be kept free from buildup of slush, ice, or accumulations of blown snow.
8. When removing ice, snow, slush, or frost from aircraft surfaces, care must be taken to prevent it entering and accumulating in auxiliary intakes or main rotor head hinge areas.

Probably the most difficult deposit to deal with is deep, wet snow when ambient temperatures are slightly above the freezing point. This type of deposit should be removed with a soft brush or squeegee. Use care to avoid damage to antennas, vents, vortex generators, etc., that may be concealed by the snow.

Light, dry snow in subzero temperatures should be blown off whenever possible; the use of hot air is not recommended, since this would melt the snow, which would then freeze and require further treatment. Moderate or heavy ice and residual snow deposits should be removed with a de-icing fluid.

When it becomes necessary to physically remove a layer of snow, all protrusions and vents should be examined for signs of damage. Flight control hinges should be moved to ascertain that they have full and free movement. The landing gear mechanism, doors and bay, and wheel brakes should be inspected for snow or ice deposits and the operation of uplocks and microswitches checked. Snow or ice can enter turbine engine intakes and freeze in the compressor. If the compressor cannot be turned by hand for this reason, hot air should be blown through the engine until the rotating parts are free. No attempt should be made to remove ice deposits or break an ice bond by force. After completion of de-icing operations, inspect the aircraft to ensure that its condition is satisfactory for flight, paying particular attention to pressure sensing ports for obstructions.

Frost deposits can be removed by placing the aircraft in a warm hangar or by using a frost remover or de-icing fluid. These fluids normally contain ethylene glycol and isopropyl alcohol and can be applied by spray or by hand. It should be applied within 2 hours of flight. De-icing fluids may adversely affect windows or the exterior finish of the aircraft, only the type of fluid recommended by the aircraft manufacturer should be used.

WINDSHIELD ANTI-ICING

Chemical anti-ice systems are also used on windshields. The liquid chemical is sprayed through a nozzle onto the outside of the windshield which prevents ice from forming. The chemical can also de-ice the windshield of ice that may have already formed. Systems such as these have a fluid reservoir, pump, control valve, filter, and relief valve. Other components may exist.

High performance aircraft windshields are typically made of laminated glass, polycarbonate, or similar ply materials. Typically, clear vinyl plies are also included to improve performance characteristics. The laminations create the strength and impact resistance of the windshield assembly.

This laminated construction facilitates the inclusion of electric heating elements into the glass layers which are used to keep the windshield clear of ice, frost, and fog. The elements can be in the form of resistance wires, or a transparent conductive material may be used as one of the window plies. To ensure enough heating is applied to the outside of the windshield, heating elements are placed on the inside of the outer glass ply. Windshields are bonded together by the application of pressure and heat without the use of cement. ***Figure 13-16*** illustrates the plies in an aircraft windshield.

Aircraft window heat systems have heat control units to supply power, and have feedback mechanisms, such as thermistors, to provide the heat control units with information to keep operating temperatures within acceptable limits. Most systems are automatic once switched on from the flight deck. Separate circuits for pilot and co-pilot are common to ensure visibility in case of a malfunction.

Consult the manufacturer maintenance information for details on the window heat system in question. Some windshield heating systems can be operated at two heat levels. On these aircraft, NORMAL heating supplies heat to the broadest area of windshield. HIGH heating supplies a higher intensity of heat to a smaller but more essential viewing area. This window heating system is always on and set in the NORMAL position. ***Figure 13-17*** illustrates a simplified windshield heat system of this type.

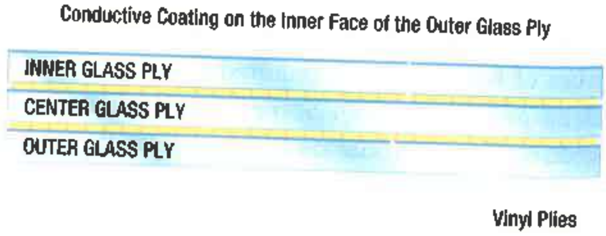


Figure 13-16. Cross-section of a transport category windshield.

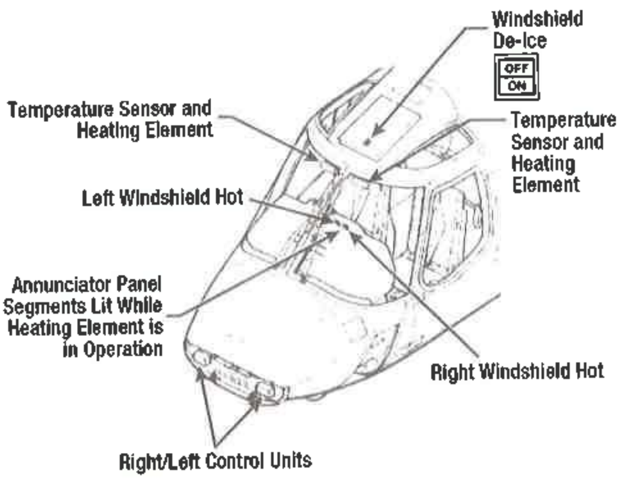


Figure 13-17. Heated windshield components.

Window heat anti-ice systems can generate a BITE test automatically. They are connected to an aircraft data bus for communication with the aircraft information management system.

Windshield Hot Air De-Ice/De-Fog

Some laminated windshields on older aircraft have a space between the plies that allows the flow of hot air to be directed between the glass to keep it warm and fog free. The source of air is bleed air or conditioned air from the environmental control system. Small aircraft may utilize ducted warm air which is released to flow over the windshield inner surface to defrost and defog. These systems are like those used in automobiles. The source of air could be ambient (de-fog only), the aircraft heating system or a combustion heater. While these pneumatic windshield heat systems are effective for de-icing or de-fogging the aircraft on which they are installed, they are not approved for flying into known icing conditions and as such, are not effective for anti-ice.

RAIN REPELLENT AND REMOVAL

There are several ways to remove rain from windshields. Most aircraft use one or a combination of chemical rain repellent, pneumatic rain removal (jet blast), hydrophobic sealants, and windshield wiper systems.

CHEMICAL RAIN REPELLENT

Water poured onto clean glass spreads out evenly. Even when the glass is held at a steep angle or subjected to air velocity, the glass remains wetted by a thin film of water. However, when glass is treated with certain chemicals, a transparent film is formed that causes the water to behave very much like mercury on glass. The water draws up into beads that cover only a portion of the glass, leaving the area between beads dry. The water is then readily removed from the glass. (***Figure 13-18***)

This principle lends itself quite naturally to removing rain from aircraft windshields. The high velocity slipstream continually removes the water beads, leaving a large part of the window dry. A rain repellent system permits application of the chemical repellent by a switch or push button in the cockpit. The proper amount of repellent is applied regardless of how long the switch is held. On some systems, a solenoid valve controlled by a time delay module meters the repellent to a nozzle which sprays it on the outside of the windshield. Separate systems exist for the forward glass of the pilot and copilot. (***Figure 13-19***)

In use, the repellent film slowly deteriorates with continuing rain, making periodic reapplication necessary. The length of time between applications depends upon rain intensity, the type of repellent used, and whether windshield wipers are used.

This system should only be used in very wet conditions. Rain repellent should not be used on dry windows because undiluted repellent restricts window visibility. Should the system be operated inadvertently, do not operate the windshield wipers or rain clearing system as that would tend to increase smearing. Also, the rain repellent residues caused by application in dry weather or light rain can cause staining or corrosion of the aircraft skin. To prevent this, any repellent residue should be removed with a thorough freshwater rinse at the earliest opportunity.

Figure 13-18. Rain repellent causes water to 'bead-up'.

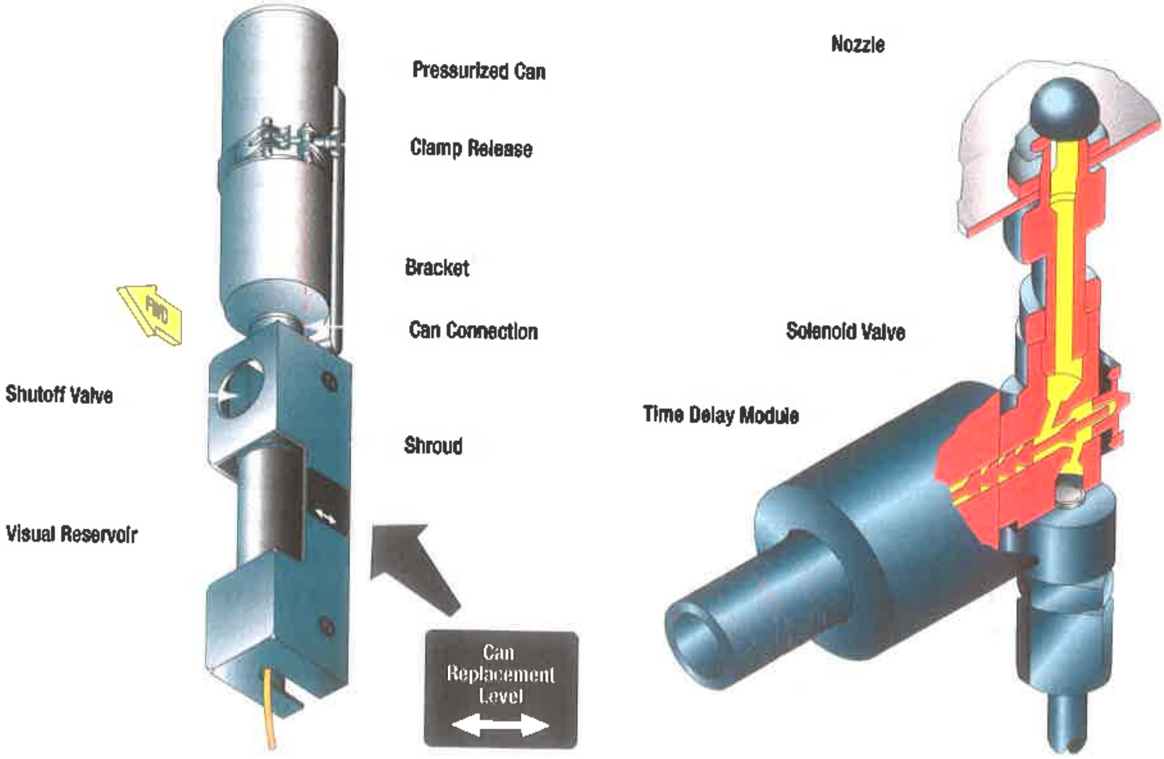


Figure 13-19. Typical rain repellent system.

HYDROPHOBIC COATINGS

Some helicopters use a surface seal called a hydrophobic coating that is applied on the outside of the pilot/copilot windshield. The word hydrophobic means to repel or not absorb water. These coatings cause raindrops to bead up and roll off, allowing the flight crew to see through the windshield with little distortion. Hydrophobic windshield coatings reduce the need for wipers and give the flight crew better visibility during heavy rain.

Most new aircraft windshields are treated with a surface seal coating. The manufacturer's coating process deeply penetrates the windshield surface providing hydrophobic action for a long time. When effectiveness declines, products made to be applied in the field are used. These liquid treatments rubbed onto the surface of the windshield maintain the beading action of rainwater.

PNEUMATIC RAIN REMOVAL SYSTEMS

The rain removal system shown in ***Figure 13-20*** controls windshield icing and removes rain by directing a flow of heated air over the windshield. This heated air serves two purposes. First, the air breaks the rain drops into small particles that are then blown away. Secondly, the air heats the windshield to prevent the moisture from freezing. The air can be supplied by an electric blower or by bleed air.

PROBE AND DRAIN HEATING

THERMAL ELECTRIC ANTI-ICING

Electricity is used to heat various components to prevent ice from forming, however because of their high amperage draw this is typically limited to small components. Effective thermal electric anti-ice is used mostly on air data probes such as pitot tubes, static ports, air temperature probes, ice detectors, and engine sensors. Inlet cowls may also be heated electrically to prevent ice from forming. A few aircraft use thermal electric anti-icing in windshields.

Probe Anti-Ice

In thermal electric anti-ice devices, current flows through a conductive element that produces heat. The temperature of the component is elevated above the freezing point so ice cannot form. Various schemes are used, such as an internal coil wire, externally wrapped blankets, or tapes, as well as conductive films and heated gaskets.

Data probes that protrude into the airstream are particularly susceptible to ice formation. Thus a device such as a pitot tube contains an internal electric element that is controlled by a switch in the cockpit. Simple heat circuits exist on most aircraft with a switch and

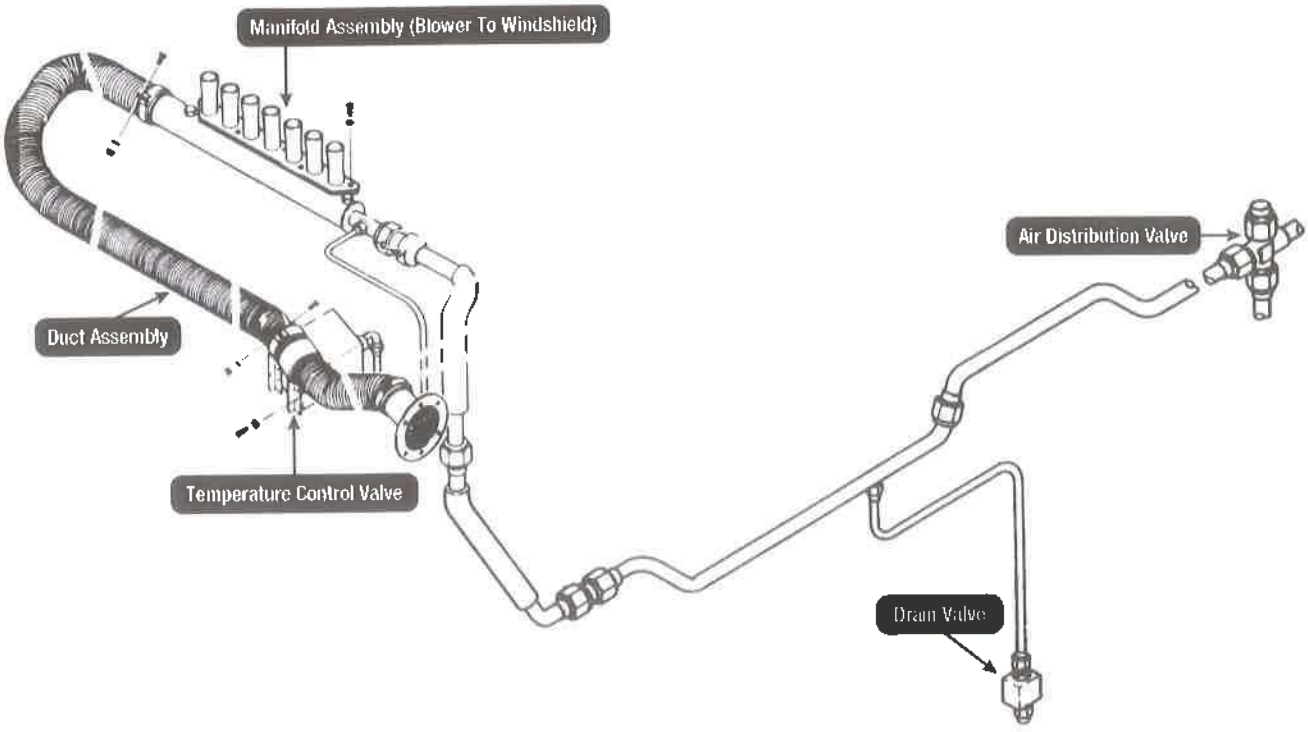


Figure 13-20. Pneumatic rain removal system.

a circuit breaker to activate and protect the device. Advanced aircraft may have more complex circuitry with computerized control in which the flight status condition is considered before thermal electric heaters are activated automatically.

NOTE: Use caution checking the function of the pitot heat on the ground. The tube gets extremely hot since it must keep ice from forming at altitude in very cold temperatures. An ammeter or load meter in the circuit can be used as a substitute to touch the probe, if so equipped.

Figure 13-21 shows such a circuit for a pitot tube. The primary flight computer supplies signals to energize relays to activate probe heat. Information concerning the speed of the aircraft, whether it is in the air or on the ground, and if the engines are running are factors considered by the system's logic. Similar controls are used for other probe heaters.

Drain Anti-Ice

On modern aircraft, the heating device and the thermostat controlling it are line replaceable units, and easily changed by the technician if they become inoperative. Drain mast electric heating elements are integral and require that the mast be replaced. Drain line heating elements are either flexible wrap type or integral. (***Figure 13-22***) Consult the manufacturer maintenance and parts manual for replacement information.

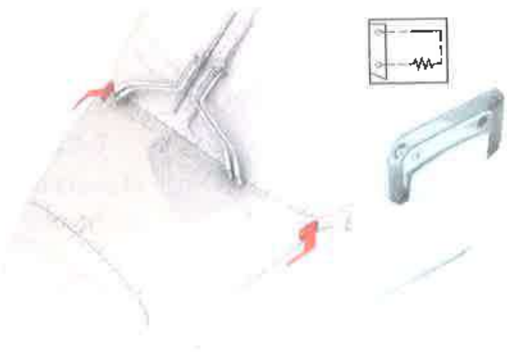


Figure 13-21. Circuit for a pitot probe heating system.

WIPER SYSTEMS

Aircraft windshield wipers characteristically have two basic problem areas. One is the tendency of the aerodynamic forces to reduce the wiper blade's loading pressure on the window, causing ineffective wiping or streaking. The other is in achieving fast enough wiper oscillation to keep up with impingement rates during heavy rain falls. As a result, many aircraft wiper systems fail to provide satisfactory vision clarity in heavy rain.

However when used, an electrical windshield wiper system uses blades which are driven by an electric motor(s). On some aircraft, the pilot and copilot wipers are operated by separate systems to ensure that clear vision is maintained through one side of the windscreen should the other system fail. Each windshield wiper assembly consists of a wiper, wiper arm, and a wiper motor/converter. Some older aircraft might be equipped with hydraulic wiper motors.

Maintenance performed on wiper systems consists of operational checks, adjustments, and troubleshooting. An operational check should be performed whenever a system component is replaced or whenever the system is suspected of not working properly. During the check, make sure that the windshield area covered by the wipers is free of foreign matter and is kept wet with water. Adjustment of a windshield wiper system consists of adjusting the wiper blade tension, the angle at which the blade sweeps across the windshield, and proper parking of the wiper blades.

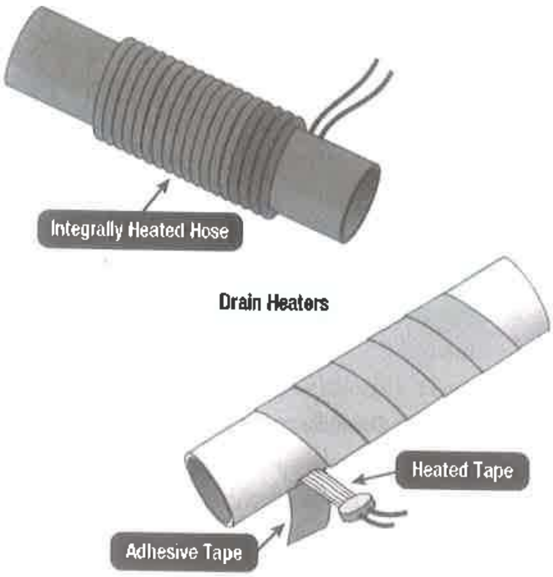


Figure 13-22. Drain line heating elements.

Question: 13-1
Name three ways in which the formation of ice can affect an airfoil such as a rotor blade.

Question: 13-2
What conditions cause the formation of rime ice?

Question: 13-3
Name the 4 events that occur when vibration repeatedly decreases on an ice detector.

Question: 13-4
What two health and safety concerns must always be observed when working near ice detection probes?

Question: 13-5
Name two areas on a helicopter in which compressor bleed air is commonly used for anti-icing.

Question: 13-6
What characteristic of de-icing fluids has the greatest effect on maximum hold over time?

Question: 13-7
Name two areas in which electrically heated tape is used for anti-icing.

Question: 13-8
What must you do if chemical rain repellent is accidentally spilled on a dry windshield?

ANSWERS

Answer: 13-1
Increased weight, out of balance weight, change in blade shape leading to reduction of lift.

Answer: 13-2
Near or below freezing temperatures; flight through clouds or light drizzle.

Answer: 13-3
The probe heater comes on, the rotor blade anti-ice system comes on, an annunciator alerts the crew.

Answer: 13-4
The probes get hot causing burns. Electrostatic precautions must be taken.

Answer: 13-5
Engine inlets; tail structure leading edges.

Answer: 13-6
Its viscosity.

Answer: 13-7
Various data probes and water drain tubes.

Answer: 13-8
Immediately and thoroughly wash it off with fresh water.

HELICOPTER AERODYNAMICS, STRUCTURES AND SYSTEMS

LANDING GEAR (ATA 32)

SUB-MODULE 14

PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY → B1.3 B1.4

Sub-Module 14

LANDING GEAR (ATA 32)

Knowledge Requirements

12.14 - Landing Gear (ATA 32)	3
Construction, shock absorbing; Extension and retraction systems: normal and emergency; Indications and warning; Wheels, Tires, brakes; Steering; Air-ground sensing; Skids, floats.	

12.14 - LANDING GEAR (ATA 32)

The functions of landing gear are to support a helicopter during ground maneuvers, dampen vibration, and absorb landing shocks. When required, it also performs the functions of steering and braking.

The landing gear are attached to the main structural elements of the aircraft. Its type depends on the design of the aircraft and its intended use. Most helicopter landing gear have wheels to facilitate operation to and from hard surfaces such as airport runways, but some smaller rotorcraft have skids. In fact, it is not necessary for a helicopter to move about on the ground because of its vertical take-off capability.

According to EASA regulations, taxiways for helicopters may be on the ground or in the air. "Ground Taxiway" means movement on the surface via its wheeled undercarriage. "Air taxiway" means a defined path above the surface established for the air taxiing of helicopters.

During the early days, the vast majority of helicopters were constructed with a wheeled undercarriage. This was due to the expectation that helicopters would mainly operate from airports. Today, commercial and military helicopters are on skids, wheels, or floats, enabling them to cope with whatever type of conditions they could meet in any environment around the world.

Regardless of the type of landing gear utilized, shock absorbing equipment, brakes, retraction mechanisms, controls, warning devices, cowling, fairings, and structural members necessary to attach the gear to the aircraft are considered parts of the landing gear system.

Note that throughout this manual and elsewhere, references to auxiliary landing gear refer to the nose gear or tail gear. The main landing gear is considered the gear located close to the aircraft center of gravity.

LANDING GEAR CONSTRUCTION AND SHOCK ABSORBING

Helicopters have two basic types of landing gear:

- Skids, which are almost always fixed.*
- Wheels, which can be fixed or retractable.

**The original Bell 209 (Cobra AH-1G) had retractable skids; these were replaced by fixed skids in the production variants.*

The engineering decision to use wheels or skids is a trade off. Skids are better suited for soft uneven ground like grass, snow, swamps, and rocky riverbeds; whereas wheels are better suited for airport and helipad operations. The retractable wheels allow better speed, handling, and the ability to ground taxi. However, they are more complex, require more maintenance, and increase the weight.

SKID GEARS

Skids are simple and light weight, so it is the best choice for small helicopters as weight is always a consideration. Also, they require very little maintenance, but the drawback is that ground handling is more difficult. In very small helicopters such as the Robinson, ground handling wheels can be attached to the skids so the helicopter can be moved around by a single person. Larger helicopters such as the Bell 206 or AS350 can be moved around with ground wheels, but it normally takes a couple of people. Another method of ground handling that is used is a platform dolly. The helicopter can land on this and is then tugged around. (***Figure 14-1***)

For operations on snow or soft fields, Bearpaws can also be added to the rear of the skids to help spread the weight of the helicopter. (***Figure 14-2***)

FLOATS

Floats are a variant of skids providing the added availability of landing on water. Floats can be fixed for everyday use or emergency deployable as pop-out floats.

Figure 14-1. Dolly.

Figure 14-2. Bearpaws.

WHEELS

On larger and more powerful twin engine helicopters weight is less of a concern and retractable wheels make sense. Wheels allow the helicopter to ground taxi (as opposed to hover taxi) around other aircraft and people without worrying about producing a strong downwash. Retracting the gear also reduces drag, allowing for a higher cruise speed. Attaching a tow bar to the nose wheel makes moving the helicopter around easier as well. The number and position of the wheels on each gear are variable (***Figure 14-3***) and can be classified as follows:

- Tail Wheel
- Tricycle Nose Wheel
- Quadricycle

Tail Wheel Type

The main gear is located forward of the center of gravity, and requires the support of an additional gear installed rear of the center of gravity. Typically the tail wheel is smaller and non-retractable. The advantage of this configuration is to reduce weight, eliminate the risk of leakage of long hydraulic hoses and reduced maintenance. (***Figure 14-4***)

Figure 14-3. Wheel positions.

Figure 14-4. A tail-wheel type landing gear.

Directional control is maintained by differential braking. A steerable rear wheel connected by cables to the rudder or rudder pedals is also common. Springs may be used for damping.

Tricycle Nose Wheel Type

Tricycle gear consists of one nose landing gear installed in the front and the main gear installed rear of the center of gravity. (***Figure 14-5***)

Tricycle type gear is used on large and small helicopters with the following advantages:

- Withstands heavy loads without deformation of the tail. This is mandatory on a very heavy helicopter.
- Provides better visibility from the cockpit, especially when landing and maneuvering on the ground.

In all cases, main gear, tail, or tricycle, each landing gear can be fitted with a single or a dual wheel system. (***Figure 14-6***)

Quadricycle Type

A quadricycle has two landing gears installed in the front and two in the rear of the center of gravity of the aircraft. (***Figure 14-7***)

Figure 14-5. A tricycle nose-wheel type landing gear.

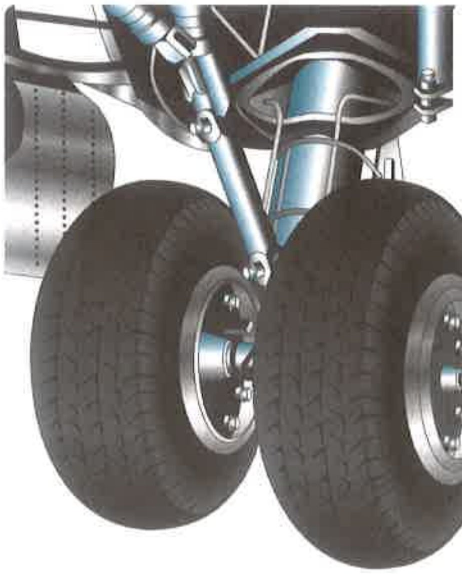


Figure 14-6. Dual main gear of a tricycle-type landing gear.

CLASSIFICATION OF LANDING GEAR

Depending on the geometry, landing gear can be broadly divided as in ***Figure 14-8***:

- Non-articulated or telescopic type.
- Articulated or levered suspension type.
- Semi articulated or semi levered suspension type.

Figure 14-7. Quadricycle type landing gear.

Non-Articulated or Telescopic Type

In a telescopic landing gear the ratio of the wheel axle travel to the shock strut stroke is identical. If there is an introduction of a mechanical advantage to the shock strut, the landing gear can be termed as an articulated gear.

Articulated or Levered Suspension Type

In an articulated gear the shock strut is hinged at both ends and results in axial only loads. The articulated gear geometry can be further subdivided into two types based

TYPES OF LANDING GEAR GEOMETRIES

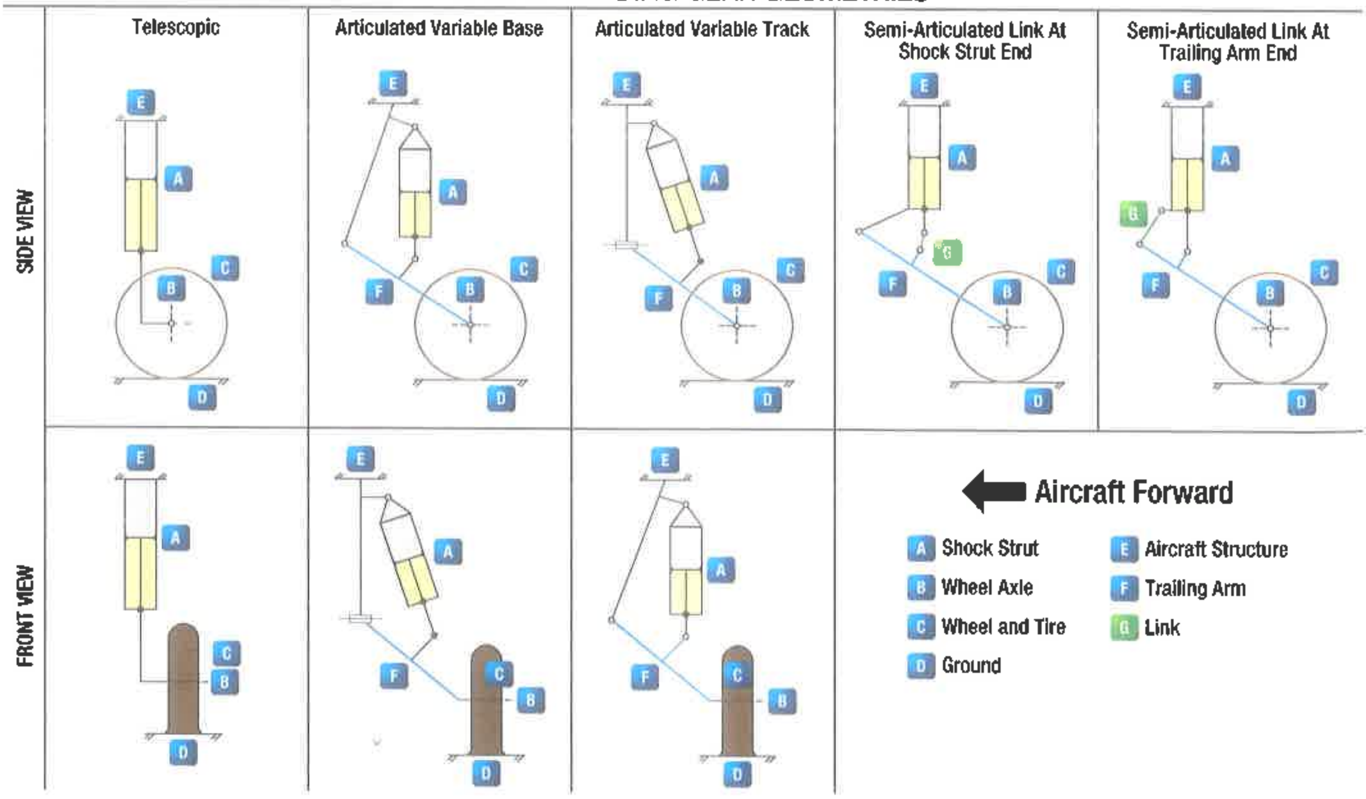


Figure 14-8. Landing gear geometries.

on the orientation of the four-bar mechanism type:

- Variable base type, which changes the wheelbase of aircraft while telescoping.
- Variable track type, which changes the wheel track of aircraft while telescoping.

Four-Bar Linkage Mechanism

A four-bar linkage is the simplest movable closed chain linkage consisting of four bars (or links), connected in a loop. It is used to lower and retract the gear, by using three members connected by pivots and a fourth being linked to the aircraft structure, as shown in ***Figure 14-9A***. Most mechanisms for landing gear retraction systems are based upon a four-bar linkage.

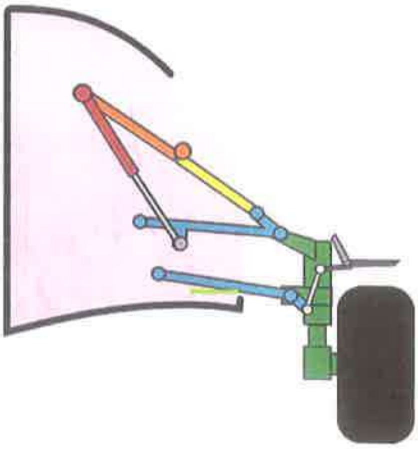


Figure 14-9A. Four-bar linkage.

Wheelbase

Wheelbase is the distance of the operating line through the wheel axles of the main landing gear to the center point of the nose or tail gear.

Wheel Track

Wheel track is the distance between the far right and the far left main wheels. In the case of tandem wheels (two wheels on each leg), track is the distance to the center of each leg. (***Figure 14-9B***)

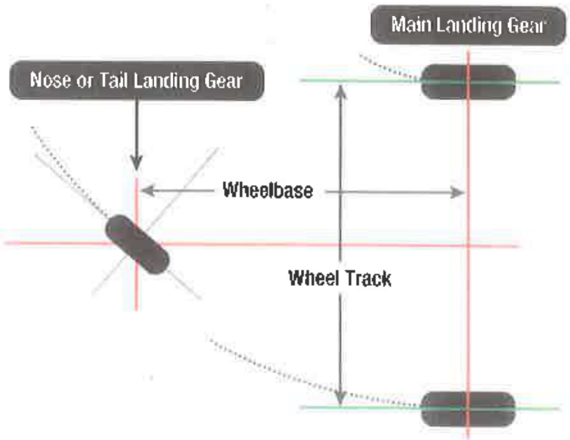


Figure 14-9B. Wheel base and track.

Semi-Articulated or Semi-Levered Suspension Type

In semi-articulated gears, the shock strut is fixed and a link rod is introduced to complete the four bar mechanism. In this case lateral load on the shock shut is not absent but can be reduced to a greater extent. Semi-articulated gear can also be further divided based on the position of the link rod in the four-bar mechanism at the shock strut end or the trailing arm end.

FIXED AND RETRACTABLE LANDING GEAR

Most lightweight and slower helicopters have fixed gear. (***Figure 14-10***) This means that the gear remains exposed as the aircraft flies. As speed increases so does parasitic drag and fixed gear become more problematic.

Figure 14-10. Fixed landing gears.

With higher performance helicopters, drag becomes progressively more important, and so the landing gear is retracted into the fuselage (or in the tail boom for a tail gear) during flight. There are, however, penalties of increased weight, greater complication and additional maintenance. Once retracted, it is out of the rotor wake and does not cause parasitic drag. (***Figure 14-11***)

Some retractable gear have tight fitting panels (or gear doors) that conform to the skin of the aircraft when the equipment is fully retracted. (***Figure 14-12***) Others have separate doors that open, allowing the landing gear to extend or retract and then close. In nearly all helicopters

Figure 14-11. Retractable landing gears.

Figure 14-12. Fitting panels on retractable landing gears.

equipped with retractable landing gear, a system is provided for emergency gear extension in the event that the gear mechanisms fail to lower the gear.

LANDING GEAR ATTACHMENT

Helicopter landing gear are attached to structural members which are designed for the specific purpose of supporting the gear and the weights and stresses they support. Retractable gear must also be engineered to move into a recess or well. A trunnion arrangement is typical. A trunnion is a fixed structural extension of the gear's upper strut cylinder with bearing surfaces that allow the entire gear assembly to move. It is attached to the structure in such a way that the gear can pivot from the vertical position for landing and taxi to the stowed position during flight. (***Figure 14-13***)

While in the gear down position, the trunnion is free to swing or pivot. Alone, it cannot support the aircraft without collapsing. Thus a drag brace is used to restrain against the pivot action built into the trunnion attachment. The upper end of the two piece drag brace is attached to the aircraft structure and the lower end to the strut. A hinge near the middle allows the brace to fold and permits the gear to retract. For ground operation, the drag brace is straightened over center to a locked into position, so the gear remains rigid. (***Figure 14-14***)

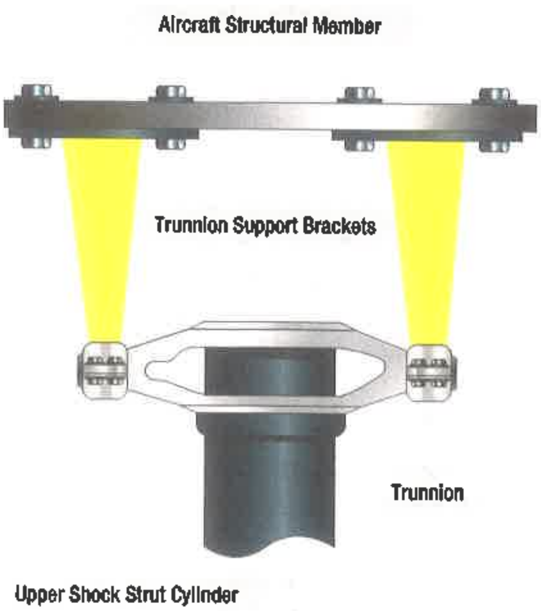


Figure 14-13. The trunnion.

Figure 14-14. A hinged drag strut.

The function of a drag brace on some aircraft is performed by the hydraulic cylinder used to raise and lower the gear. Hydraulic cylinder internal locks replace the over center action of the brace for support during ground maneuvers.

LANDING GEAR ELEMENTS

A typical landing gear consists of the following components:

- A barrel, housing an oleo-pneumatic shock strut that attaches the landing gear to the airframe with an attachment fork.
- Alignment units such as pistons and axles.
- Wheel and brake assemblies, and tires.
- Toggle links, drag stays.
- Retraction/extension actuators and safety devices such as Up/Down locks.

In addition to the above it may also consist of:

- Nose wheel steering jack.
- Jacking, mooring and towing points.
- Earthing cable.
- Micro switches such as a weight-on-wheel switch.
- Electrical and hydraulic routings.

Some landing gear systems may also include:

- A retraction/extension selector valve.
- A steering selector valve.
- Accumulator and cockpit indicator controls.
- A circuit for sequencing.
- Hydraulic and electrical routings etc.

Main Landing Gear Elements

A main landing gear is illustrated in ***Figure 14-15*** with many of the parts identified.

Shock Strut Assembly

The shock strut assembly, one each for the left and right side, is an air/oil shock absorber that consists of a cylinder, and upper and lower pistons. The shock strut assembly absorbs or dampens the vertical forces induced by landing. The upper shock strut fitting is attached to the fuselage pivot mount. The lower shock strut fitting is attached to the center mount of the trailing arm. The cylinder contains the upper and lower pistons. For servicing the upper cylinder, a hydraulic fill port and bleed port are located on the external side. The upper piston has a nitrogen fill port for servicing. A spherical bearing is installed in the upper end of the strut that serves as an attachment point for the pivot mount. The lower piston has a drain plug for servicing. A spherical bearing is located at the lower end of the strut and serves as a pivot attachment point and a kneeling coupling for maintenance or shipping of the helicopter.

Weight-On-Wheels Switch

Certain systems of some helicopters require varying electrical control depending upon the condition of the helicopter (airborne or on the ground). This condition is sensed by the landing gear weight-on-wheels sensors (also known as ground or squat switches). These sensors transmit signals to circuits to provide air or ground control of the required component.

Nose/Tail Landing Gear Elements

Most modern helicopters fitted with wheels are the nose wheel (tricycle) design. (***Figure 14-16***) Some older designs have tail wheels. (***Figure 14-17***) The nose/tail wheel is sometimes dampened with a shock absorber or friction ring to prevent shimmy. A torque link can be fitted to maintain correct nose/tail wheel alignment.

Most helicopters have steering nose or tail wheels. When on the ground this steering linkage is actuated between the anti-torque pedals in the flight deck which then disconnects automatically once the helicopter leaves the ground.

The nose/tail gear shock strut works on the same principle as the shock struts of the main gear. In some cases, the long metering pin which closes the calibrated opening during retraction is missing. In these nose gears, the hole is not closed until the last movement of retraction and so does not influence the principle of shock absorption. Many nose/tail gears are also steerable when the gear's shock absorber is compressed on the ground.

Shock Strut Assembly

The shock strut assembly is a steel cylinder fitted with an upper and lower piston assembly with pivot points which attach to the tail boom mount at one end and a trailing arm mount on the other. A hydraulic service valve is provided at the center of the upper piston of the cylinder. A nitrogen charge valve is provided for servicing of the upper piston.

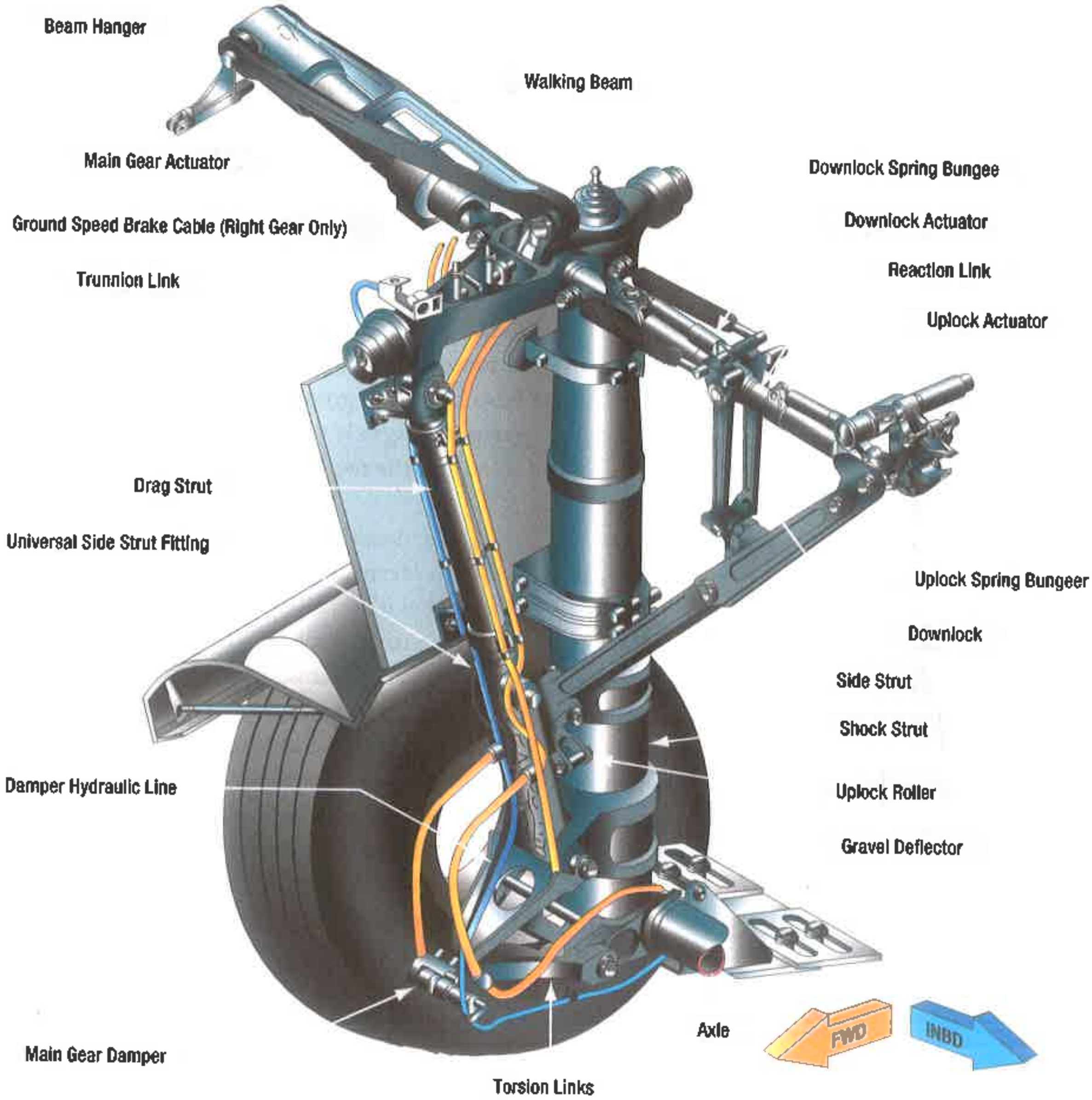


Figure 14-15. Nomenclature of a main landing gear bogie truck.

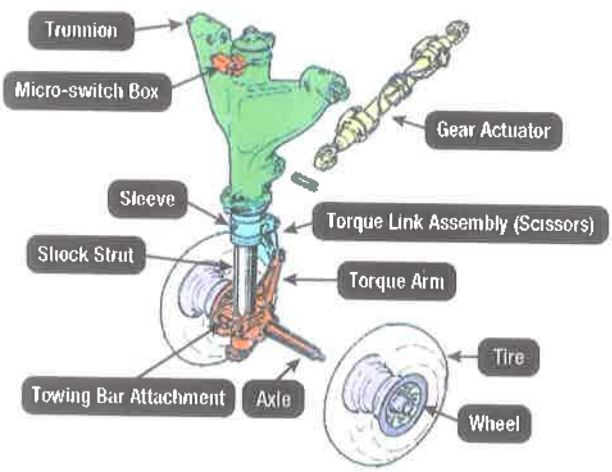


Figure 14-16. Nose landing gear elements.

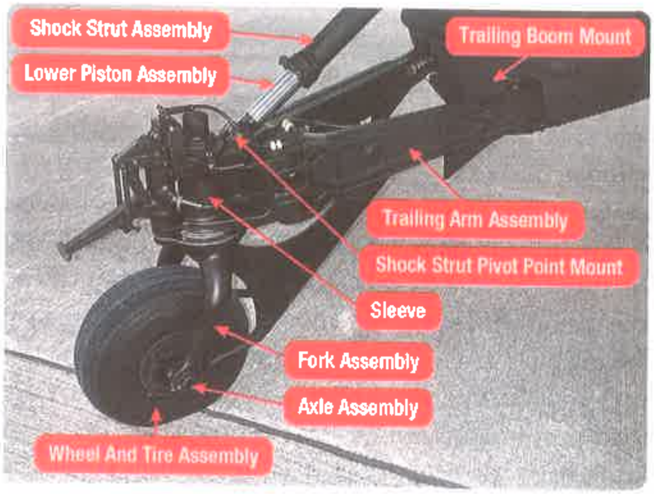


Figure 14-17. Tail landing gear elements.

Trailing Arm Assembly

The trailing arm assembly provides support for the tail wheel and its components. Two trailing arm attachments are provided for mounting of the arm assembly. The aft end of the trailing arm forms an integral housing and socket to provide mounting for the fork assembly. A jack pad may be mounted on the lower surface of the housing forward of the tail wheel.

Fork and Axle Assembly

The fork and axle assembly is mounted by inserting the lower cam of the fork in a trailing arm socket. The center pivot shaft of the fork is hollow and provides mounting for the wheel's centering mechanism. A wear plate is mounted on the fork assembly and is drilled with a single locking hole for the actuator lock pin to make contact until fork centering is accomplished. A static ground cable may be attached to the bottom of the fork.

Centering Cams (Locating Cams)

Because the nose/tail wheel can be steered, it is necessary to be able to center the inner strut in relation to the outer strut when the helicopter leaves the ground. Thus each strut has a centering cam. (***Figure 14-18 and Figure 14-19***) Centering means that the inner and the outer strut are lined up in relation to each other. The wheels first return to the center position to prevent damage to the fuselage and gear when they are retracted into the wheel well. If the wheels are not centered, the landing gear cannot be retracted.

An upper cam is free to mate into a lower cam recess when the gear is fully extended. This aligns the gear for retraction. When weight returns to the wheels after landing, the shock strut is compressed, and the centering cams separate allowing the lower strut (piston) to rotate in the upper strut cylinder. This rotation is controlled to steer the aircraft.

As the strut is folded into the wheel well during retraction, the roller or guide pin engages a ramp or track mounted to the wheel well structure. The ramp/track guides the roller or pin in such a manner that the nose wheel is straightened as it enters the well.

Landing Gear Material

Due to the intense levels of pressure they encounter, every component of the landing gear must be made of high quality materials such as steel, aluminum, and titanium alloys. Steel components include pistons, braces, links, switch brackets, plugs, axles, shafts, springs, clamps, sleeves, arms, etc. The main landing gear structure is often made from titanium. Most modern brake discs are carbon composite as carbons are superior to steel at resisting the high temperatures in braking systems.

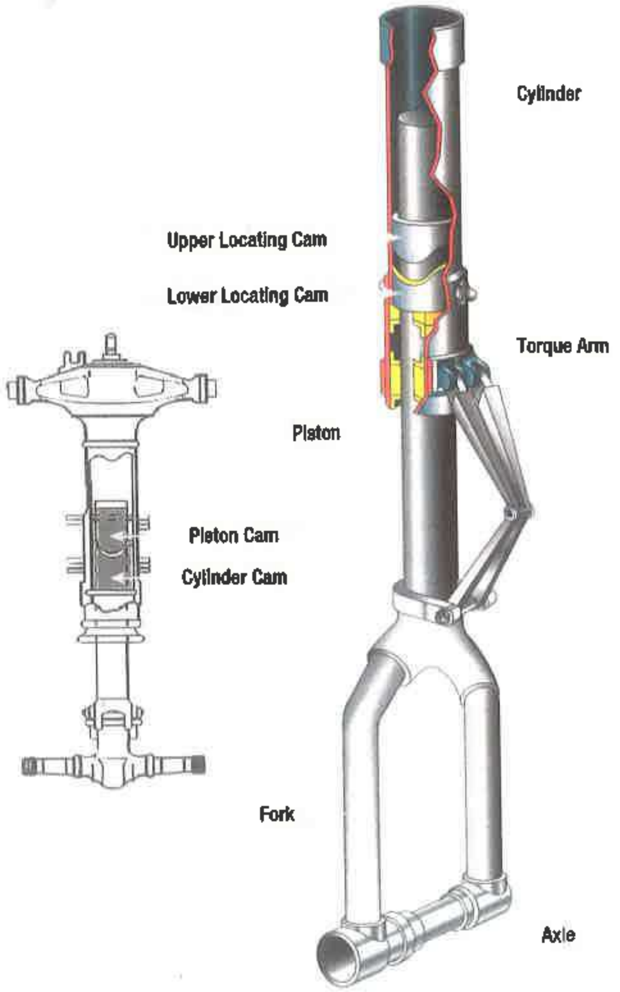


Figure 14-18. Centering cams.

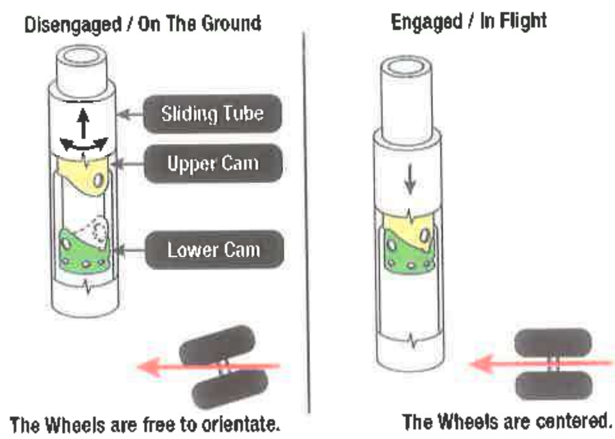


Figure 14-19. Centering cam details.

Corrosion Protection

Corrosion protection is important for the landing gear components as they are easily susceptible to environmental attack. Apart from normal electrolytic finishes like cadmium plating, epoxy or polyurethane primer and polyurethane topcoats are applied on the exposed landing gear parts. Use of non-metallic corrosion resistant materials is also increasing.

SHOCK ABSORBING

The forces of impact on an aircraft during landing must be controlled by the landing gear. This is done in two ways:

1. The shock energy is transferred throughout the airframe at a different rate and time than a single strong pulse of impact.
2. The shock is absorbed by converting its energy into heat which is then dissipated to the surroundings via a shock strut.

Shock Struts

Shock struts, (often called oleos or air/oil struts), use a combination of nitrogen (or compressed air) and hydraulic fluid to absorb and dissipate the loads of landing. This is the most common method of landing shock dissipation on aircraft of all sizes. Shock struts are self contained hydraulic units that support an aircraft while on the ground and protect the structure during landing. They must be inspected and serviced regularly to ensure proper operation. There are many different designs of shock struts, but most operate in a similar manner. The following is general in nature. For information on the construction, operation, and servicing of a specific aircraft shock, consult the manufacturer's maintenance instructions.

Shock Strut Construction

A shock strut is constructed of two telescoping cylinders or tubes that are closed on the external ends. The upper cylinder is fixed to the aircraft and does not move. The lower cylinder is called the piston and is free to slide in and out of the upper cylinder. Two chambers are formed. The lower chamber is always filled with hydraulic fluid and the upper chamber is filled with compressed air or nitrogen. An orifice located between the cylinders provides a passage for the fluid from the bottom to the top cylinder chamber when the strut is compressed. (***Figure 14-20***)

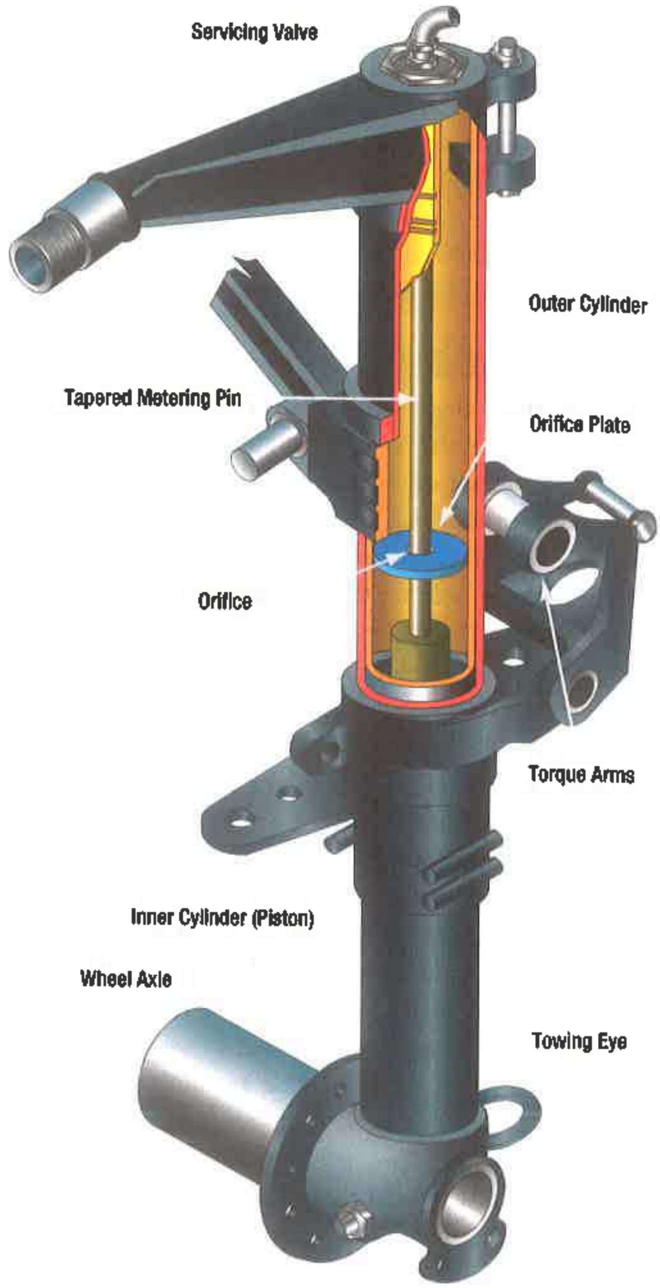


Figure 14-20. A shock strut.

Metering Pins

Most shock struts employ a metering pin as in ***Figure 14-21*** for controlling the rate of fluid flow from the lower chamber into the upper chamber. During the compression stroke, the rate of fluid flow is not constant. It is automatically controlled by the taper of the metering pin in the orifice. When a narrow portion of the pin is in the orifice, more fluid can pass to the upper chamber. As the diameter of the portion of the metering pin in the orifice increases, less fluid passes. Pressure build up caused by strut compression and the hydraulic fluid being forced through the metered orifice causes heat which is dissipated through the structure of the strut.

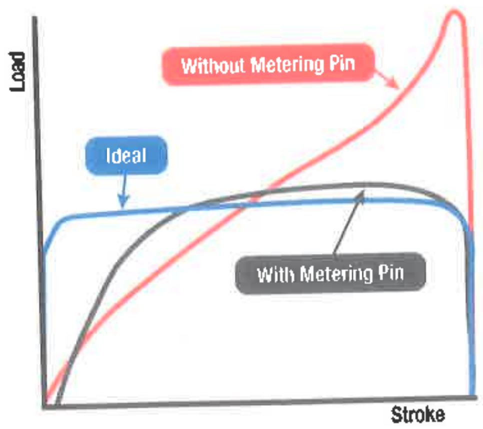


Figure 14-21. Metering pin.

Tests have shown that without the metering pin, the energy absorption characteristic would be very different, with the majority of the load being absorbed towards the end of the stroke (and an uncomfortable landing). However, a carefully shaped metering pin, enabling a variation in the amount of the flow will modify the energy absorption characteristic.

On some types of shock struts, a metering tube is used. The operational concept is the same as that with metering pins except the holes in the metering tube control the flow of fluid from the bottom chamber to the top chamber during compression. (***Figure 14-22***)

Damping Or Snubbing Devices

Upon lift off or a rebound from compression the shock strut tends to extend rapidly. This could result in a sharp impact at the end of the stroke and damage to the strut. It is typical for shock struts to be equipped with a damping or snubbing device to prevent this. A recoil valve on the piston or a recoil tube restricts the flow of fluid during the extension stroke, slowing the motion and prevents damaging impact forces.

Axles

Most shock struts are equipped with an axle as part of the lower cylinder to provide installation of the wheels. Shock struts without an integral axle have provisions on the end of the lower cylinder for installation of the axle assembly. (***Figure 14-23***)

Valve Fitting Assembly

The upper cylinder of a shock strut typically contains a valve fitting assembly located at or near the top of the cylinder. The valve provides a means of filling the strut

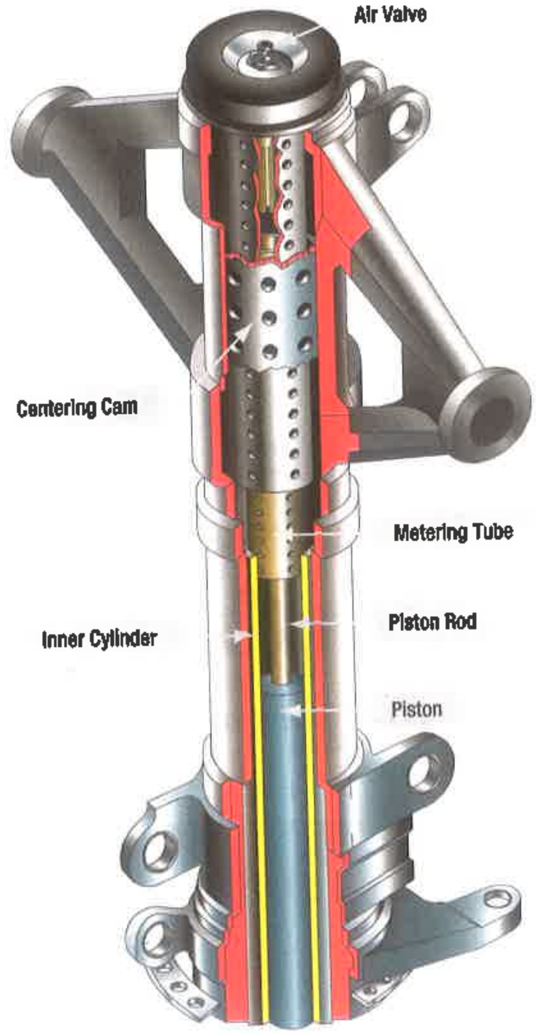


Figure 14-22. Some landing gear shock struts use an internal metering tube.

with hydraulic fluid and inflating it with air or nitrogen. A packing gland seals the sliding joint between the upper and lower cylinders and is installed in the open end of the outer cylinder. A wiper ring is also installed in a groove in the lower bearing or gland nut on most struts. It is designed to keep the sliding surface of the piston from carrying dirt or ice into the gland and upper cylinder. Regular cleaning of the exposed portion of the piston helps the wiper do its job and decreases the possibility of damage to the gland which could cause the strut to leak.

Torque Links

To keep the piston and wheels aligned, most shock struts are equipped with torque links. One end of the link is attached to the fixed upper cylinder. The other end is attached to the lower cylinder (piston) and so cannot rotate. This keeps the wheels aligned. The links

Figure 14-23. Axles machined out of the same material.

Figure 14-24. Torque links.

also retain the piston in the end of the upper cylinder when the strut is extended, such as after takeoff. (***Figure 14-24***)


Figure 14-25. Tail bumper assembly.

Nose Gear Shock Struts
As described before, nose gear shock struts are provided with a locating cam to keep the gear aligned. A cam protrusion is attached to the lower cylinder, and a mating lower cam recess is attached to the upper cylinder. These cams line up the wheel and axle in the straight ahead position when the strut is fully extended, allowing the nose wheel to enter the wheel well when the gear is retracted.

Nose gear struts are also often equipped with a disconnect pin to enable quick turning of the aircraft while towing or for positioning it on the ramp or in a hangar. Disengaging this pin allows the wheel fork to rotate up to 360° to turn in a tight radius. At no time should the nose wheel of any aircraft be rotated beyond the limit lines marked on the airframe.

Tail Bumper Assembly
Some heavy helicopters are equipped with a tail bumper which deforms during a hard landing in the nose-up position. Steel spars mounted in a pendulum on 2 fittings, are equipped with a hydraulic shock absorber acting as a strut capable of absorbing high speed falls in a nose-up landing. (***Figure 14-25***)

Jacking Points And Towing Lugs
Nose and main gear shock struts on many aircraft are equipped with jacking points and towing lugs. Jacks should always be placed under the prescribed points. When towing lugs are provided, the tow bar should be attached only to these lugs. (***Figure 14-26***)

Shock Strut Instruction Plate
An instruction plate is attached to the shock struts giving directions for filling the strut with fluid and/or gas. (***Figure 14-27***) It is usually attached near the filler

Figure 14-26. A towing lug on a landing gear.

Figure 14-27. Instruction plate and Nitrogen filling port.

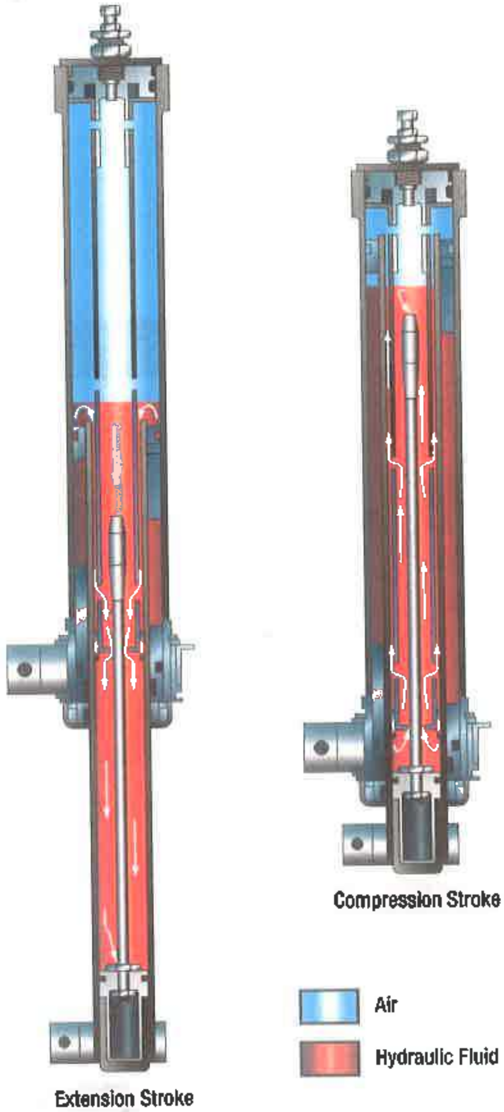


Figure 14-28. Fluid flow.

inlet and air valve assembly. It specifies the correct type of hydraulic fluid and the pressure to which it should be inflated. It is of utmost importance to become familiar with these instructions prior to filling a strut with hydraulic fluid or inflating it with air or nitrogen.

Shock Strut Operation

Figure 14-28 illustrates the inner construction of a shock strut. Arrows show the movement of the fluid during compression and extension of the strut. The compression stroke of the strut begins as the aircraft wheels touch the ground. As the center of mass of the aircraft moves downward the strut compresses and the lower piston is forced upward into the upper cylinder. The metering pin is therefore moved up through the orifice. The taper of the pin controls the rate of fluid flow from the bottom to the top cylinder at all points during the compression stroke. In this manner, the greatest amount of heat is dissipated through the walls of the strut. At the end of the stroke, the compressed air in the upper cylinder is further compressed which limits the compression stroke of the strut with minimal impact. During taxi operations, the air in the tires and the strut combine to smooth out bumps.

Insufficient fluid or air in the strut causes the compression stroke to not be properly limited. The strut could bottom out resulting in impact forces to be transferred directly to the airframe. In a properly serviced strut, the extension stroke occurs at the end of the compression stroke. Energy stored in the compressed air in the upper cylinder causes the aircraft to start moving upward in relation to the ground as the strut tries to rebound to its normal position. Fluid is forced back down into the lower cylinder through restrictions and snubbing orifices. The snubbing of fluid flow during the extension stroke dampens the rebound and reduces oscillation caused by the spring action of the compressed air. A sleeve, spacer, or bumper ring incorporated into the strut limits the extension stroke. (***Figure 14-29***)

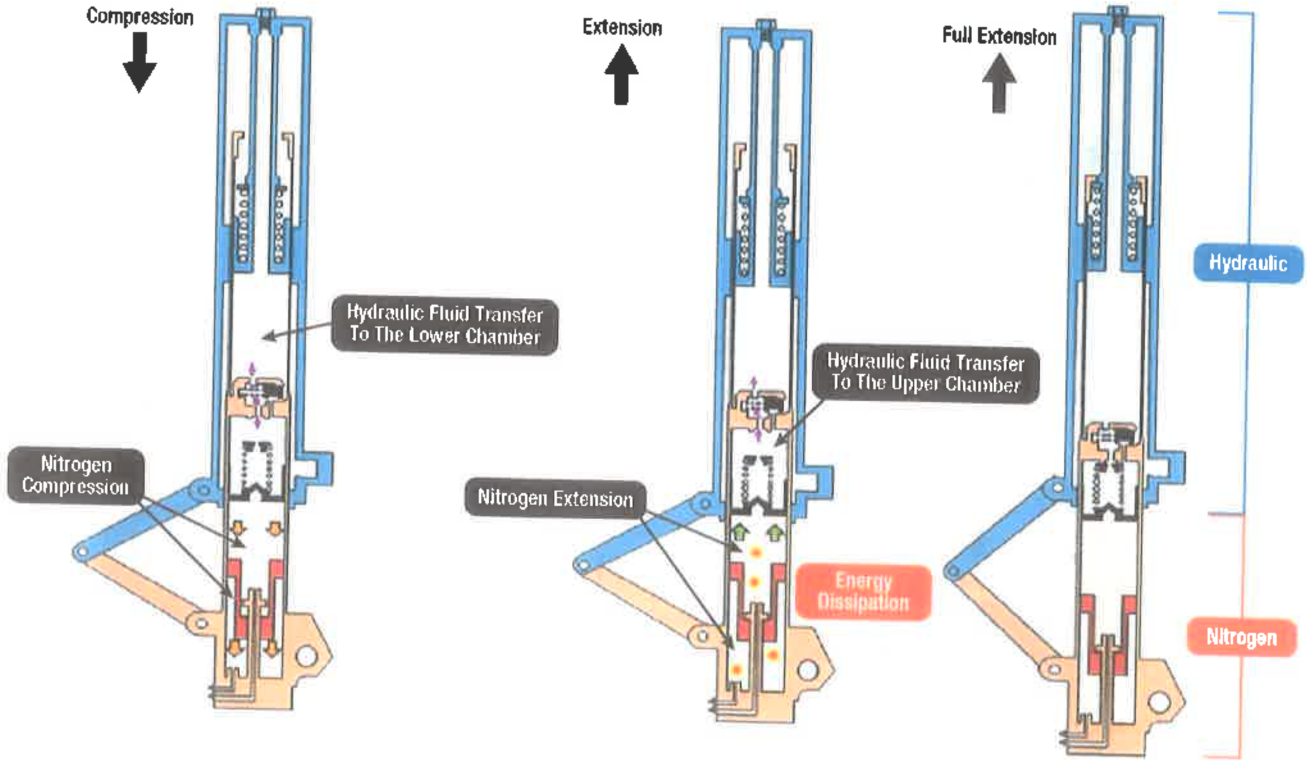


Figure 14-29. Compression-Extension phases of a shock strut.

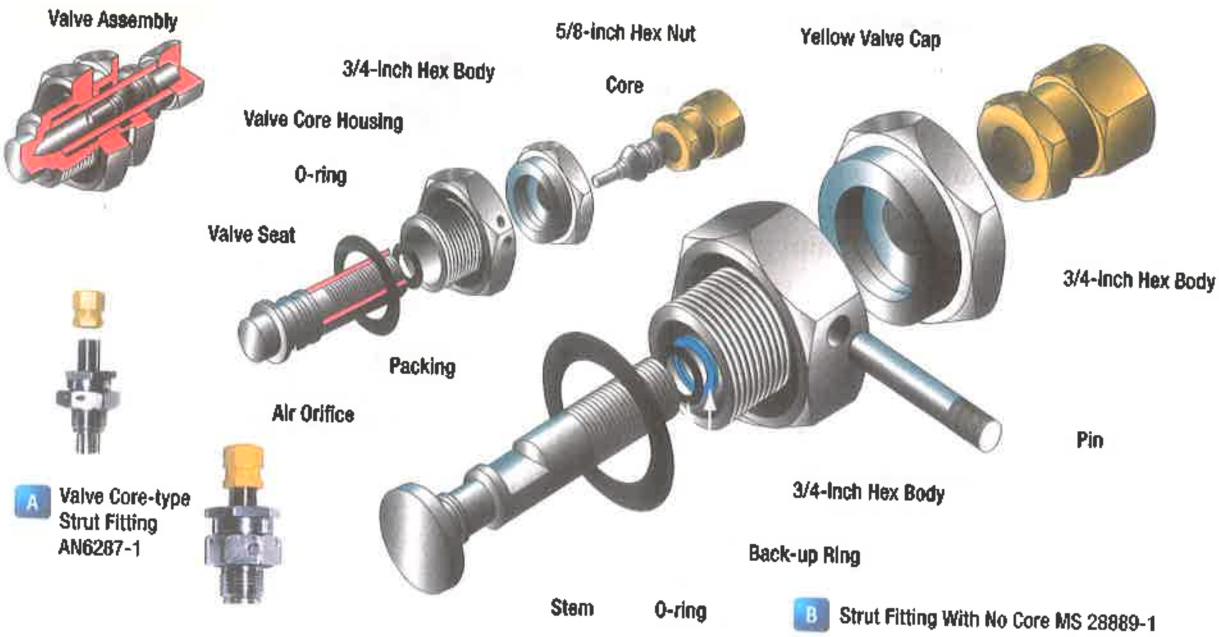


Figure 14-30. Valve core-type (A) and core-free valve fittings (B).

To check the fluid level, most struts need to be deflated and compressed into the fully compressed position. Deflating a strut can be a dangerous operation. The technician must be thoroughly familiar with the operation of the high pressure valve at the top of the strut. Refer to the manufacturer's instructions for proper deflating technique and follow all safety precautions.

Servicing Valves

Two common types of high pressure strut servicing valves are illustrated in ***Figure 14-30***. The AN6287-1 valve in ***Figure 14-30A*** has a valve core assembly and is rated to 3 000 pounds per square inch (psi). However, the core itself is only rated to 2 000 psi.

The MS28889-1 valve in ***Figure 14-30B*** has no valve core. It is rated to 5 000 psi. The swivel nut on the AN6287-1 valve is smaller than the valve body hex. The MS28889-1 swivel nut is the same size as the valve body hex. The swivel nuts on both valves engage threads on an internal stem that loosens or draws tight the valve stem to a metal seat.

Servicing Shock Struts

Typical deflating shock struts procedures used are:

1. Position the aircraft so that the struts are in the normal ground operating position. Make certain that personnel, work stands, and other obstacles are clear of the aircraft. If the procedures require, securely jack the aircraft.
2. Remove the cap from the air servicing valve.
3. Check the swivel nut for tightness.
4. If the servicing valve is equipped with a valve core, depress it to release any air pressure that may be trapped under the core in the valve body. Always stand to the side of the trajectory of any valve core in case it releases.
5. Loosen the swivel nut. For a valve with a core, rotate the swivel nut one turn counterclockwise. Using a tool designed for the purpose, depress the valve core to release the air in the strut. For a valve without a core, rotate the swivel nut sufficiently to allow the air to escape.
6. When all air has escaped from the strut, it should be compressed completely. Aircraft on jacks may need to have the lower strut lifted with an exerciser jack to achieve full compression of the strut. (***Figure 14-31***)
7. Remove the core of an AN6287 valve using a valve core removal tool. (***Figure 14-32***) Then, remove the entire valve by unscrewing its body from the strut.
8. Fill the strut with hydraulic fluid to the level of the valve port with the approved hydraulic fluid.
9. Reinstall the air valve assembly using a new O-ring packing. Torque to applicable manufacturer's specifications. If an AN2687 valve, install a new valve core.
10. Inflate the strut. A threaded fitting from a controlled source of high pressure air or nitrogen should be screwed onto the valve. Control the flow with the valve swivel nut. The correct amount of inflation is measured in psi on some struts. Other manufacturers specify struts to be inflated until extension of the lower strut is a certain measurement. Shock struts should always be inflated slowly to avoid excess heating and over inflation.
11. Once inflated, tighten the swivel nut and torque as specified.
12. Remove the fill hose fitting and finger tighten the valve cap of the valve.

Bleeding Shock Struts

It may be necessary to bleed a shock strut during the service operation or when air becomes trapped in the hydraulic fluid inside the strut. This can be caused by low hydraulic fluid quantity in the strut. Bleeding is normally done with the aircraft on jacks to facilitate repeated extension and compression of the strut to expel

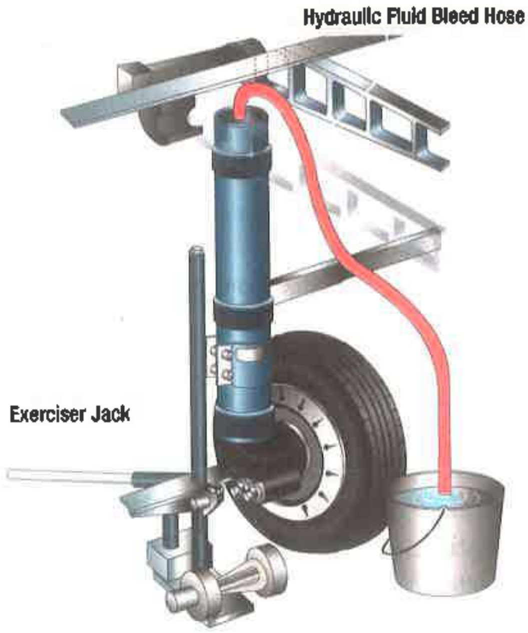


Figure 14-31. Air trapped in shock strut hydraulic fluid.

Figure 14-32. Valve removal tool.

the entrapped air. An example procedure for bleeding the strut is as follows:

1. Attach a bleed hose containing a fitting suitable for making an airtight connection at the strut service valve port. Ensure a long enough hose to reach the ground while the aircraft is on jacks.
2. Jack the aircraft up until the shock struts are fully extended.
3. Release any air pressure in the shock strut.
4. Remove the air service valve assembly.
5. Fill the strut to the level of the service port with approved hydraulic fluid.
6. Attach the bleed hose to the service port and insert the free end of the hose into a container of clean hydraulic fluid. The hose end must remain below the surface of the fluid.
7. Place an exerciser jack under the strut's jacking point. Compress and extend the strut fully by raising and lowering the jack. Continue this process until all air bubbles cease to form in the container of hydraulic fluid. Compress the strut slowly and allow it to extend by its own weight.
8. Remove the exerciser jack. Lower the aircraft and remove all other jacks.
9. Remove the bleed hose assembly and fitting from the service port of the strut.
10. Install the air service valve, torque, and inflate the shock strut to the manufacturer's specifications.

EXTENSION/RETRACTION SYSTEMS: NORMAL AND EMERGENCY

The retraction of landing gear reduces drag, makes possible faster flight and/or reduced fuel consumption. For most helicopters, the main gears are retracted in a sideward direction and the nose landing gear is retracted in a forward or aft direction.

HYDRAULIC EXTENSION AND RETRACTION SYSTEMS

Hydraulic systems are widely used to operate the landing gear systems due to their compact size, high response rate, load holding capabilities and excellent power to weight ratio. Landing gear hydraulic systems normally take power from engine driven pumps and alternative backup systems in case of pump failure.

Hydraulic System Components

A typical retraction/extension hydraulic system consists of the following:

- A selector valve.
- A safety lock.
- Downlock and uplock mechanisms.
- Sequence valves.
- Restrictors and non-return valves.
- Downlock actuator/retract actuator/door unlatch actuator/door actuator.

Selector Valves

Landing gear retraction and/or extension is monitored from the cockpit with a gear handle that can be set in an UP, OFF, or DOWN position, and which is connected mechanically or electrically to a selector valve.

With the handle in the UP position an internal circuit in the selector valve supplies pressure from the hydraulic system for:

- Unlocking and opening the wheel well doors by means of the unlatch and door actuators.
- Unlocking the landing gears by means of the downlock actuator.
- Retracting the landing gears by means of the retract actuator.
- Closing the wheel well doors by means of the door and unlatch actuators.

In this case, the hydraulic high pressure is on one side of the retract actuator while the other side is then connected to a low pressure return line. The piston installed in the retract actuator moves and the landing gear is retracted and locked.

With the handle in the OFF (neutral position) all hydraulic components on the UP as well as the DOWN side are connected to the return line. In this state, depending on the type of helicopter, the landing gears are locked by mechanical means, such as an uplock mechanisms. With the handle in the DOWN position the hydraulic pressure is released via an internal circuit in the selector valve. This is used for:

- Unlocking and opening the wheel well doors.
- Unlocking the up-lock.
- Extending the landing gear.
- Closing the wheel well doors.

Safety Locks

When the helicopter is resting on its wheels, a safety device prevents movement of the selector lever, thus avoiding an inadvertent retraction of the landing gear.

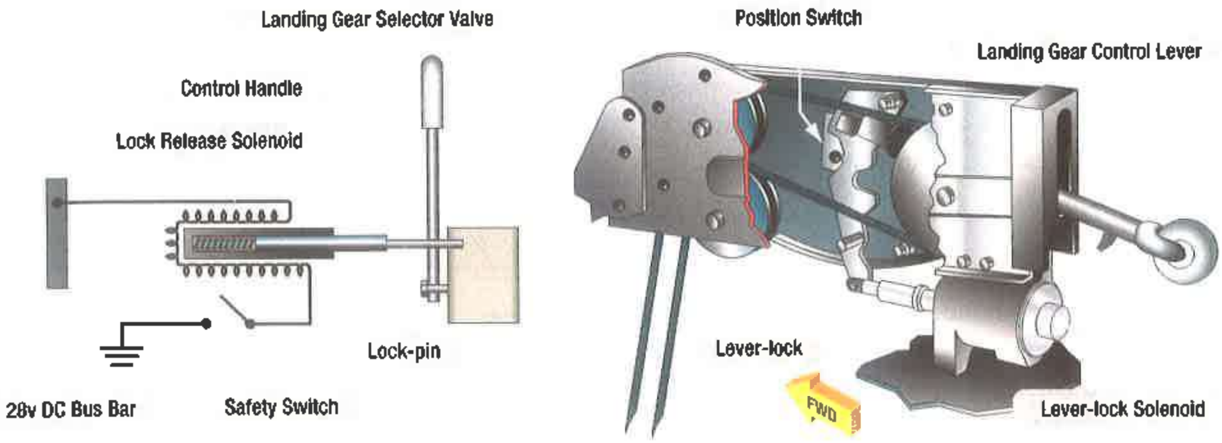


Figure 14-33. Safety lock mechanism.

The safety lock consists of a spring loaded plunger which retains the selector in the down position and is released by a solenoid. Electric power to the solenoid is controlled by a switch mounted on the shock strut. (***Figure 14-33***)

When the strut is compressed, the switch is open. When the strut extends after take off, the switch contacts close and the solenoid releases the selector lever lock to allow the gear position to be selected up. For emergency and maintenance purposes, a means of overriding the lock is provided, such as a separate gated switch to complete the circuit, or a mechanical means of avoiding the locking plunger.

Downlock And Uplock Mechanisms

Undesired retraction of the landing gear in the DOWN position is avoided by a downlock mechanism. Over center links between the strut and the side brace ensure that the side brace cannot pivot when in the over center position. The spring force of bungee springs maintain these links in the over center position. For safety reasons when on the ground or for towing, the mechanism is locked by lock pins installed and removed before flight by the ground staff.

If the landing gear is retracted, a downlock actuator pulls the over center links from the over center position. The side brace can pivot when the landing gear is pulled up by the retraction cylinder. Depending on the type of helicopter, the landing gear can be kept in the UP position by an uplock mechanism (***Figure 14-34***) or by hydraulic power. The uplock mechanism consists of a hook in which the landing gear is secured in the retracted position. The over center links can also be used for this purpose. In this case, they work as described in the downlock mechanism. In both cases, an uplock actuator is necessary to unlock the uplock.

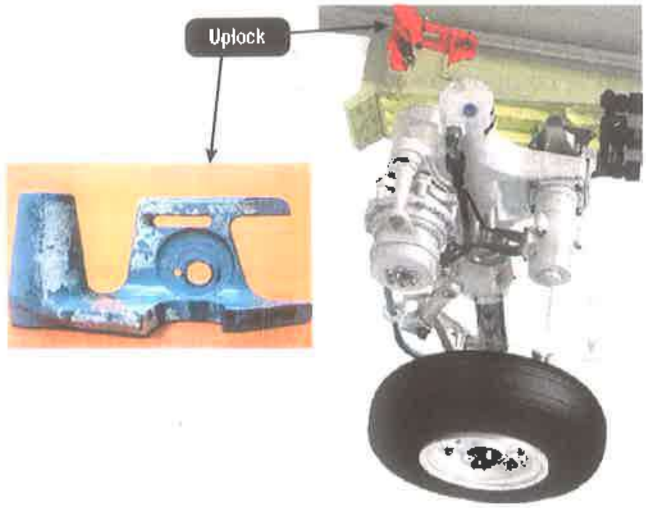


Figure 14-34. Uplock mounted into a landing gear bay structure.

In ***Figure 14-35*** a locking actuator installed on each gear leg secures the uplock or downlock positions. Here, this is a linear actuator type, electrically operated by an internal motor to disengage the lock by a spring loaded plunger fully extended by an internal spring. Prior to extension and retraction, the locking actuator electrically retracts the spring loaded plunger and maintains its position until the retraction actuator has fully retracted/extended the gear. The locking actuator is then de-energized and the plunger is allowed to be extended by the spring force into its slot. If the gear is unlocked, it extends due to its mass and reaches the "down and locked" position with the help of bungee springs. The hydraulic fluid which flows away from the retraction actuator slows this process slightly to reduce the down shock.

Instead of an up-lock mechanism, the landing gears of some helicopters are kept in the UP position by hydraulic pressure. This is possible because hydraulic pressure is available during the whole flight. If this pressure is lost for any reason, the gears come to rest in the uplocks or on the wheel well doors.

Ground Locks

As mentioned before, ground locks are used to prevent the gear from retracting or collapsing while on the ground. They are commonly used on aircraft landing gear as extra insurance that the gear will remain down and locked while the aircraft is on the ground. They are external devices that are placed in the retraction mechanism to prevent its movement. A ground lock can be as simple as a pin placed into the predrilled holes of gear components to keep the gear from collapsing. Another commonly used ground lock clamps onto the exposed piston of the retraction cylinder, preventing it from retracting.

All ground locks should have a red streamer attached to them so they are visible and removed before flight. (*Figure 14-36*) Streamers are carried in the aircraft and put into place by the flight crew during the post landing walk around.

Sequence Valves

When equipped with hydraulically operated gear doors and uplock/downlock mechanisms, the wheel well doors should be opened or closed at the right moment of the extension/retraction sequence to let the landing gear pass. In this case, sequence valves ensure operation of the hydraulic components at the correct time and direction.

Restrictors and non-return valves are also used to ensure the correct sequence of the operation. Some parts are supplied with hydraulic pressure later than others. This creates pressure differences which cause operation in the desired order. To limit the speed of lowering of the main gear due to gravity, restrictors limit the exit of fluid from the retraction side of the actuator during the gear extension sequence. The nose landing gear often lowers against the slipstream and does not need the protection of a restrictor.

Actuators

Unlock and door actuators are used for hydraulically operated doors. If the landing gear handle is moved to "UP" or "DOWN", the unlock actuators will first unlock the doors, then the door actuators will open the doors. After the passage of the landing gear, the door actuators will close the doors again. Apart from sequence valves, restrictors and non-return valves, actuators based on the difference in piston surface, can also help determine the correct order of a sequence.

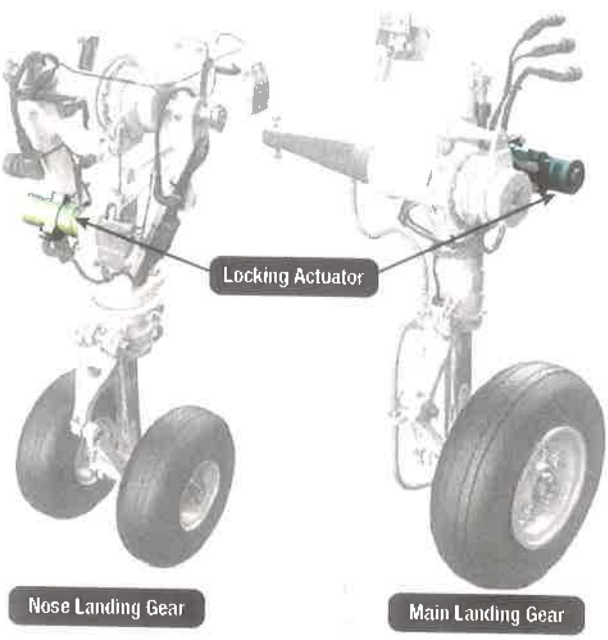
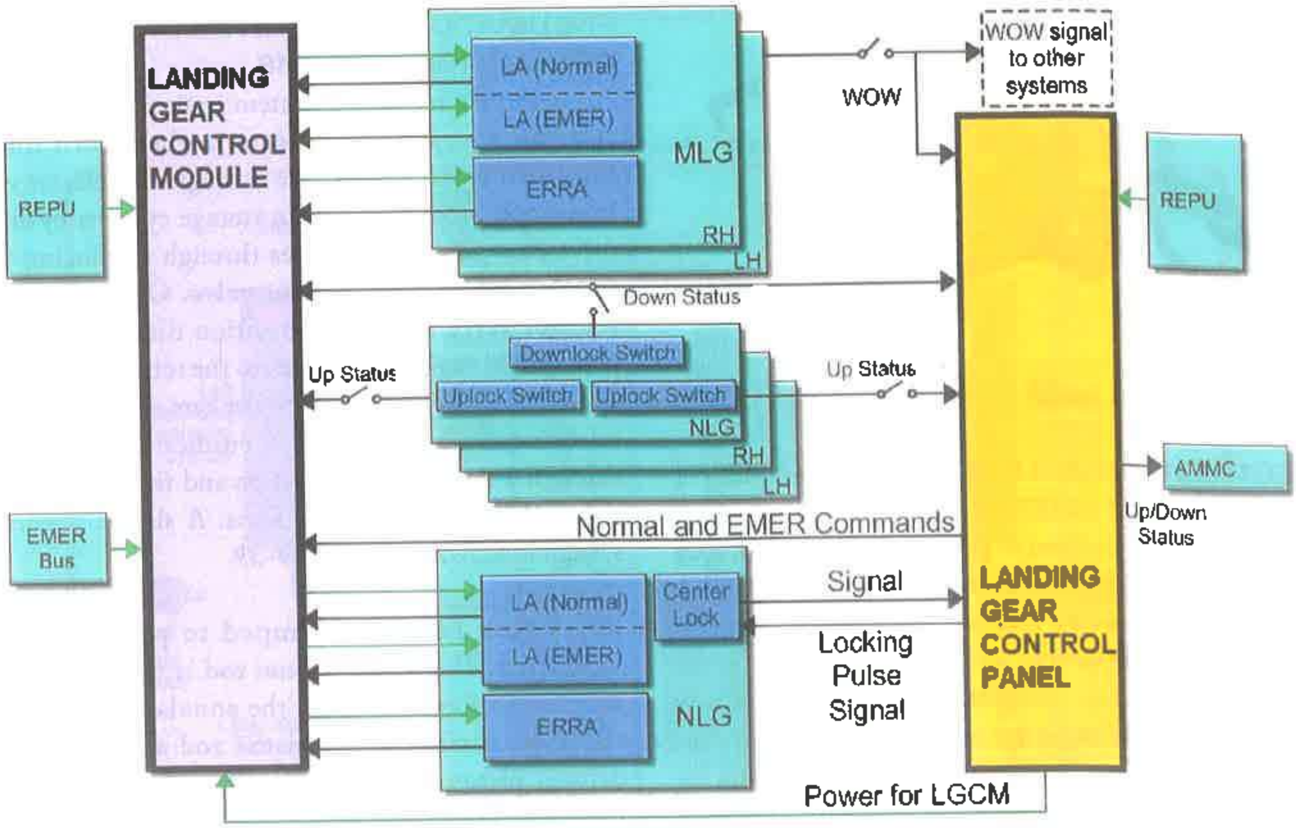


Figure 14-35. Locking actuator.

Figure 14-36. Gear pin ground lock devices.



LA / LOCKING ACTUATORS
MLG / MAIN LANDING GEAR
NLG / NOSE LANDING GEAR
EMER / EMERGENCY
LGCM / LANDING GEAR CONTROL MODULE
REPU / REMOTE ELECTRICAL POWER UNIT
WOW / WEIGHT ON WHEEL
ERRA / ELECTRICAL RETRACTION ROTARY ACTUATOR
AMMC / AIRCRAFT AND MISSION MANAGEMENT COMPUTER

Figure 14-37. Example of an electrically driven extension-retraction system on a AW169.

There are numerous hydraulic landing gear system designs. Priority valves are sometimes used instead of mechanically operated sequence valves. This controls some gear component activation timing via hydraulic pressure. Particularities of any gear system are found in aircraft maintenance manuals. The aircraft technician must be thoroughly familiar with the operation and maintenance requirements of this crucial system.

ELECTRIC EXTENSION AND RETRACTION SYSTEMS

Electrically driven extension/retraction systems exist as well, such as on AW169 helicopters (*Figure 14-37*), where the system is managed by a dedicated landing gear control module and controlled by a landing gear control panel (an electromechanical assembly) for control of the extension/retraction system, of the parking brake and of the nose gear center lock.

All electric systems work with an electric motor and gear reduction to move the landing gear. The gear is actuated by converting the rotary motion of the motor to linear motion. This type of mechanism is practical only with smaller aircraft because of the gear's lesser weight. In general, the electric motor works through a combination of gears, shafts, adapters, a torque tube, and an actuator screw. The whole system is initialized with a switch in the cockpit. When the switch is in the UP position, force is transmitted to the drag strut linkages to retract and lock the gear. This also activates the struts that open and close the gear doors.

When the switch is in the DOWN position, the motor's motion is reversed. As a result, the gear moves down and locks. Once the switch is depressed, the gear motor continues to operate until an up or down limit switch on the motor's gearbox is triggered.

Figure 14-38. Example of a hydraulic power pack.

ELECTRIC/HYDRAULIC EXTENSION AND RETRACTION SYSTEMS

Another common method of gear retraction system is an electric/hydraulic system, also known as a power pack system. (*Figure 14-38*) This is a self contained system which houses a reservoir and a set of valves for the landing gear and flap. An electrically driven pump may also be included or the system may be powered by engine driven pumps. This type of system normally provides for powered retraction of the landing gear with extension being by free-fall with the assistance of spring struts.

PNEUMATIC EXTENSION AND RETRACTION SYSTEMS

A pneumatic retraction system is similar to hydraulic systems, except that pressure in the return lines is exhausted to the atmosphere through the selector valve. Pressure is built up in a main storage cylinder by engine driven air pumps and passes through a reducing valve to the landing gear selector valve. Operation of the selector valve to the UP position directs pneumatic pressure through the UP lines to the retraction rams and opens the DOWN line to atmosphere. Operation of the selector valve to the DOWN position directs pressure through a second reducing valve and the down lines, to the up-lock and retraction rams. A simple pneumatic system is shown in *Figure 14-39.*

Retraction rams are damped to prevent violent movement. The hollow piston rod is filled with oil or grease, and forced through the annular space between the inner surface of the piston rod and a stationary damper piston whenever the ram extends or retracts, thus slowing movement.

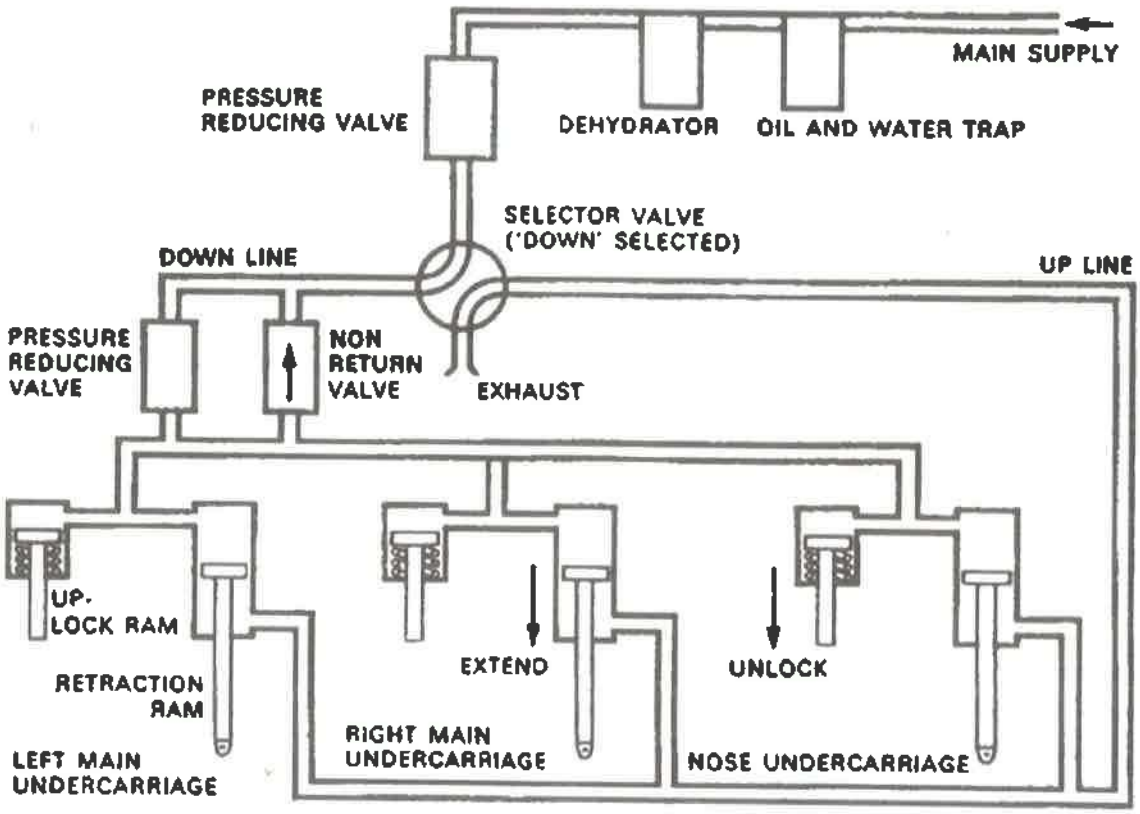


Figure 14-39. A simple pneumatic system.

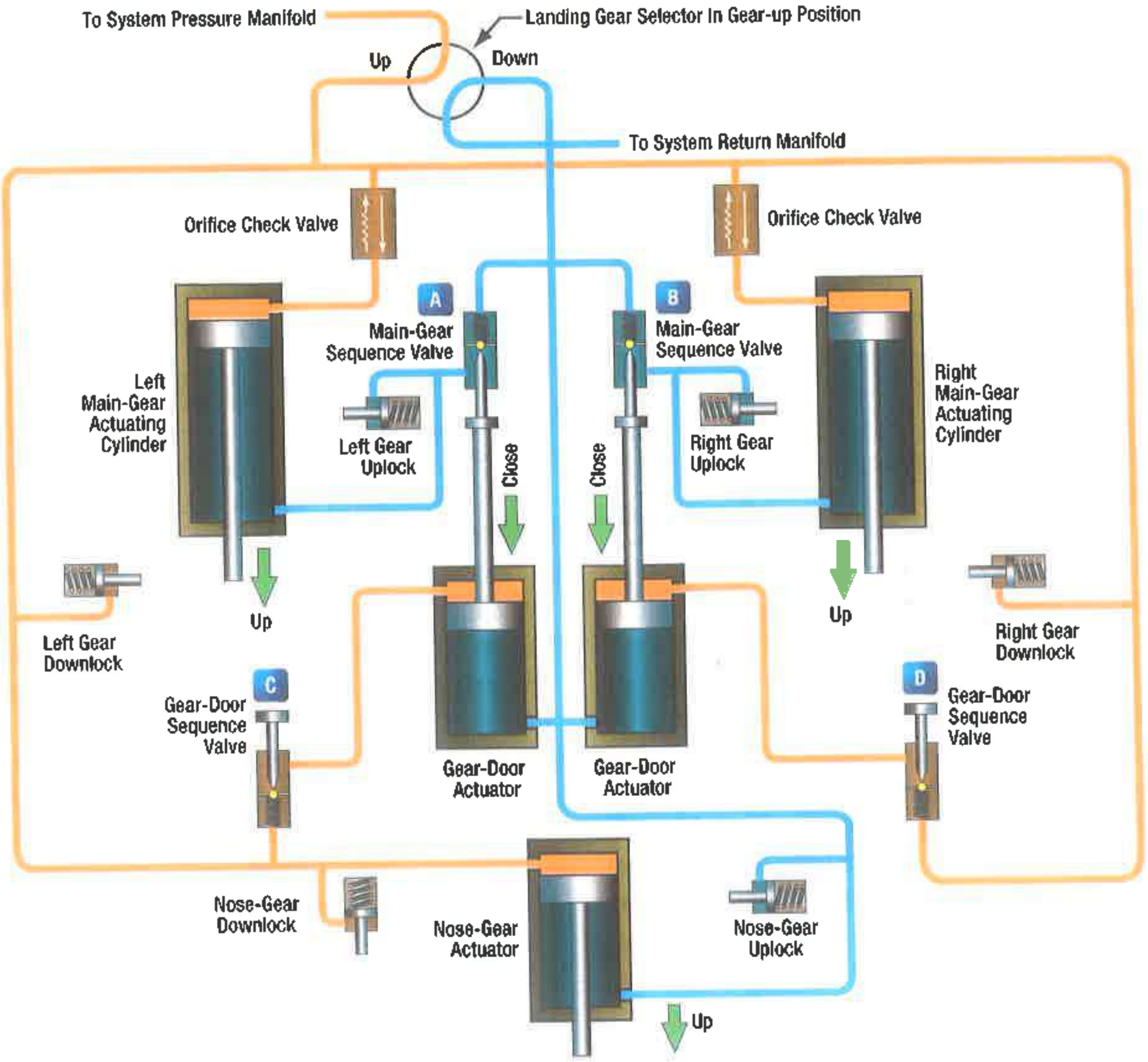


Figure 14-40. A simple hydraulic gear retraction system.

NOTE: Low pressure is used for landing gear extension for the same reason restrictor valves are used in hydraulic systems, to prevent damage through too rapid extension of the undercarriage.

Uplocks and downlocks are similar to those with hydraulic systems, with geometric downlocks being imposed by over centering of the drag strut at the end of retraction ram stroke, and the uplocks by spring ram operated locks. Downlocks are released by the initial movement of the retraction rams during retraction. Up locks are released by pneumatic pressure in the spring rams during extension. Undercarriage doors are operated mechanically, by linkage on the shock absorber housing.

NORMAL EXTENSION

The correct operation of any gear retraction system is extremely important. *Figure 14-40* illustrates an example of a simple hydraulic landing gear system. This system is on an aircraft that has doors that open before the gear is extended and close after the gear is retracted. The nose gear doors operate via mechanical linkage and do not require hydraulic power. Some aircraft have gear doors that close to fair the wheel well after the gear is extended. Others have doors mechanically attached to the outside of the gear so that when it stows inward, the door stows with the gear and fairs with the fuselage skin.

In the system illustrated, when the gear selector is moved to the UP position, it positions a selector valve to allow pump pressure from the hydraulic system manifold to access eight different components.

The three downlocks are pressurized and unlocked so the gear can be retracted. At the same time, the actuator cylinder on each gear also receives pressurized fluid to the upside of the piston through an unrestricted orifice check valve. This drives the gear into the wheel well. Two sequence valves (C and D) also receive pressure. Gear door operation must be controlled so that it occurs after the gear is stowed. The sequence valves are closed and delay flow to the door actuators. When the gear cylinders are fully retracted, they mechanically contact the sequence valve plungers that open the valves and allow fluid to flow into the side of the door actuator cylinders. This closes the doors. Sequence valves A and B act as check valves during retraction allowing fluid to flow one way from the gear downside of the main gear cylinders back into the hydraulic system return manifold through the selector valve.

To lower the gear, the selector is put in the DOWN position. Pressurized fluid flows from the manifold to the nose gear uplock, which unlocks the gear. Fluid flows to the gear downside of the nose actuator and extends it. Fluid also flows to the open side of the main door actuators. As the doors open, sequence valves A and B block fluid from unlocking the main uplocks and prevent fluid from reaching the downside of the main gear actuators. When the doors are fully open, the door actuator engages the plungers of both sequence valves to open the valves. The main gear uplocks then receives fluid pressure and unlock. The main gear cylinder actuators receive fluid on the downside through the sequence valves to extend the gear. Fluid from each main gear cylinder flows to the hydraulic system return manifold through restrictors in the orifice check valves to slow the extension of the gear.

Figure 14-41 illustrates an example of a landing gear extension sequence. ***Figure 14-42*** illustrates an example of a hydraulic power pack gear retraction system in the gear up condition.

EMERGENCY EXTENSION SYSTEMS

The emergency extension system lowers the landing gear if the main power system fails. There are numerous ways in which this is done depending on the size and complexity of the aircraft.

Mechanical Emergency Extension Systems

Some aircraft have an emergency release handle (***Figure 14-43***) in the flight deck that is connected through a mechanical linkage to the gear uplocks. When the handle is operated, it releases the uplocks and allows the gear to free fall to the extended position by gravity.

In this example, the wheel well doors and the landing gears can be mechanically unlocked by means of control cables. After the wheel well doors have been opened, the landing gear will come down by gravity. The hydraulic

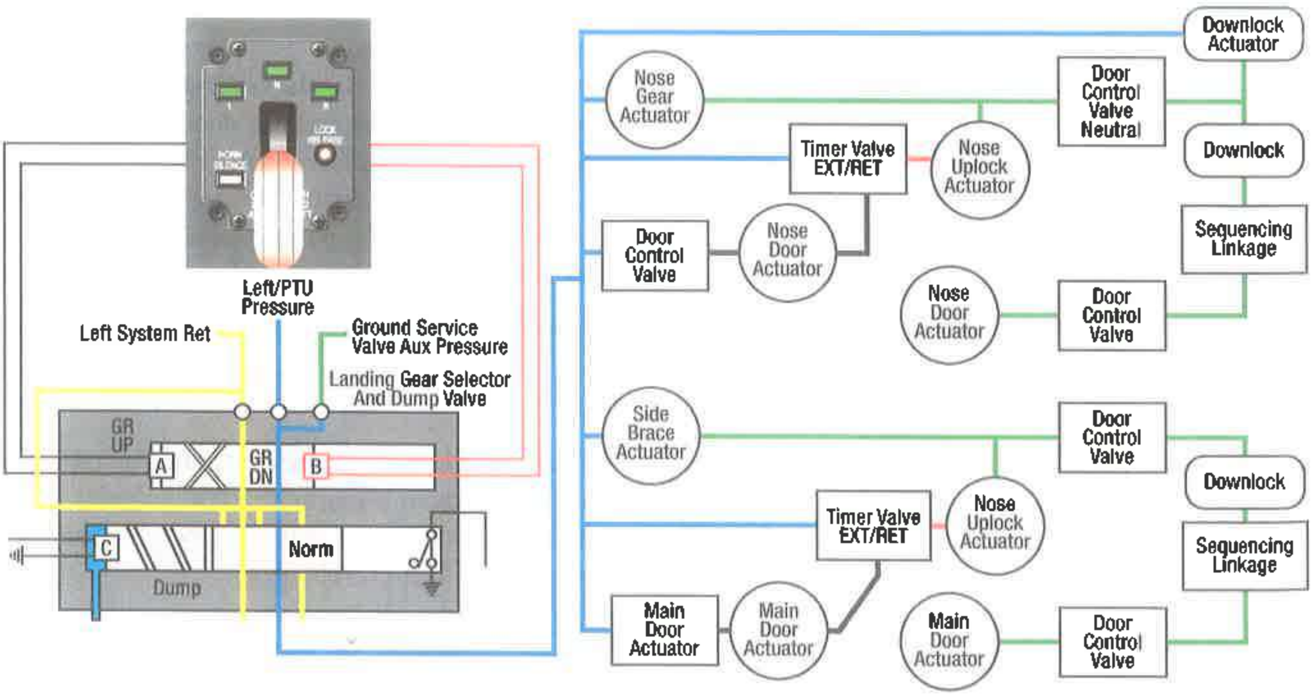


Figure 14-41. Landing gear extend sequence.

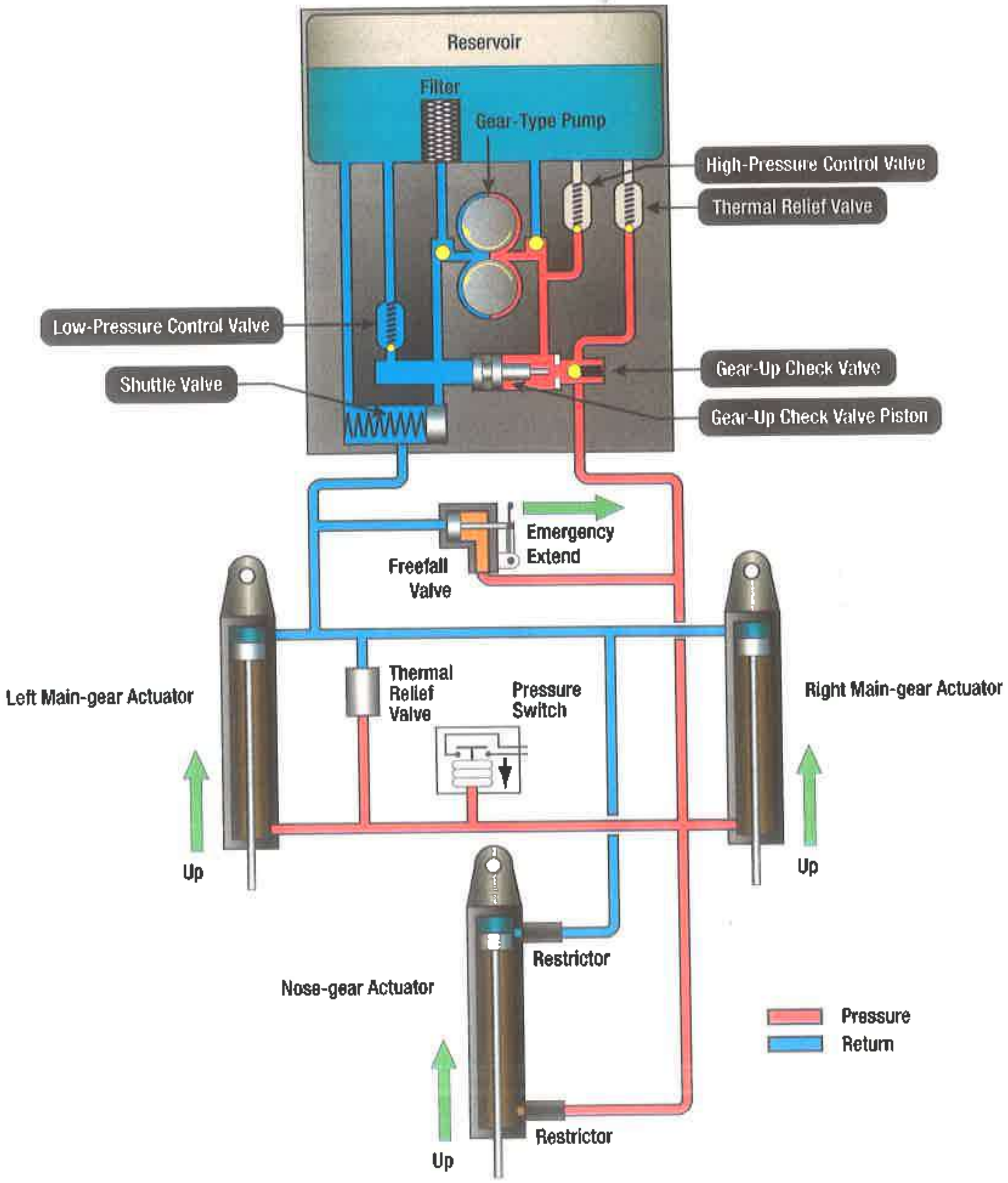


Figure 14-42. Hydraulic power pack gear retraction system in the gear up condition.

fluid in the system must be able to flow around all hydraulically operated cylinders. For this reason, bypass valves are opened as the emergency extension system is operated. Some aircraft use a non-mechanical back-up, such as pneumatic power, to unlatch the gear.

Hydraulic Emergency Extension Systems

High performance aircraft are equipped with redundant hydraulic systems. (***Figure 14-44***) The faulty circuit is replaced by an emergency hydraulic circuit. The release and the operating actuators are powered by positioning an emergency selector or a safe guarded control switch. (***Figure 14-45***)

This makes emergency extension less common since a different source of hydraulic power can be selected if the gear does not function normally. If the gear still fails to extend, some sort of unlatching device is used to release the uplocks and allow the gear to free fall. Pressure for the emergency system may be supplied by a hydraulic accumulator, a hand pump, a pneumatic storage cylinder, or an electrically powered pump. Consult the aircraft maintenance manual for all emergency landing gear extension system descriptions of operation, performance standards, and emergency extension tests as required.

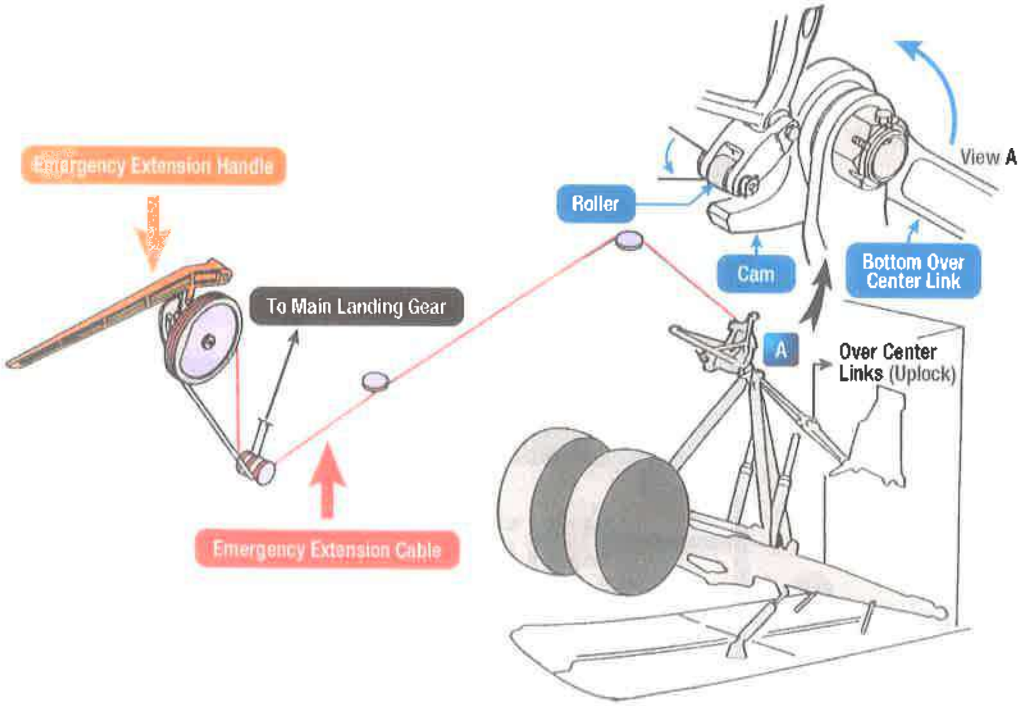


Figure 14-43. Emergency release handle and mechanism.

LANDING GEAR RETRACTION TEST

The proper functioning of a landing gear system and its components can be checked by performing a gear retraction test, also known as "swinging the gear". The aircraft is properly supported on jacks and the gear cleaned and lubricated if needed. The gear is then raised and lowered as though the aircraft were in flight while a close visual inspection is performed. All parts of the system should be observed for security and proper operation. The emergency back-up extension system should also be checked whenever swinging the gear.

Retraction tests are performed at various times, such as during annual inspection. Any time a landing gear component is replaced that could affect the correct functioning of the gear, a retraction test should follow or when adjustments to gear linkages or components that affect gear system performance are made. It also may be necessary to swing the gear after a hard or overweight landing. It is also common to test the gear when attempting to locate a malfunction within the system. For all required retraction tests and the specific points to check, consult the manufacturer's manual for the aircraft in question as each landing gear system is unique.

The following is a general list of inspection items to be performed while swinging the gear:

- Check the gear for proper extension and retraction.
- Check all switches, lights, and warning devices for proper operation.
- Check the gear doors for clearance and freedom from binding.
- Check all linkages for proper operation, adjustment, and general condition.
- Check the alternate/emergency extension/retraction systems for proper operation.
- Investigate any unusual sounds, such as those caused by rubbing, binding, chafing, or vibration.

INDICATIONS AND WARNING

Although the landing gear, when selected DOWN, may be visible from the crew compartment, it is not usually possible to be certain that each is securely locked. An electrical indicating system is used to provide a positive indication to the crew of the position and status of the landing gear.

Control of the landing gear and annunciating its position is usually done through micro-switches on the uplocks and downlocks, which make or break when the locks operate, and which are connected to a landing gear position indicator on the instrument panel. A mechanical indicator may also be provided, to show the gear status when the electrical system is inoperative.

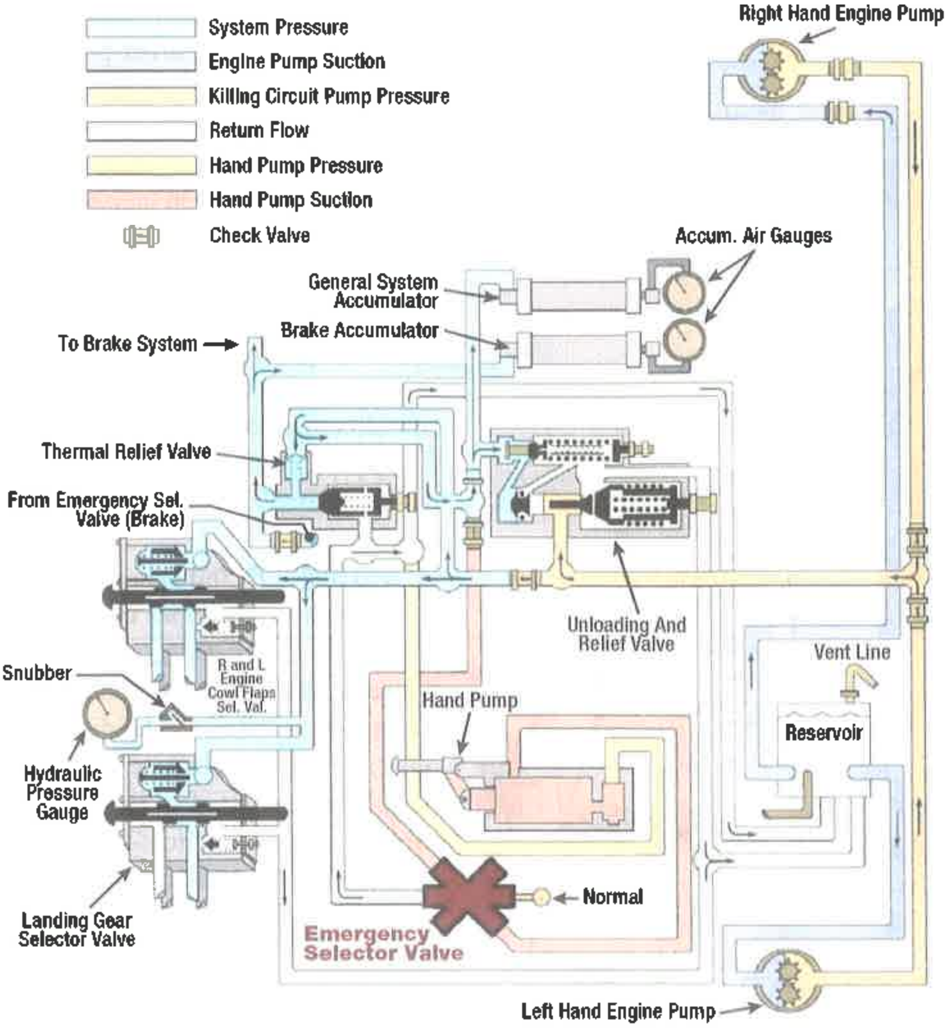


Figure 14-44. Landing gear hydraulic schematic.

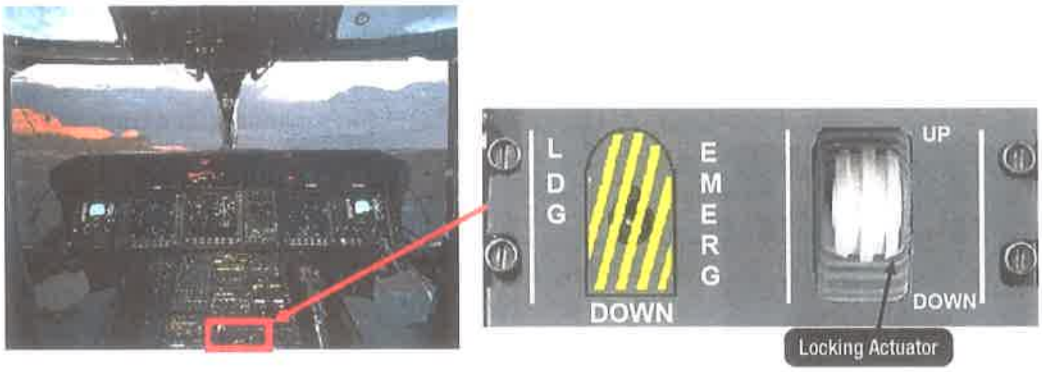


Figure 14-45. Emergency control switch.

Solid state circuits are controlled with proximity switches located on the gear so the position is always known. For example, DOWN and LOCKED versus DOWN and NOT LOCKED. Position indicators are located on the instrument panel adjacent to the gear selector handle. (*Figure 14-46*)

There are many arrangements for gear indication, usually a dedicated light for each gear. The most common display for the gear being down and locked is an illuminated green light. (*Figure 14-47*) Three green lights means it is safe to land. All lights out typically indicates that the gear is up and locked, or there may be

Figure 14-46. Landing gear selector panels.

Figure 14-47. Landing gear lights.

gear up indicator lights. Gear in transit lights are used on some aircraft as barber pole displays indicating that the gear is in transit. In the example, three indication lamps show the landing gear operation status:

- Retracted = position lamps OFF
- In transit = position lamps ON AMBER
- Extended = position lamps ON GREEN

No visible indications are present on the control panel when the gear is fully retracted. The nose gear center push button with an indication lamp controls the nose wheel center lock mechanism. An electrical EMER DOWN button controls the emergency selection of the control valve for emergency extension that is displayed as:

- AMBER ON indication on emergency mode.
- No visible indications in the normal mode.

Some manufacturers install a gear disagreement annunciation when the landing gear is not in the same position as the selector. (***Figure 14-48***) A warning horn is incorporated in the system and connected to a throttle operated switch. If one or more throttle levers are less than one third open, as would be the case during approach to land, the horn sounds and the red lamp illuminates if the gear is in any position other than down and locked. A horn isolation switch is often provided to allow certain flight exercises and ground servicing to be done without hindrance. Many aircraft monitor gear door position in addition to the gear itself. If the warning system in the cockpit should fail to work, many helicopters have view windows which make it possible to check the position of the landing gear. Consult the manufacturer's manuals for a complete description of the indication system.

SAFETY SWITCHES

As mentioned above, gear position indicators are safety devices used to communicate to the pilot the position status of each individual landing gear at any time. Various other safety circuits ensuring sequential operation of the landing gear and system components dependent on the air-ground status of the aircraft are also common.

Squat Switches, Proximity Sensors

Such switches are positioned to open and close depending on the extension or compression of the main landing gear strut. (***Figure 14-49***) The squat switch is wired into

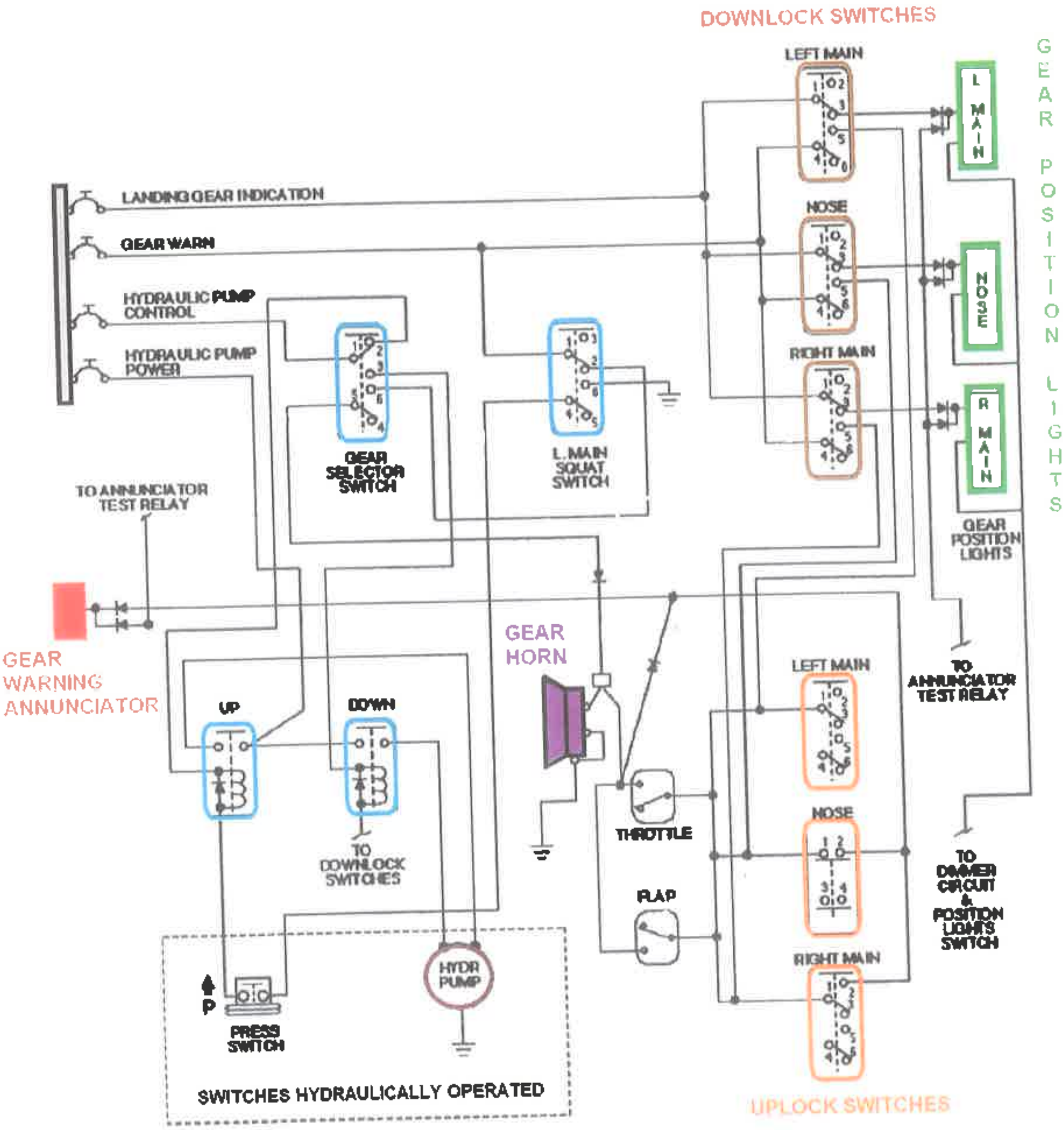


Figure 14-48. Landing gear electrical diagram with a warning annunciator.

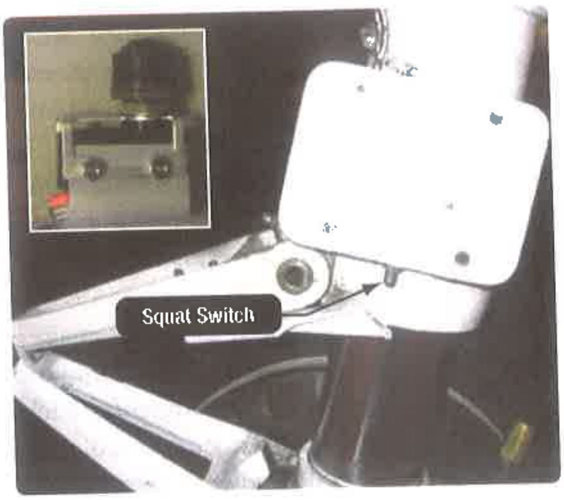


Figure 14-49. Typical landing gear squat switches.

any number of system operating circuits. One circuit prevents the gear from being retracted while the aircraft is on the ground. There are different ways to achieve this lockout.

A solenoid that extends a shaft to physically disable the gear position selector is one method found on many aircraft. When the landing gear is compressed, the squat switch is open, and the center shaft of the solenoid protrudes a hardened lock pin through the landing gear control handle so that it cannot be moved to the up position. At takeoff, the landing gear strut extends. The safety switch closes and allows current to flow in the safety circuit. The solenoid energizes and retracts

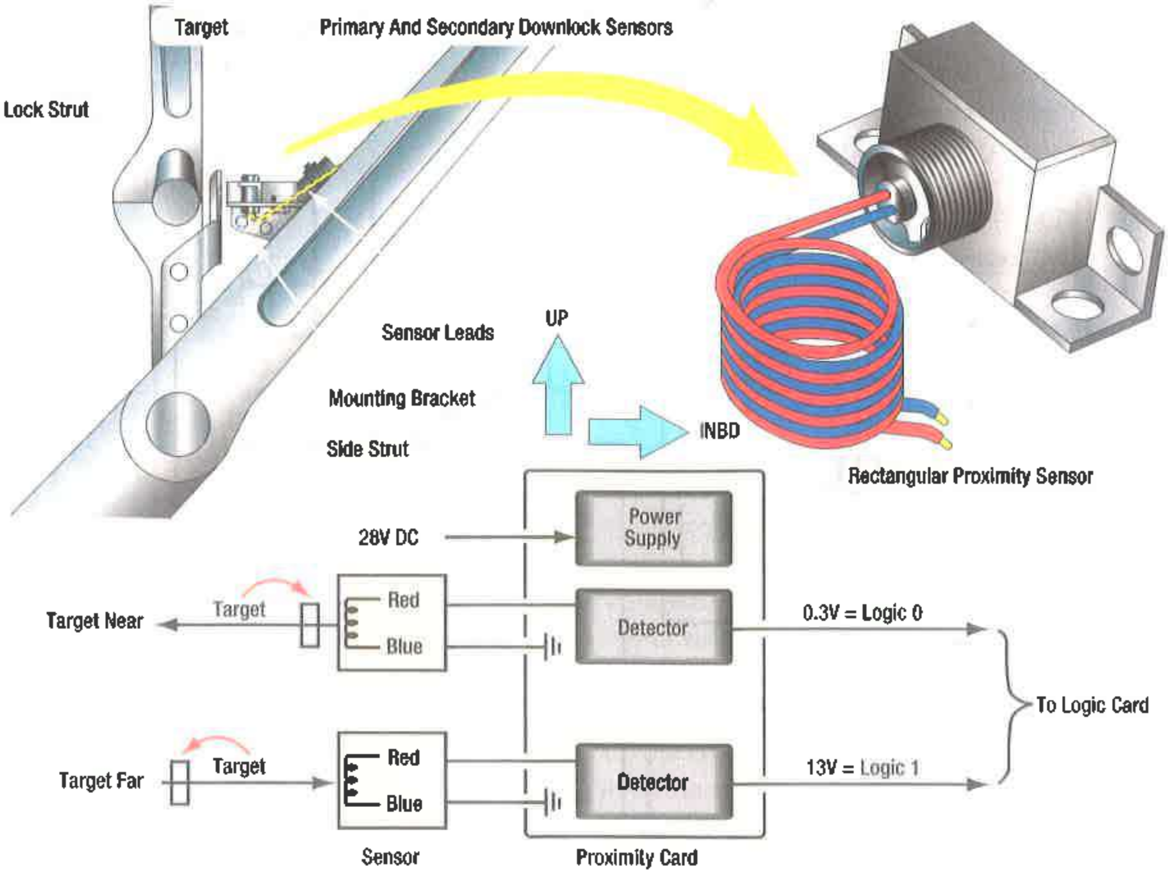


Figure 14-50. Proximity sensors.

the lock pin from the selector handle. This permits the gear to be raised. An electromagnetic sensor returns a different voltage to a logic unit depending on the proximity of a conductive target to the switch. Thus no physical contact is made. When the gear is in the designed position, the metallic target is close to an inductor in the sensor which reduces the return voltage to an electronic logic unit located in the equipment bay.

This type of sensing is especially useful in the landing gear environment where switches with moving parts can become contaminated with dirt and moisture. The technician is required to ensure that sensor targets are installed the correct distance away from the sensor. Go-No-Go gauges are often used to set the distance. (***Figure 14-50***)

WHEELS, TIRES, BRAKES

WHEELS

Most current helicopter wheels are made of metallic alloy materials, usually aluminum or magnesium, that offer sufficient strength and stiffness at relatively low density. Forging processes are ideal for wheel support and braking applications. These processes enable the production of one-piece parts with complex geometries and optimal mechanical characteristics. The use of materials such as aluminum, titanium and high performance steels also helps reduce part weights while ensuring superior performance and reduced maintenance phases.

The future of helicopter wheels is composite materials, allowing them to be lighter, with higher performance and an extended working life. Composite wheels for rotorcraft will soon be common due to improvements in resin system toughness, lower carbon fiber prices, improvements in process automation, and 30-40% weight savings over forged aluminum. Moreover, composite wheels have an improved corrosion resistance and better noise, and vibration performances.

As aircraft tires were improved, they were made stiffer to better absorb the forces of landing without blowing out or separating from the rim. As stretching such a tire over a single piece wheel rim is not possible, the two piece wheel was developed. Early two piece wheels were essentially one piece wheels with a removable rim to

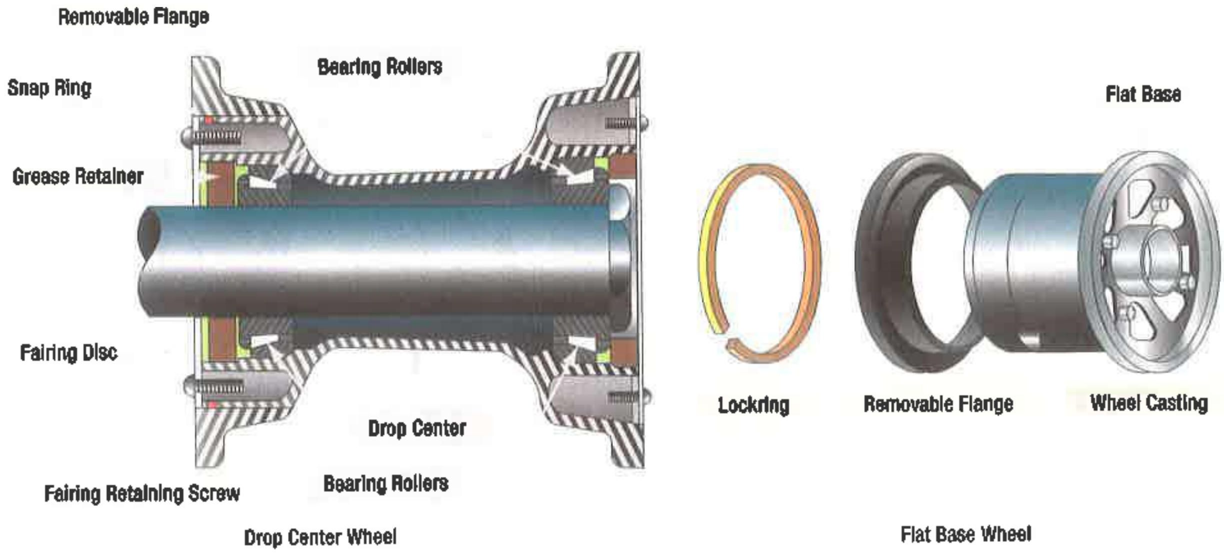


Figure 14-51. Removable flange wheels.

allow mounting for the tire. (***Figure 14-51***) Later, wheels with two nearly symmetrical halves were developed. Nearly all modern aircraft wheels are of this two piece construction. (***Figure 14-52 and Figure 14-53***)

Wheel Construction

Typical two piece wheel halves are bolted together and contain a groove at the mating surface for sealing the assembly with tubeless tires. The bead seat area of a wheel is where the tire contacts the wheel. It is the critical area that accepts the significant loads from the tire during landing. To strengthen this area during manufacturing, the bead seat is typically rolled to prestress it with a compressive load.

Inboard Wheel Half

Wheel halves are not identical as the inboard wheel half must have a means for accepting the rotor(s) of the brakes that are mounted on the main wheels. Tangs on the rotor are fitted into steel reinforced keyways on most wheels.

Other wheels have steel keys bolted to the inner halves to fit slots in the perimeter of the brake rotor. Some small wheels have provisions for bolting the brake rotor to the inner wheel half. Regardless, the inner wheel half is distinguishable from the outer half by its brake mounting feature. (***Figure 14-54***)

Both wheel halves contain a bearing cavity formed into the center that accepts the polished steel bearing cup, tapered roller bearing, and grease retainer of a typical

Figure 14-52. Two-piece split-wheel aircraft wheels.

bearing setup. A groove may also be machined to accept a retaining clip to hold the bearing assembly in place when the wheel is removed. The inner wheel half used on a high performance aircraft may also likely have one or more thermal plugs. (***Figure 14-55***)

An over inflation safety plug may be installed in the inner wheel half. This is designed to rupture and release all the air in the tire should it be over inflated. The fill valve is also often installed in the inner wheel half with the stem extending through holes in the outer wheel half to permit access for inflation and deflation.

Outboard Wheel Half

The outboard wheel half bolts to the inboard wheel half to make up the wheel assembly upon which the tire is mounted. The center boss is constructed to receive a bearing cup and bearing assembly as it does on the inboard wheel half. The outer bearing and end of the axle is capped to prevent contaminants from entering this area. The outboard half provides a convenient location for the valve stem used to inflate and deflate tubeless tires. Alternately, it may contain a hole through

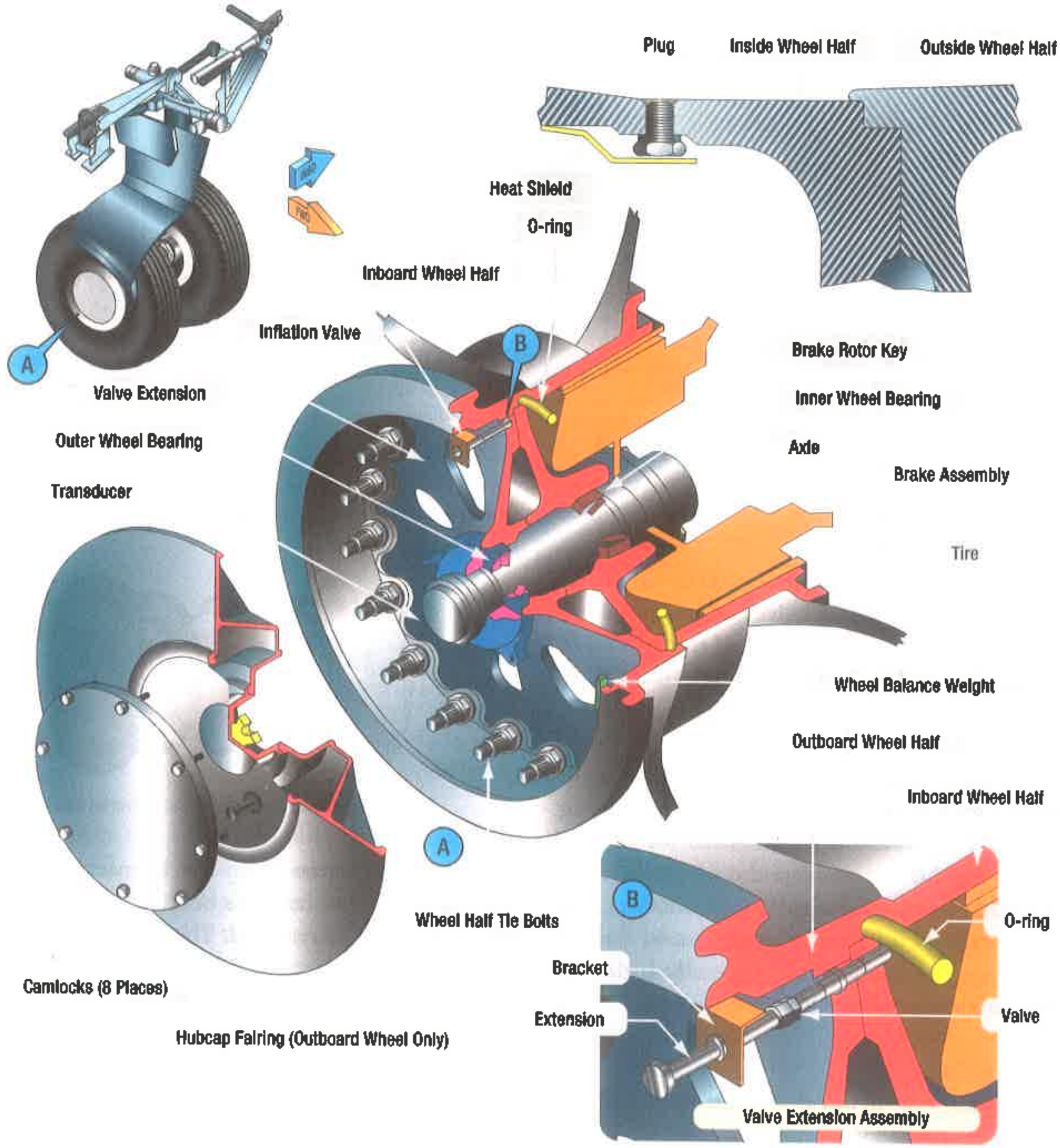


Figure 14-53. Features of a two piece aircraft wheel.

Figure 14-54. Keys on the inner wheel half.

which a valve stem extension may pass from the inner wheel half, or the valve stem itself may fit through such a hole if a tube type tire is used.

On Aircraft Wheel Inspection

The landing gear area is such a hostile environment that the technician should inspect the landing gear including the wheels, tires, and brakes whenever possible. The general condition can typically be inspected while on the aircraft. However, any signs of suspected damage may require further testing and so the removal of the wheel assembly from the aircraft may be required.

Figure 14-55. Location of a thermal plug.

Proper Installation

Proper installation of the wheels should not be taken for granted. All tie bolts and nuts must be in place and secure. A missing bolt is cause for the wheel's removal. A thorough inspection of the wheel halves in accordance with the manufacturer's procedures must be performed due to the stresses that may have occurred. The inboard wheel half should interface with the brake rotor with no signs of chafing or excessive movement. All brake keys on the wheel must be present and secure. Each wheel should be observed to ensure it is not abnormally tilted. Flanges should not be missing any pieces, and there should be no areas on the wheel that show significant impact damage.

Examine the wheels for cracks, flaked paint, or evidence of overheating. Inspect thermal plugs to ensure no sign that the fusible alloy has melted. Thermal plugs that have allowed pressure loss in the tire require that the wheel assembly be removed for inspection. All other wheels with brakes and thermal plugs should be inspected closely while on the aircraft to determine if they too have overheated.

Axle Nut Torque

Axle nut torque is of extreme importance on an aircraft wheel installation. If the nut is too loose, the bearing and wheel assembly may have excessive movement. The bearing cup(s) could loosen and spin which could damage the wheel. There could also be impact damage from the bearing rollers which lead to bearing failure. An over torqued axle nut prevents the bearing from properly accepting the weight of the aircraft. The bearing spins without sufficient lubrication to absorb the heat caused by the higher friction level. This too leads to bearing failure. All axle nuts must be installed and torqued in accordance with the airframe manufacturer's maintenance procedures.

Off Aircraft Wheel Inspection

Discrepancies found while inspecting a wheel mounted on the aircraft may require further inspection with the wheel removed from the aircraft. Other items such as bearing condition, can only be performed with the wheel assembly removed. A complete inspection of the wheel requires that the tire be removed from the rim.

CAUTION: Deflate the tire before removing the wheel assembly from the aircraft. Wheel assemblies have been known to explode while removing the axle nut, especially when dealing with high pressure tires. The torque of the nut can be the only force holding together a defective wheel with broken tie bolts. It is also important to let aircraft tires cool before removal. Three hours or more is needed for cool down. Approach the wheel assembly only from the front or rear. Do not stand in the path of the released air and valve core trajectory when removing air as it could seriously injure the technician should it release from the stem.

As an additional precaution, remove only one wheel assembly from a pair at a time. This leaves a tire and wheel in place should the aircraft fall off its jack, resulting in less chance of damage to the aircraft or injury to personnel.

Disassembly Of The Wheel

After inflation and usage, a tire tends to adhere to the wheel, and the bead must be broken to remove it with a mechanical or hydraulic press designed for this purpose.

Disassembly of the wheel should take place in a clean area on a flat surface. Remove the wheel bearing first and set aside for cleaning and inspecting. The tie bolts can then be removed. Do not use an impact tool to disassemble the tie bolts as aircraft wheels are not designed to receive the repeated hammering and will be damaged.

Clean the wheel halves with the solvent recommended by the manufacturer. Use of a soft brush helps this process. Avoid abrasive techniques or materials that are capable of removing the finish from the wheel as corrosion can quickly form and weaken the wheel if the finish is missing. When the wheels are clean, they can be dried with compressed air.

Inspection Of The Wheel Halves
A thorough visual inspection of each wheel half should be conducted for discrepancies specified in the wheel manufacturer's maintenance data. Use of a magnifying glass is recommended.

Corrosion is one of the most common problems encountered while inspecting wheels. Locations where moisture is trapped should be checked closely. It is possible to dress out some corrosion according to the manufacturer's instructions. An approved protective surface treatment and finish must be applied before returning the wheel to service. Corrosion beyond stated limits is cause for rejection of the wheel.

Cracks in certain areas of the wheel are particularly prevalent, such as in the bead seat area. (***Figure 14-56***) The high stress of landing is transferred to the wheel by the tire in this contact area. This is a concern on all wheels and is most problematic in high pressure forged wheels. Dye penetrant inspection is generally ineffective when checking for cracks in the bead area as cracks tend to close tightly once pressure is removed from the metal. Eddy current inspection of the bead seat area is typically required.

Tie Bolt Inspection
Wheel half tie bolts are under great stress and require inspection. The tie bolts stretch and change dimension usually at the threads and under the bolt head. These are areas where cracks are most common. Magnetic particle inspection can reveal these cracks.

Key And Key Screw Inspection
On most inner wheel halves, keys are screwed or bolted to the wheel to drive the brake disc(s). The keys are subject to extreme forces when the brakes are applied. There should be no movement between the wheel and the keys. The area around the keys should be inspected for cracks. There is also a limitation on how worn the keys can be since too much wear allows excessive movement. The manufacturer's instructions should be used to perform a complete inspection of this critical area. (***Figure 14-57***)

Fusible Plugs
Fusible or thermal plugs must be inspected visually. These threaded plugs have a core that melts at a lower temperature than the outer part of the plug. This is to release air from the tire should the temperature rise to a dangerous level. A close inspection should reveal whether any core has experienced deformation that might be due to high temperature. If detected, all thermal plugs in the wheel should be replaced with new plugs.

Balance Weights
When manufactured, each wheel set is statically balanced. Weights are added to accomplish this if needed. These are a permanent part of the wheel assembly and must be installed to use the wheel. The balance weights are bolted to the wheel halves and can be removed when cleaning and inspecting the wheel. They must be re-fastened in their original position.

Figure 14-56. The bead seat areas of a light aircraft wheel set.

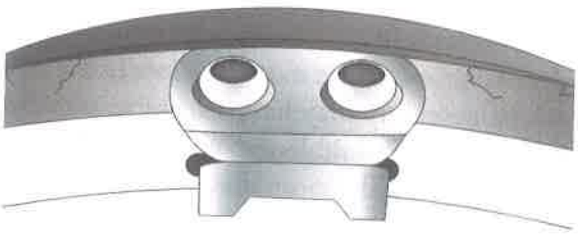


Figure 14-57. Wheel disc drive key area.

When a tire is mounted to a wheel, balancing of the wheel and tire assembly may require additional weights. These are usually installed around the circumference of the wheel and should not be taken as substitutes for the factory wheel set balance weights. (***Figure 14-58***)

Wheel Bearings

Cleaning Wheel Bearings

The bearings should be removed from the wheel to be cleaned with the recommended solvent. Soaking the bearings in solvent is acceptable to loosen any dried-on grease. Bearings are brushed clean with a soft bristle brush and dried with compressed air. Never rotate the bearing while drying with compressed air. The high speed metal to metal contact of the rollers with the race causes heat that damages the metal surfaces.

Inspection Of Wheel Bearings

When inspecting the bearing, check for defects that would render it unserviceable, such as cracks, flaking, broken bearing surfaces, roughness due to impact pressure or surface wear, corrosion or pitting, discoloration from excessive heat, cracked or broken bearing cages, and scored or loose bearing cups or cones that would affect proper seating on the axle or wheel. If any discrepancies are found, replace the bearing with a serviceable unit. Bearings should be lubricated immediately after cleaning and inspection to prevent corrosion.

The bearing cup does not require removal for inspection; however it must be firmly seated in the wheel half boss. There should be no evidence that a cup is loose or able to spin. (***Figure 14-59***) If necessary, the cup is usually removed by heating the wheel in a controlled oven and pressing it out or tapping it out with a non metallic drift. The installation procedure is similar. The wheel is heated and the cup is cooled with dry ice before it is tapped into place with a non metallic hammer or drift. The outside of the bearing race is often sprayed with primer before insertion. Consult the manufacturer's manual for specific instructions.

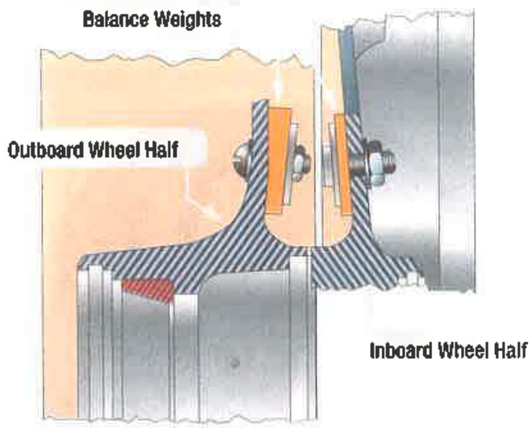


Figure 14-58. Wheel balance weights.

Lubrication of Bearings

Proper lubrication is a partial deterrent to environmental impacts on a bearing. Use of a pressure bearing packing tool is recommended as the best method to remove any contaminants from inside the bearing that may have remained after cleaning. (***Figure 14-60***) Use only the lubricant recommended by the manufacturer.

Figure 14-59. Loose bearing cup damage.

Figure 14-60. A bearing packing tool.

TIRES AND TUBES

A helicopter's tire must withstand a wide range of operational conditions. When on the ground, it must support the weight of the helicopter. During taxi, it must provide a stable cushioned ride while resisting heat generation, abrasion and wear. At take-off, it must be able to endure not only the helicopter's load but also the forces generated at high angular velocities. Landing incurs impact shocks while also transmitting braking loads to the ground. All of this must be accomplished while providing a long, dependable, reliable service life.

Tire Classification

Aircraft tires are classified by type, ply rating, tube type or tubeless, and whether they are bias ply tires or radials. Identification by its dimensions is also used.

Ply Rating

Tire plies are reinforcing layers of fabric encased in rubber that are laid into the tire to provide strength. In early tires, the number of plies used was directly related to the load the tire could carry. Now, refinements to construction techniques and the use of modern materials makes the exact number of plies somewhat irrelevant when determining its strength. However, a ply rating is still used to convey the relative strength. A tire with a high ply rating is of higher strength and able to carry heavy loads regardless of the actual number of plies used.

Tube Type Or Tubeless

Tires made to be used without a tube have an inner liner specifically designed to hold air. Tube type tires do not contain this liner since the tube holds the air from leaking. Tires meant to be used without a tube have the word tubeless on the sidewall. If this designation is absent, the tire requires a tube.

Bias Ply Or Radial

Another means of classifying tires is by the direction of the plies used in its construction, either bias or radial. Traditional aircraft tires are bias ply. The plies are wrapped to form the tire and give it strength. The angle of the plies in relation to the direction of rotation of the tire varies between 30° and 60°. In this manner, the plies have the bias of the fabric from which they are constructed towards the direction of rotation. The result is flexibility, as the sidewall can flex with the plies laid on the bias. (***Figure 14-61***)

Some modern aircraft tires are radial. The plies in radial tires are laid at a 90° angle to the direction of rotation. This configuration puts the non-stretchable fiber of the plies perpendicular to the sidewall and direction of rotation. This creates strength in the tire allowing it to carry high loads with less deformation. (***Figure 14-62***)

Tire Terminology

It is useful to the understanding of tire construction to identify the various components of a tire and the functions contributed to the overall characteristics of a tire. Refer to ***Figure 14-63*** for tire nomenclature used in this discussion.

Bead

The tire bead anchors the tire body and provides a dimensioned, firm mounting surface for the tire on the rim. Tire beads are typically made from high strength carbon steel wire bundles encased in rubber. One, two, or three bead bundles may be found on each side of the tire depending on its size and the load it is designed to handle. Radial tires have a single bead bundle on each side of the tire. The bead transfers the impact and deflection forces to the wheel rim. The bead toe is closest to the tire centerline and the bead heel fits against the flange of the wheel rim.

Figure 14-61. A bias ply tire.

Figure 14-62. A radial tire.

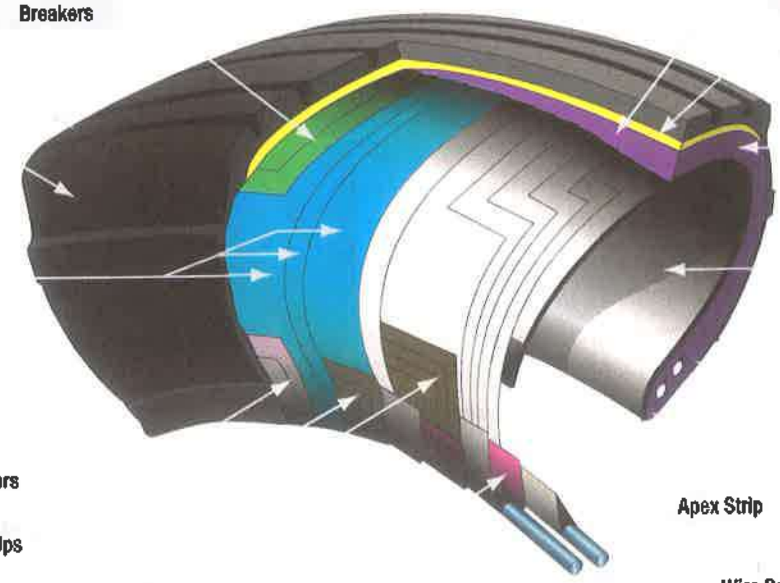


Figure 14-63. Construction nomenclature.

An apex strip is additional rubber formed around the bead to give a contour for anchoring the ply turn-ups. Layers of fabric and rubber called flippers are placed around the beads to insulate the carcass from the beads and improve tire durability. Chafers are also used in this area. Chafer strips made of fabric or rubber are laid over the outer carcass plies after the plies are wrapped around the beads. The chafers protect the tire from damage during mounting and demounting. They also help reduce the effects of wear and chafing between the wheel rim and the tire bead.

Carcass Plies

Carcass plies, (or sometimes called casing plies) are used to form the tire. Each ply consists of fabric sandwiched between two layers of rubber. The plies are applied in layers to give the tire strength and form the body of the tire. The ends of each ply are anchored by wrapping them around the bead on both sides of the tire to form the ply turn-ups. As mentioned, the angle of the fiber in the ply is manipulated to create a bias or radial tire as desired. Typically, radial tires require fewer plies than bias tires. Once the plies are in place, both bias and radial tires have a protective layer on top of the plies and under the tread of the running surface. On bias tires, these single or multiple layers of nylon and rubber are called tread reinforcing plies.

On radial tires, an under tread and a protector ply do the same job. These additional plies stabilize and strengthen the crown area of the tire. They reduce tread distortion under load and increase stability of the tire at high speeds. The reinforcing and protector plies also help resist puncture and cutting while protecting the body of the tire.

Tread

The tread is the crown which contacts the ground. It is a rubber compound formulated to resist wear, abrasion, cutting, cracking and heat buildup. Most aircraft tire tread is formed with circumferential grooves that create ribs. The grooves provide cooling and help channel water from under the tire in wet conditions to increase adhesion to the ground surface. Tires designed for aircraft frequently operated from unpaved surfaces may have some type of cross tread pattern. Older aircraft without brakes or brakes designed only to aid in taxi may not have these grooves.

All-weather treads may be found on some aircraft tires. This tread has circumferential ribs in the center of the tire with a diamond cross tread at the edge. (***Figure 14-64***)

The tread is designed to stabilize the aircraft on the operating surface. Extra tread reinforcement is sometimes accomplished with breakers. These are layers of nylon fabric under the tread that strengthen the tread

Figure 14-64. Aircraft tire treads.

while protecting the carcass plies. Tires with reinforced tread are often designed to be retreaded and used again once the tread has worn beyond limits. Consult the manufacturer's data for acceptable tread wear and retread capability for a particular tire.

Sidewall

The sidewall is a layer of rubber designed to protect the carcass plies. It may contain compounds designed to resist the negative effects of ozone. It is also where information about the tire is contained. The tire sidewall imparts little strength to the cord body. Its main function is protection. The inner sidewall of a tire is covered by the tire inner liner. A tube type tire has a thin rubber liner adhered to the inner surface to prevent the tube from chafing on the carcass. Tubeless tires are lined with a thicker, less permeable rubber. This replaces the tube and contains the nitrogen or air within the tire to keep it from seeping through the carcass plies. The inner liner does not contain 100 percent of the inflation gas. Small amounts seep through the liner into the plies. This seepage is released through vent holes in the lower outer sidewall. These are typically marked with a green or white dot of paint and must be kept unobstructed. Gas trapped in the plies could expand with temperature changes and cause separation of the plies, thus leading to failure. Tube type tires also have seepage holes in the sidewall to allow air trapped between the tube and tire to escape. (***Figure 14-65***)

Helicopters Tire Types

Helicopter tires have typically been classified into different Types:

- Type I tires are primarily for helicopters with non-retractable landing gear.
- Type III tires are generally used for low pressure service providing a larger footprint or flotation effect. These tires have smaller rim diameters relative to the overall diameter as compared to the other types.
- Type VII are high pressure tires widely used on large helicopters.

Figure 14-65. A sidewall vent.

Tire Inspection

Tire condition is inspected while mounted on the aircraft on a regular basis. Inflation pressure, tread wear and condition, and sidewall condition are continuously monitored to ensure proper performance.

Inflation

To perform as designed, an aircraft tire must be properly inflated. The manufacturer's maintenance data must be used to ascertain the correct inflation pressure for a tire on a particular aircraft. Do not inflate an unmounted tire to the pressure displayed on the sidewall or by how the tire looks. Tire pressure is checked while under load and is measured with the weight of the aircraft on the wheels. A calibrated pressure gauge should always be used to measure inflation pressure. Digital and dial type pressure gauges are more consistently accurate and preferred. (***Figure 14-66***)

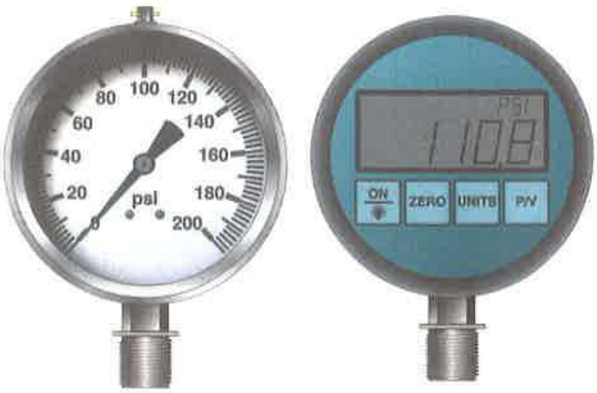


Figure 14-66. Tire pressure gauges.

In addition to overheating, under inflated tires wear unevenly, which leads to premature replacement. They may also creep or slip on the wheel rim when under stress or when the brakes are applied. Severely under inflated tires can pinch the sidewall between the rim and the runway causing sidewall and rim damage. Damage to the bead and lower sidewall area are also likely. This type of abuse, like any over flexing, damages the integrity of the tire and it must be replaced. In dual wheel setups, a severely under inflated tire affects both tires and both should be replaced.

Over inflation of aircraft tires is another undesirable condition. While carcass damage due to overheating does not result, adherence to the landing surface is reduced. Over a period of time, over inflation leads to premature tread wear and so reduces the number of cycles in service before the tire must be replaced. It also makes the tire more susceptible to bruises, shock damage, and blowout. (*Figure 14-67*)

Tread Condition

Condition of an aircraft tire tread can be determined while the tire is inflated and mounted on the aircraft. Evenly worn tread is a sign of proper tire maintenance. Uneven tread wear has a cause that should be investigated and corrected. A properly maintained evenly worn tire usually reaches its wear limits at the centerline of the tire. (*Figure 14-68*)

Asymmetrical tread wear may be caused by the wheels being out of alignment. Follow the manufacturer's instructions while checking caster, camber, tow-in, and tow-out to correct this situation. (*Figure 14-69*)

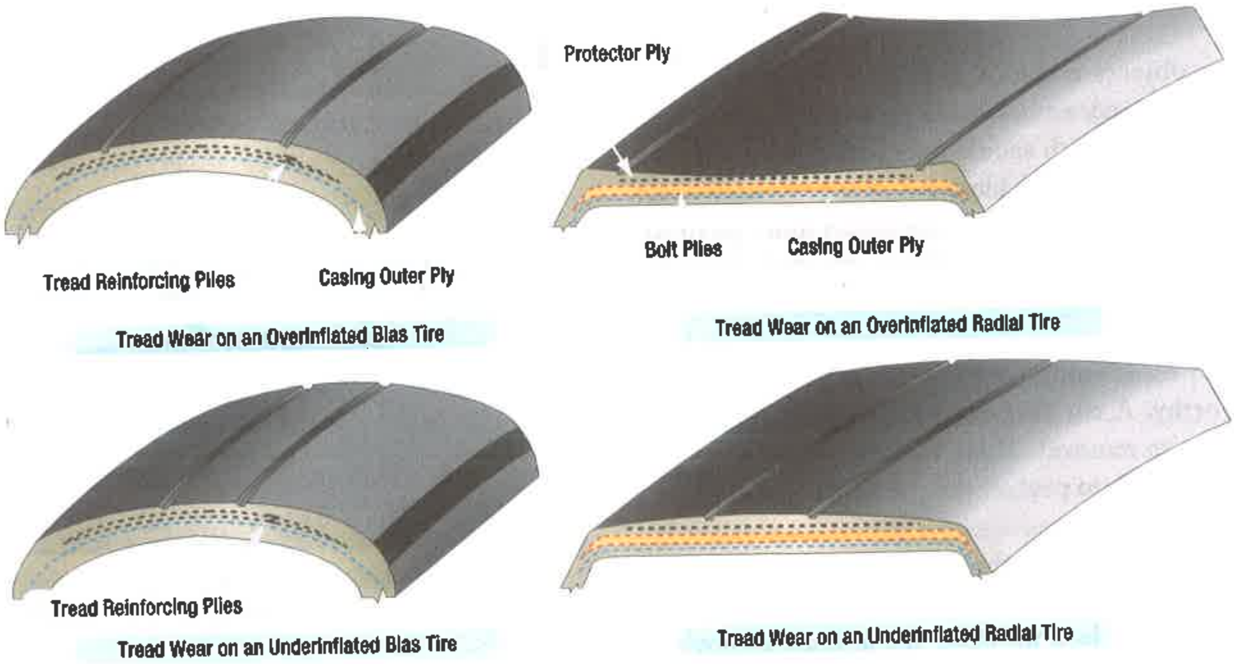


Figure 14-67. Improper inflation tire damage.

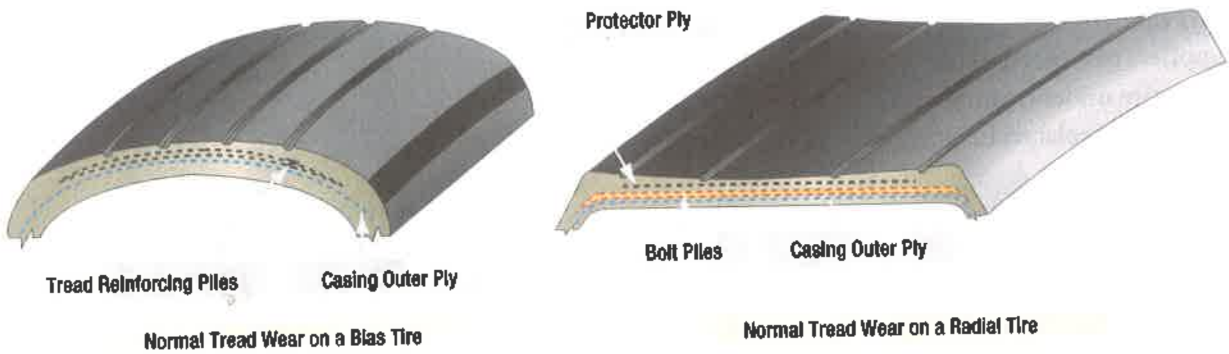


Figure 14-68. Normal tread wear.

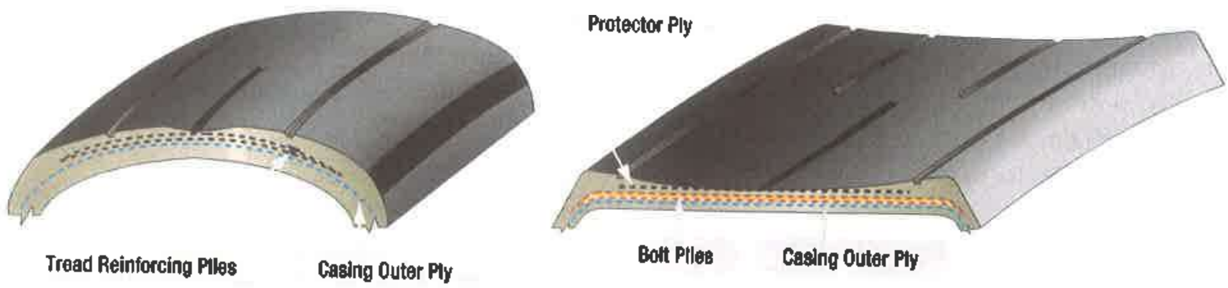


Figure 14-69. Unserviceable tread wear.

Tread Damage

In addition to tread wear, tires must be inspected for damage. Cuts, bruises, bulges, embedded objects, chipping, and other damage must be within limits to continue the tire in service. All damage, suspected damage, and areas with leaks should be marked with chalk, a wax marker, or other device before the tire is deflated or removed. Often, it is impossible to relocate these areas once the tire is deflated. (***Figure 14-70***)

Figure 14-70. Mark damaged area before deflating.

Foreign objects embedded in the tread should be removed when not embedded beyond the tread. Objects of questionable depth should only be removed after the tire has been deflated. A blunt awl or appropriately sized screwdriver can be used to pry the object from the tread. Care must be taken to not enlarge the damaged area during removal. (***Figure 14-71***)

Figure 14-71. Removal of FOD.

Cuts and tread undercutting can also render a tire unairworthy. A cut that extends across a tread rib is cause for tire removal. These can sometimes lead to a section of the rib to peel off the tire. (***Figure 14-72***)

Figure 14-72. Remove unairworthy tires from service.

A flat spot on a tire is the result of skidding on the runway surface while not rotating. This typically occurs when the brakes lock up while the aircraft is moving. If the flat spot damage does not expose the reinforcing ply of a bias tire or the protector ply of a radial tire, it may remain in service. However, if the flat spot causes vibration, the tire must be removed. Landing with a brake applied can often cause a severe flat spot that exposes the tire under tread. It can also cause a blowout. The tire must be replaced in either case. (***Figure 14-73***)

A bulge or separation of the tread from the tire carcass is cause for immediate removal and replacement of the tire. Mark the area before deflation as it could easily become undetectable without air in the tire. (***Figure 14-74***)


Figure 14-73. Tire damage caused by landing with the brake on.

Operation on a grooved runway can cause an aircraft tire tread to develop shallow chevron shaped cuts. These cuts are allowed for continued service unless chunks or cuts into the fabric of the tire result. Deep chevrons that cause a chunk of the tread to be removed should not expose more than 1 square inch of the reinforcing or protector ply. Consult the applicable inspection parameters to determine the allowable extent of chevron cutting. (*Figure 14-75*)

Tread chipping and chunking sometimes occurs at the edge of the tread rib. Small amounts of rubber lost in this way are permissible. (*Figure 14-76*)

Cracking in a tread groove is generally not acceptable if more than 1 cm of the reinforcing or protector ply is exposed. Groove cracks can lead to undercutting of the tread, which eventually can cause the entire tread to be thrown from the tire. (*Figure 14-77*)

Oil, hydraulic fluid, solvents, and other hydrocarbon substances soften tire rubber and make it spongy. A contaminated tire must be removed from service. If any volatile fluid is in contact with the tire, it is best to wash


Figure 14-75. Chevron cuts.

Figure 14-76. Tread chipping and chunking.

the tire with denatured alcohol followed by soap and water. Protect tires from contact with harmful fluids by covering tires during maintenance in the landing gear area. Tires are also subject to degradation from ozone and weather. Tires on aircraft parked outside for long periods of time should also be covered for protection from the elements. (*Figure 14-78*)

Sidewall Condition

If the sidewall cords are exposed due to a cut, gouge, snag, or other injury, the tire must be replaced. Mark the area of concern before removal of the tire. Damage

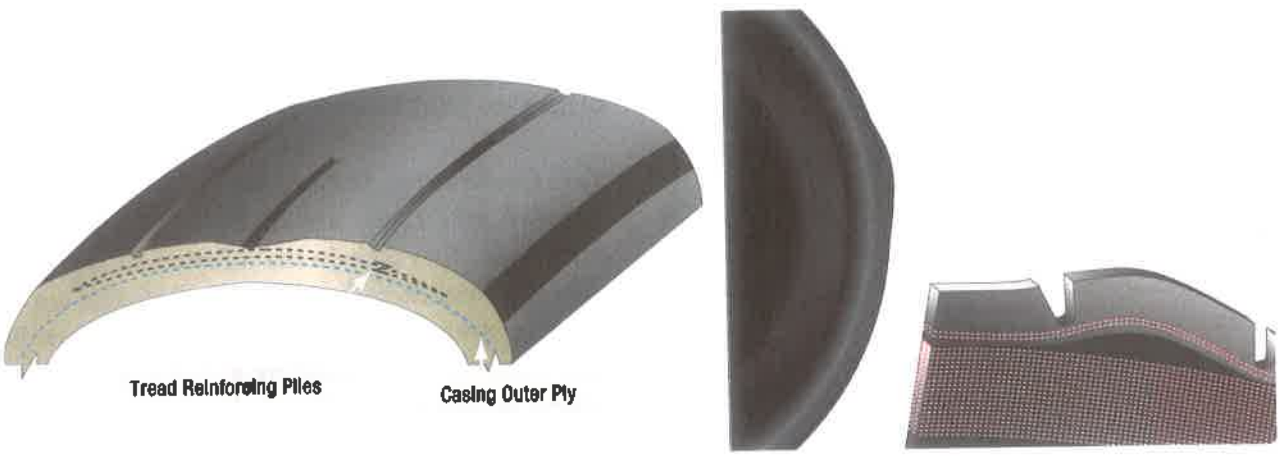


Figure 14-74. Bulges and tread separation.

Figure 14-77. A thrown tread.

Figure 14-78. Cover tires for protection.

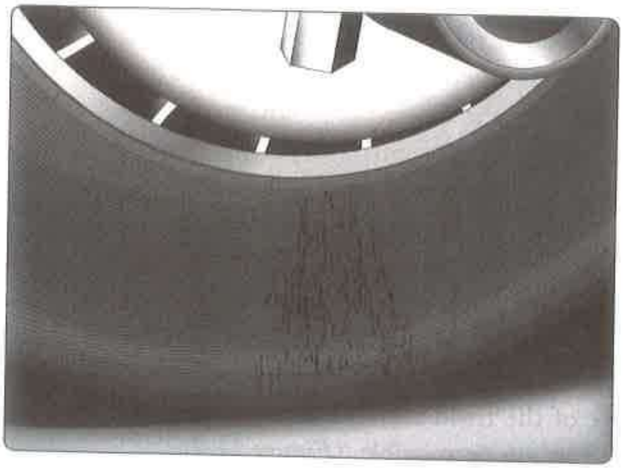


Figure 14-79. Cracking and checking in the sidewall of a tire.

to the sidewall that does not reach the cords is typically acceptable. Circumferential cracks or slits in the sidewall are unacceptable. A bulge in a tire sidewall indicates possible delamination of the carcass plies. The tire must immediately be removed from service. Weather and ozone can cause cracking and checking of the sidewall. If this extends to the sidewall cords, the tire must be removed. Otherwise, sidewall checking as shown in ***Figure 14-79*** does not affect the performance of the tire and it may remain in service.

Tire Storage

An aircraft tire can be damaged if stored improperly. A tire should always be stored vertically so that it is resting on its treaded surface. Horizontal stacking of tires is not recommended. Storage of tires on a tire rack with a minimum 3 cm flat resting surface for the tread is ideal and avoids tire distortion. Follow the manufacturer's instructions for storage.

The environment in which an aircraft tire is stored is critical. The ideal location in which to store an aircraft tire is cool, dry, and dark, free from air currents and dirt. Storage should be at cool temperatures to minimize degradation. A general range for safe aircraft tire storage is between 0°C and 40°C. Temperatures below this are sometimes acceptable but higher temperatures must be avoided.

Tubes

Tube type tires are handled and stored in similar fashion as tubeless tires. Several issues concerning the tubes themselves must be addressed.

Tube Construction And Selection

Aircraft tire tubes are made of a natural rubber compound. They contain the inflation air with minimal leakage. Unreinforced and special reinforced heavy duty tubes are available with nylon reinforcing fabric layered into the rubber for strength, to resist chafing and to protect against heat such as during braking. Tubes come in a wide range of sizes. Only the tube specified for the applicable tire size must be used.

Tube Storage And Inspection

Tubes should be kept in the original carton until put into service to avoid deterioration by environmental elements. If the original carton is not available, it can be wrapped in several layers of paper to protect it. Alternately, for short time periods, a tube may be stored in the correct size tire it is made for while inflated just enough to round out the tube. Application of talc in the inside of the tire and outside of the tube prevents sticking. Remove the tube and inspect it and the tire before permanently mounting the assembly. Regardless of storage method, always store tubes in a cool, dry, dark place away from ozone producing equipment and moving air. When handling and storing tubes, creases are to be avoided. These weaken the rubber and eventually cause failure. Creases and wrinkles also tend to be chafe points for the tube when mounted inside the tire. Never hang a tube over a nail or peg for storage.

An aircraft tube must be inspected for leaks and damage that may eventually cause a leak or failure. To check for leaks, remove the tube from the tire. Inflate the tube just enough to have it take shape but not stretch. Immerse a small tube in a container of water and look for the source of air bubbles. A large tube may require that water be applied over the tube. The valve core should also be wetted to inspect it for leaks.

There is no mandatory age limit for an aircraft tube. It should be elastic without cracks or creases to be considered serviceable. The valve area is prone to damage and should be inspected thoroughly. Bend the valve to ensure there are no cracks at the base where it is bonded to the tire or in the area where it passes through the hole in the wheel rim. Inspect the valve core to ensure it is tight and does not leak. If an area of a tube experiences chafing to the point where the rubber is thinned, the tube should be discarded. The inside diameter of the tube should be inspected to ensure it has not been worn by contact with the toe of the tire bead. Tubes that have taken an unnatural set should be discarded. (***Figure 14-80***)

Tire Mounting

A certified technician may be called upon to mount an aircraft tire onto the wheel rim in preparation for service. In the case of a tube type tire, the tube must also be mounted. Be sure to have the proper equipment and training to perform the work according to instructions.

Proper mounting ensures tires perform to the limits of their design. Consult and follow all manufacturer's service information including bolt torques, lubrication and balancing requirements, and inflation procedures. A wheel assembly that is to have a tire mounted upon it must be thoroughly inspected to ensure it is serviceable. Pay close attention to the bead seat area, which should be smooth and free from defects. The wheel half mating surface should be in good condition. The O-ring should be lubricated and in good condition to ensure it seals the wheel for the entire life of the tire. (***Figure 14-81***)

Most important is to check that the tire is specified for the aircraft application. It should say tubeless on the sidewall. The part number, size, ply rating, speed rating, and Technical Standard Order number should also be on the sidewall and be approved for the aircraft installation. Visually check the tire for damage from shipping and handling. There should be no permanent deformation of the tire. It should pass all inspections for cuts and other damage previously discussed. Clean the tire bead area with a clean shop towel and soap and water or denatured alcohol. Inspect the inside of the tire for condition. There should be no debris inside the tire. Tire beads are sometimes lubricated when mounted on aluminum wheels. Follow the manufacturer's instructions and use only the non-hydrocarbon lubricant specified. Do

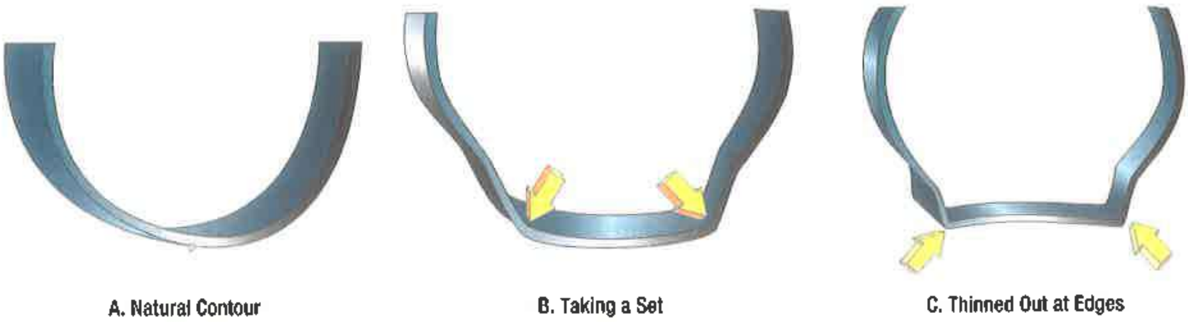


Figure 14-80. Inspect aircraft tire tubes.

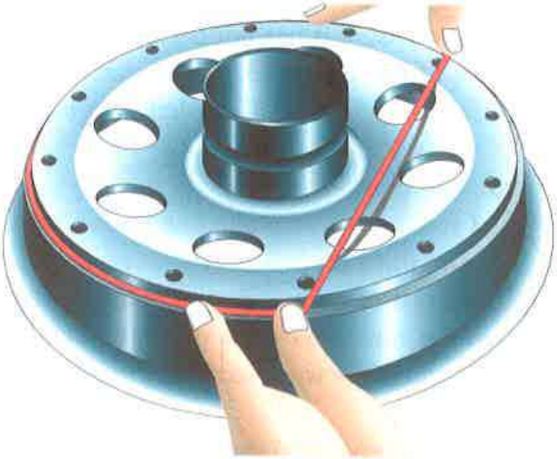


Figure 14-81. The wheel half O-ring.

not use lubricants with magnesium alloy wheels. Most radial tires are mounted without lubricant. The airframe manufacturer may specify lubrication for a radial tire in a few cases.

When the wheel halves and tires are ready to be mounted, thought must be given to tire orientation and the balance marks on the wheel halves and tire. Typically, the tire serial number is mounted to the outboard side of the assembly. The marks indicating the light portion of each wheel half should be opposite each other. The mark indicating the heavy spot of the wheel assembly should be mounted aligned with the light spot on the tire, which is indicated by a red mark. If the wheel lacks a mark indicating the heavy spot, align the red spot on the tire (the light point) with the valve fitting location on the wheel. A properly balanced tire and wheel assembly improves the overall performance of the tire. It promotes smooth operation free from vibration, which results in uniform tread wear and extended tire life.

Follow the manufacturer's instructions for tie bolt tightening sequences and torque specification. Anti-seize lubricants and wet-torque values are common on wheel assemblies. Use a calibrated hand torque wrench. Never use an impact wrench on an aircraft tire assembly.

For the initial inflation of an aircraft tire and wheel assembly, the tire must be placed in a tire inflation safety cage and treated as though it may explode due to wheel or tire failure. (***Figure 14-82***) The inflation hose should be attached to the tire valve stem, and inflation pressure should be regulated from a safe distance away. Air or nitrogen should be introduced gradually as specified. Dry nitrogen keeps the introduction of water into the

Figure 14-82. Modern tire inflation cages.

tire to a minimum, which helps prevent corrosion. Observe the tire seating progress on the wheel rim while it inflates. Depressurize the tire before approaching it to investigate any observed issue.

Aircraft tires are inflated to their full specified operating pressure. Then, they are allowed to remain with no load applied for 12 hours. During this time, the tire stretches and tire pressure decreases. A 5-10 percent reduction is normal. Upon bringing the tire up to full pressure again, less than five percent loss per day of pressure is allowable. More should be investigated.

Tube Type Tires

Wheel and tire inspection should precede the mounting of any tire, including tube tires. The tube to be installed must also pass inspection and be the correct size for the tire. Tire talc is commonly used when installing tube type tires to ensure easy mounting and free movement between the tube and tire as they inflate. (***Figure 14-83***)

Inflate the tube so that it just takes shape with minimal pressure. Install the tube inside the tire. Tubes are typically produced with a mark at the heavy spot of the tube. In the absence of this balance mark, it is assumed that the valve is located at the heaviest part of the tube. For proper balance, align the heavy part of the tube with the red mark on the tire (the light spot on the tire). Once wheel balance is marked and the tube balance mark and the tire balance mark are all positioned correctly, install the outboard wheel half so the valve stem of the tube passes through the valve stem opening. (***Figure 14-84***)

Figure 14-83. Tire talc.

Figure 14-84. Mounting a tube type tire on an outboard wheel half.

Mate the inboard wheel half to it, being careful not to pinch the tube between the wheel rims. Install the tie bolts, tighten, and torque as specified. Inflate the assembly in an inflation cage. The inflation procedure for a tube type tire differs slightly from a tubeless tire. The assembly is slowly brought up to full operating pressure. Then, it is completely deflated. Reinflate the assembly a second time to the specified pressure and allow it to remain with no load for 12 hours. This allows any wrinkles in the tube to smooth out, helps prevent the tube from being trapped under a bead, and generally evens how the tube lays within the tire to avoid any stretched areas and thinning of the tube. The holding time allows air trapped between the tube and the tire to work its way out of the assembly, typically through the tire sidewall or around the valve stem.

Tire Balancing

Once an aircraft tire is mounted, inflated, and accepted for service, it can be balanced to improve performance. Vibration is the main result of an imbalanced assembly. Nose wheels tend to create the greatest disturbance in the cabin when imbalanced. Static balance is all that is required for most aircraft wheels.

A balance stand typically accepts the assembly on cones. The wheel is free to rotate. The heavy side moves to the bottom. (***Figure 14-85***)

Temporary weights are added to stop the wheel from rotating and dropping the heavy side down. Once balanced, permanent weights are installed. Many aircraft wheels have provisions for securing the permanent weight to the wheel. Weights with adhesive designed to be glued to the wheel rim are also in use. Occasionally, a weight in the form of a patch glued to the inside of the tire is required. Follow all manufacturer's instructions and use only the weights specified for the wheel assembly. (***Figure 14-86***)

Some aviation facilities offer dynamic balancing of aircraft tire and wheel assemblies. While rarely specified by manufacturers, a well balanced tire and wheel assembly helps provide shimmy free operation and reduces wear on brakes and other components.

BRAKES

As wheeled helicopters are capable of ground taxiing, brakes are required. As high performance and military helicopters will sometimes perform forward moving takeoffs and landings, braking systems capable of controlling the mass of a helicopter at those speeds are also then required.

Figure 14-85. A typical aircraft tire and wheel static balancing stand.

Modern aircraft use disc brakes. Each of the main wheels is equipped with a brake unit. The nose wheel or tail wheel does not have a brake. In the typical brake system, mechanical and/or hydraulic linkages to brake pedals allow the pilot to control the brakes. Pushing on the top of the right brake pedal activates the brake on the right main wheel(s) and pushing on the top of the left brake pedal operates the brake on the left main wheel(s).

The basic operation of brakes involves converting the kinetic energy of motion into heat energy through the creation of friction. A great amount of heat is developed and forces on the brake system components are demanding. Proper adjustment, inspection, and maintenance of the brakes is essential for effective operation.

Types And Construction Of Helicopter Brakes

Helicopter brakes are friction brakes. The disc rotates with the turning wheel assembly while a stationary caliper resists the rotation by causing friction against the disc when the brakes are applied. The size, weight, and landing speed of the aircraft influence the design and complexity of the disc brake system. The different types of brakes are described below.

Single Disc Brakes

Light helicopters typically achieve effective braking using a single disc keyed or bolted to each wheel. As the wheel turns, so does the disc. Braking is accomplished by applying friction evenly to both sides of the disc from a non-rotating caliper bolted to the landing gear axle flange. Pistons in the caliper housing under hydraulic pressure force wearable brake pads or linings against the disc when the brakes are applied. Hydraulic master cylinders connected to the brake pedals supply pressure when the upper halves of the rudder pedals are pressed. The AW169 helicopter is equipped with a single cylinder dual piston metallic lined brake assembly. A detailed single disc brake is shown in ***Figure 14-87.***

Floating Disc Brakes

A floating disc braking system is designed with two calipers per disc with one caliper located on either side of the disc. This principle is shown in ***Figure 14-88.*** A floating disk brake is illustrated in ***Figure 14-89.*** A detailed, exploded view is shown in ***Figure 14-90.***

The caliper straddles the disc. It has three cylinders bored through the housing, but on other brakes this number may vary. Each cylinder accepts an actuating piston assembly comprised mainly of a piston, a return spring, and an automatic adjusting pin. Each brake assembly has six brake linings or pucks. Three are located on the ends of the pistons, which are in the outboard side of the caliper. They are designed to move in and out with the pistons and apply pressure to the outboard side of the

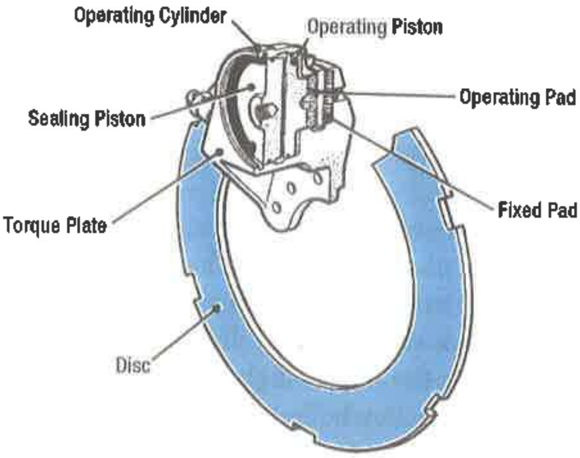


Figure 14-87. A single-disc brake.

Figure 14-86. Tire and wheel balancing weights.

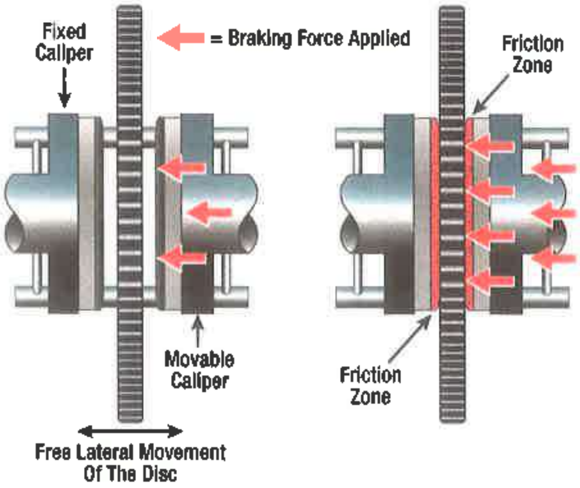


Figure 14-88. Floating-disc principle.

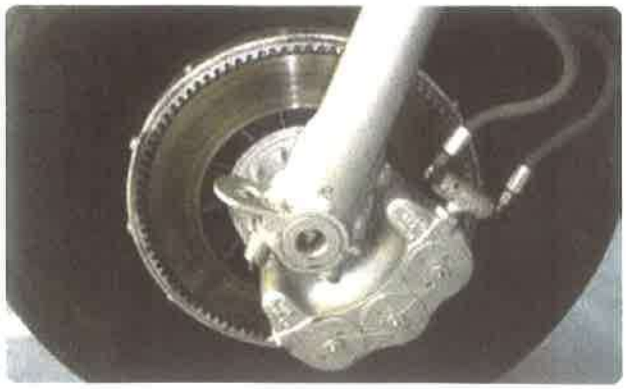


Figure 14-89. A single disc brake; floating-disc, fixed caliper.

disc. Three more linings are located opposite of these pucks on the inboard side of the caliper. These linings are stationary.

The brake disc is keyed to the wheel. It is free to move laterally in the key slots. This is known as floating. When the brakes are applied, the pistons move out from the outboard cylinders and the pucks contact the disc. The disc slides slightly in the key slots until the inboard stationary pucks also contact the disc. The result is a fairly even amount of friction applied to each side of the disc and thus the wheel rotation is slowed.

When brake pressure is relieved, the force of the return spring is sufficient to move the piston back away from the disc, but not enough to move the adjusting pin held by the friction of the pin's grip. The piston stops when it contacts the head of the adjusting pin. Thus, regardless of the amount of wear, the same travel of the piston is required to apply the brake. The stem of the pin protruding through the cylinder head serves as a wear indicator. The maintenance manual states the minimum length of the pin that needs to protrude for the brakes to be considered airworthy. (***Figure 14-91***)

The brake caliper has passages machined into it to facilitate hydraulic fluid movement and the application of pressure when the brakes are applied. The caliper housing also contains a bleed port used by the technician to remove air from the system.

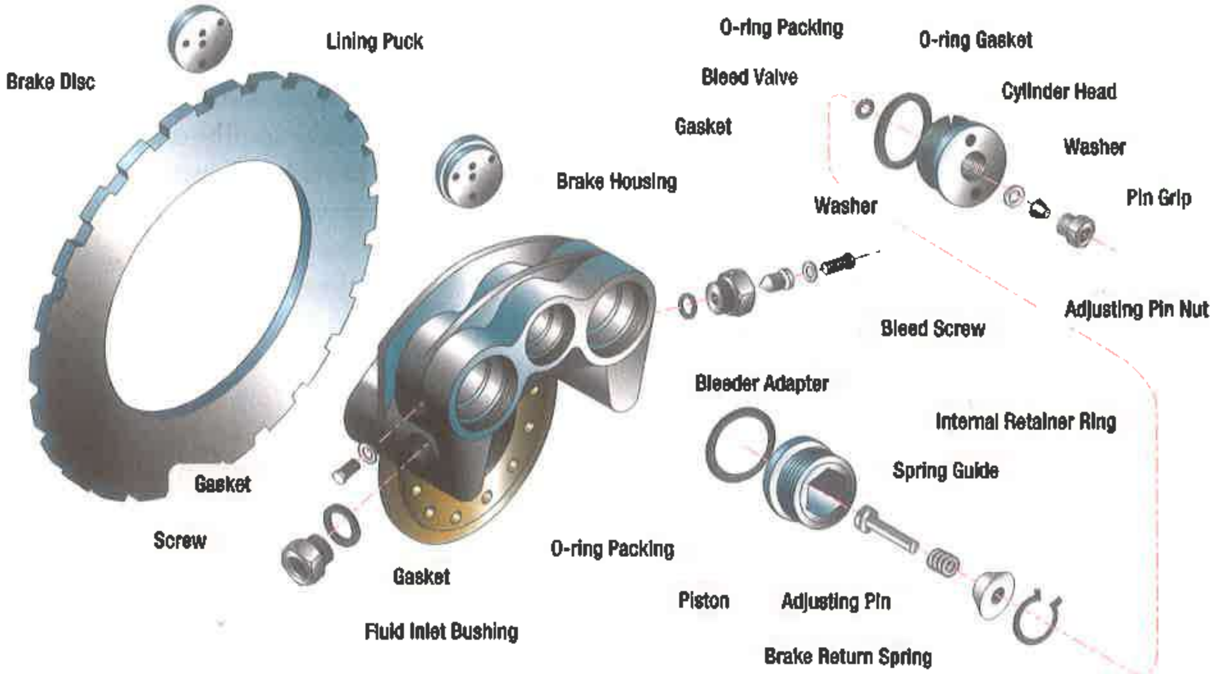


Figure 14-90. An exploded view of a single-disc brake assembly.

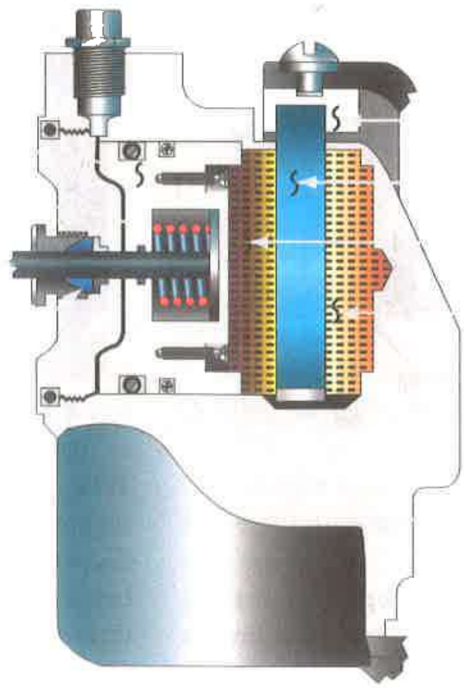
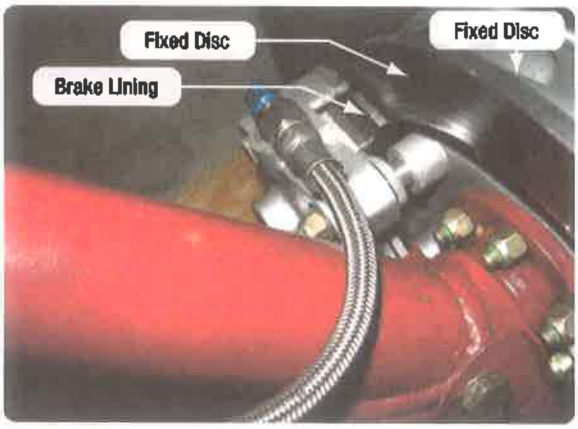


Figure 14-91. A cross-sectional view of a floating disc brake.

Fixed Disc Brakes

An alternate single disc arrangement is the fixed disc. This works by fixing the disc rigidly to the wheel and allowing the calipers to move laterally as the brake is applied. Similar to the floating arrangement, this ensures that the brake pads apply an even force to the disc throughout the braking operation. Even pressure must be applied to both sides of the brake disc to generate the required friction and obtain consistent wear properties from the brake linings. This design is common on light aircraft. An example is shown in ***Figure 14-92***.

The fixed disk, floating caliper design allows the brake caliper and linings to adjust position in relationship to the disc. Linings are riveted to the pressure plate and back plate. Two anchor bolts that pass through the pressure plate are secured to the cylinder assembly. The other ends of the bolts are free to slide in and out of bushings in the torque plate which is bolted to the axle flange. The cylinder assembly is bolted to the backplate to secure the assembly around the disc. When pressure is applied, the caliper and linings center on the disc via the sliding action of the anchor bolts in the torque plate bushings. This provides equal pressure to both sides of the disc to slow its rotation. A unique feature of the Cleveland brake is that the linings can be replaced without removing the wheel. Unbolting the cylinder assembly from the backplate allows the anchor bolts to slide out of the torque plate bushings. The entire caliper assembly is then free and provides access to all components.



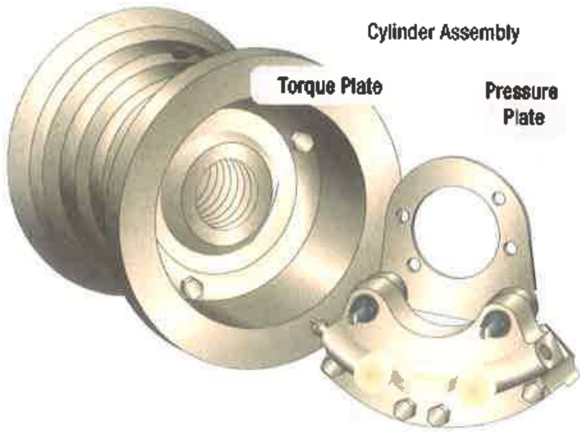


Figure 14-92. A Cleveland fixed disc brake, common on light aircraft.

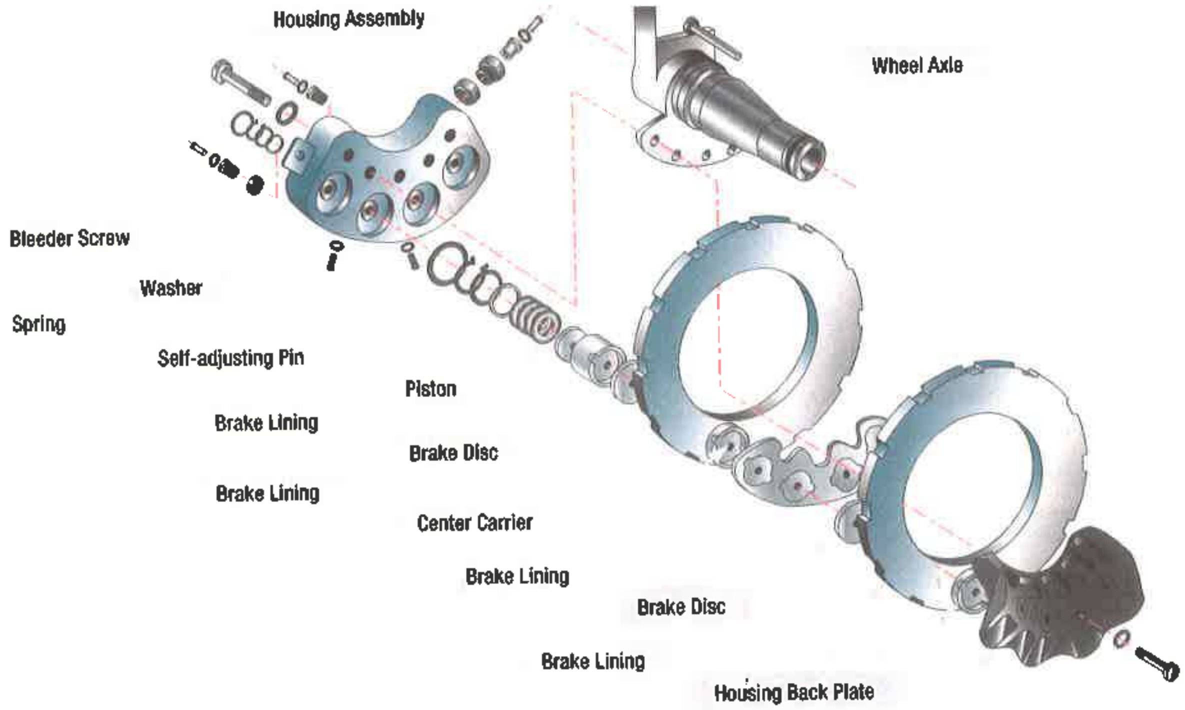


Figure 14-93. An exploded view of a dual-disc brake assembly.

Maintenance requirements on all single disc brake systems include regular inspection for any damage and for wear on the linings and discs. Replacement of parts worn beyond limits is always followed by an operational check, performed while taxiing the aircraft. The braking action for each wheel should be equal with equal application of pedal pressure. Pedal pressure should be firm, not soft or spongy. When pedal pressure is released, the brakes should release without evidence of drag.

Dual Disc Brakes

Dual disc brakes are used on aircraft where a single disc on each wheel does not supply sufficient braking friction. Two discs are keyed to the wheel instead of one. A center carrier is located between the two discs. It contains linings on each side that contact each of the discs when the brakes are applied. The caliper mounting bolts are long and mount through the center carrier, as well as the backplate which bolts to the housing assembly. (***Figure 14-93***)

Multiple Disc Brakes

A multiple brake assembly consists of an extended bearing carrier similar to a torque tube type unit that bolts to the axle flange. It supports the various brake parts, including an annular cylinder and piston, a series of steel discs alternating with copper or bronze plated discs, a backplate, and a backplate retainer. The steel stators are keyed to the bearing carrier, and the copper or bronze plated rotors are keyed to the rotating wheel. Hydraulic pressure applied to the piston causes the entire stack of stators and rotors to be compressed. This creates enormous friction and heat and slows the rotation of the wheel.

As with the single and dual disc brakes, retracting springs return the piston into the housing chamber of the bearing carrier when hydraulic pressure is relieved. The hydraulic fluid exits the brake to the return line through an automatic adjuster. The adjuster traps a predetermined amount of fluid in the brakes that is just sufficient to provide the correct clearances between the rotors and stators. Brake wear is typically measured with a wear gauge. This type of brake is found on AW189 helicopters, as shown in ***Figure 14-94*** where the main brake assembly consists of an aluminum housing containing five pistons which operate through an independent hydraulic circuit.

Carbon Brakes

In this type, carbon fiber is used to construct the brake rotors. Carbon brakes are approximately forty percent lighter than conventional types. The carbon fiber discs are noticeably thicker than steel rotors but are still extremely light. (***Figure 14-95***)

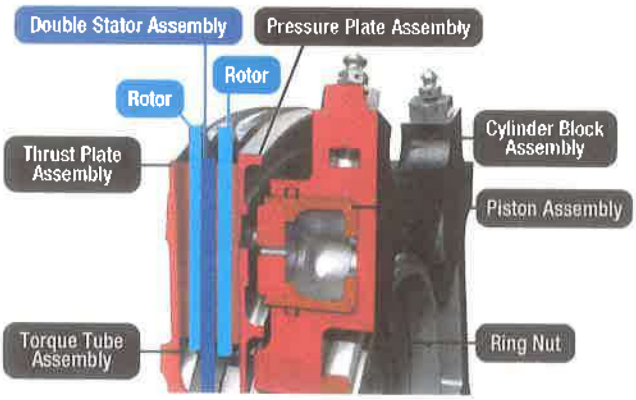


Figure 14-94. Multiple-disc brake assembly.

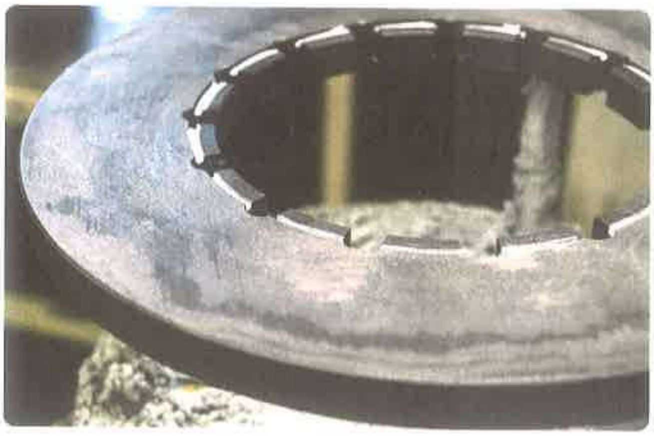


Figure 14-95. A carbon fiber brake rotor.

Carbon can withstand temperatures 50% higher than steel brake components. The maximum operating temperature is limited by the ability of adjacent components to withstand the heat. Carbon rotors also dissipate heat faster than steel rotors and can better maintain its strength and dimensions at high temperatures. Carbon brakes last twenty to fifty percent longer than steel which results in reduced maintenance. The only impediment to carbon brakes being used on all aircraft is the high cost of manufacturing.

Electric Carbon Brakes

The H160 Airbus helicopter is equipped with an electrical braking system, and fitted with carbon discs. This electric braking system is made up of two brakes and two sensors under the pilot's pedals which indicates the desired braking level to a computer) and a computer. This system has many advantages including the absence of hydraulic pipes, a reduction in weight and size of the equipment which allows more possibilities for interior fittings, and simplified maintenance throughout its use.

Brake Actuating Systems

Except on the H160 above, there are three basic actuating systems:

- Independent system, not part of the aircraft main hydraulic system.
- Booster system that uses the aircraft hydraulic system intermittently when needed.
- Power brake system that only uses the aircraft main hydraulic system.

Independent Master Cylinders

In general, light aircraft and those without hydraulic systems use independent braking systems. This stand alone system consists of a hydraulic tank, master cylinders connected to a brake pedal, and a piston in the brake caliper that actuates to apply the braking force.

Master cylinders are used to develop the necessary hydraulic pressure to operate the brakes, similar to the brake system of an automobile. In most actuating systems, the pilot pushes on the brake pedals to apply the brakes. A master cylinder for each brake is mechanically connected to the corresponding pedal. (***Figure 14-96***)

When the pedal is depressed, a piston inside a sealed chamber in the master cylinder forces hydraulic fluid through a line to the piston(s) in the brake assembly. The brake piston(s) push the brake linings against the brake rotor to create the friction that slows the wheel rotation. Pressure is increased throughout the entire systems and against the rotor as the pedal is pushed harder. Many master cylinders have built-in reservoirs for the brake hydraulic fluid. Others have a single remote reservoir that services both of the aircraft master cylinders. (***Figure 14-97***)

A Goodyear model master cylinder used with a remote reservoir is illustrated in ***Figure 14-98***. When the pedal is depressed, the piston arm moves forward into the master cylinder. It pushes the piston against the fluid which is forced through the line to the brake. When pedal pressure is released, the return springs in the brake assembly retract the brake pistons back into the brake housing. The hydraulic fluid behind the pistons is displaced and must return to the master cylinder. As it does, a return spring in the master cylinder moves the piston, piston rod and pedal back to the original position. The fluid behind the piston flows back into the reservoir. The brake is ready to be applied again. The

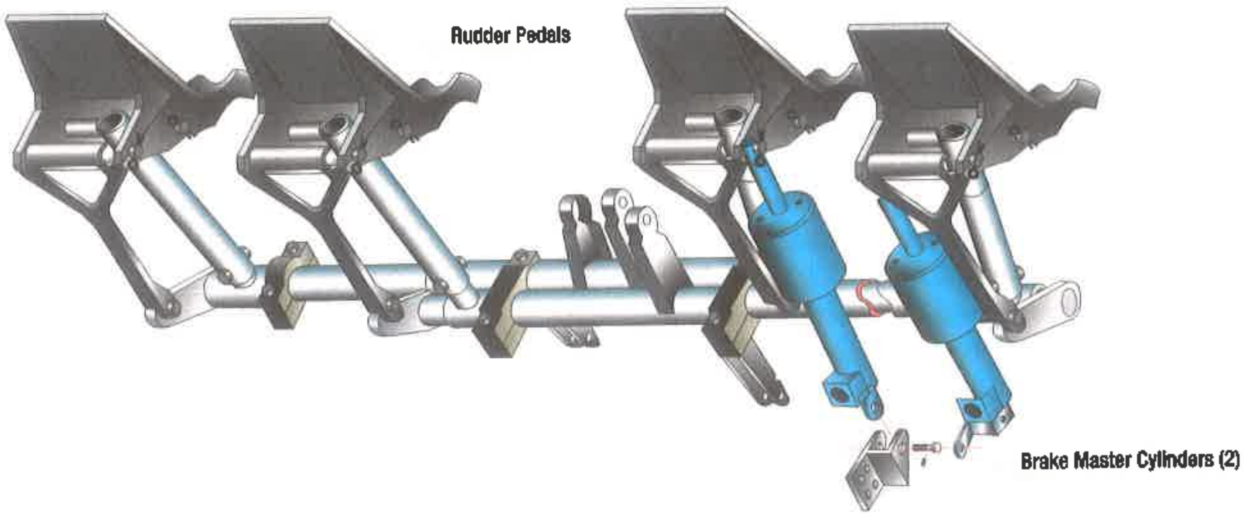


Figure 14-96. Brake master cylinders.

forward side of the piston head contains a seal that closes off the compensating port when the brakes are applied so that pressure can build. The seal is only effective in the forward direction. When the piston is returning, or fully retracted, fluid behind the piston is free to flow through piston head ports to replenish any fluid that may be lost downstream of the master cylinder. The aft end of the master cylinder contains a seal that always prevents leakage. A rubber boot fits over the piston rod and the aft end of the master cylinder to keep out dust.

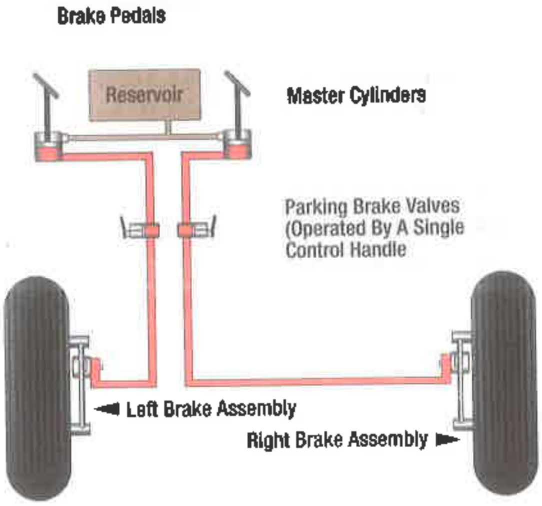


Figure 14-97. A remote reservoir system.

A parking brake for a remote reservoir master cylinder system is a ratcheting mechanical device between the master cylinder and the pedals. With the brakes applied, the ratchet is engaged by pulling the parking brake handle. To release, the rudder pedals are depressed further allowing the ratchet to disengage. With the parking brake set, any expansion of hydraulic fluid due to temperature is relieved by a spring in the mechanical linkage.

An alternative arrangement of independent braking systems incorporates two master cylinders, each with its own integral fluid reservoir. Except for the reservoir location, the brake system is basically the same as just described. The master cylinders are mechanically linked

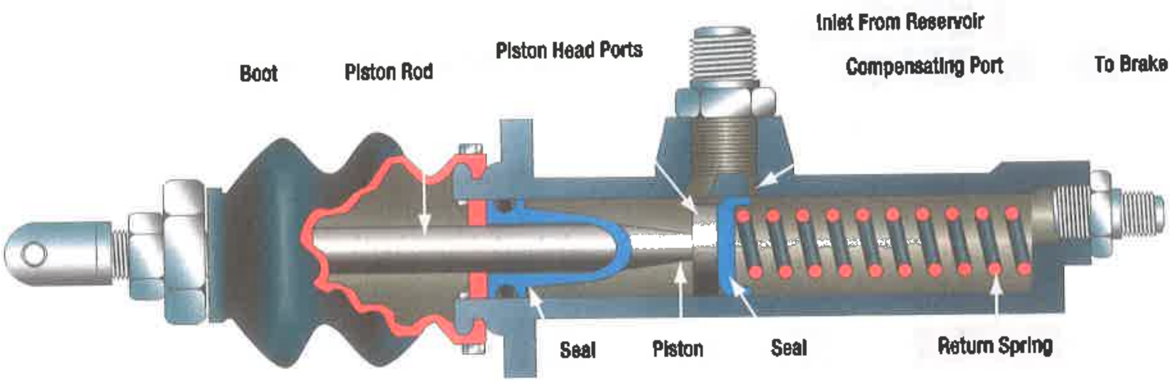


Figure 14-98. A Goodyear brake master cylinder.

to the rudder pedals. Depressing the pedal pushes the piston into the cylinder forcing fluid to the brake. The piston rod rides in a compensator sleeve and contains an O-ring that seals the rod to the piston when the rod is moved forward. This blocks the compensating ports. When released, a spring returns the piston to its original position which refills the reservoir. The rod end seal retracts away from the piston head allowing a free flow of fluid from the cylinder through the compensating ports in the piston to the reservoir. (***Figure 14-99***)

Boosted Brakes

In an independent braking system, the pressure applied to the brakes is only as great as the foot pressure applied to the pedal. Boosted actuating systems augment the force developed by the pilot with hydraulic pressure when needed, resulting in greater pressure applied to the brakes than the pilot alone can provide. A boosted brake master cylinder for each brake is mechanically attached to the brake pedals. (***Figure 14-100***)

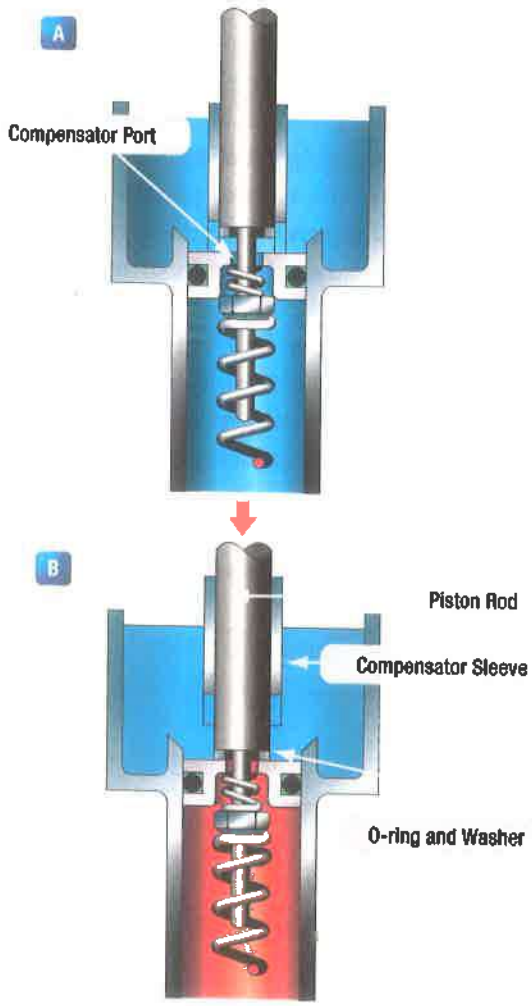


Figure 14-99. A common master cylinder.

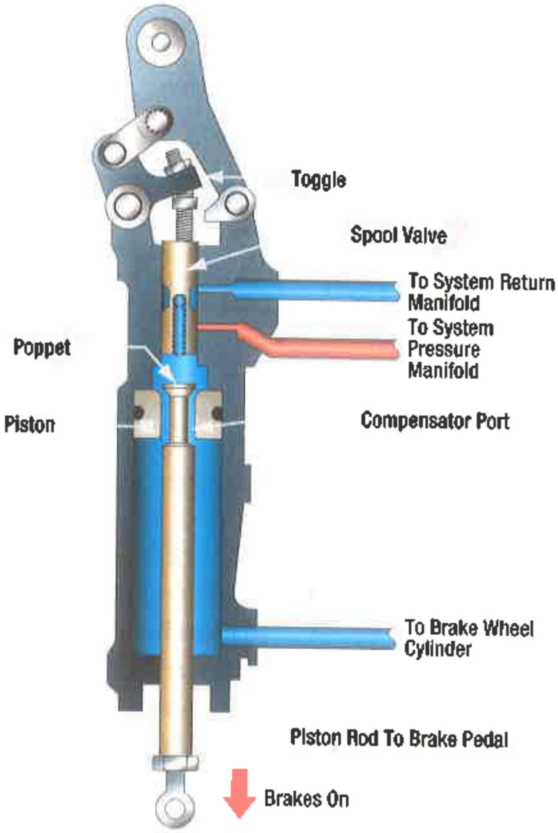


Figure 14-100. A master cylinder for a boosted system.

When the brakes are applied, the pressure from the pilot's foot through the mechanical linkage moves the master cylinder piston in the direction to force fluid to the brakes. The initial movement closes a compensator poppet used to provide thermal expansion relief when the brakes are not applied. As the pilot pushes harder on the pedal, a spring loaded toggle moves a spool valve in the cylinder. Hydraulic system pressure flows through the valve to the back side of the piston. Pressure is increased, as is the force developed to apply the brakes. When the pedal is released, the piston rod travels in the opposite direction, and the piston returns to the piston stop. The compensating poppet reopens. The toggle is withdrawn from the spool via linkages, and fluid pushes the spool back to expose the system return manifold port. System hydraulic fluid used to boost brake pressure returns through the port.

Brake Metering Valve, Hydraulic And Pressure Transducers

A brake metering valve is controlled through brake hydraulic transducers and the brake pedals. Hydraulic power moves the valve distributors that are part of the normal brake circuit. The fluid stored in each left and right brake hydraulic transducer (***Figure 14-101***)

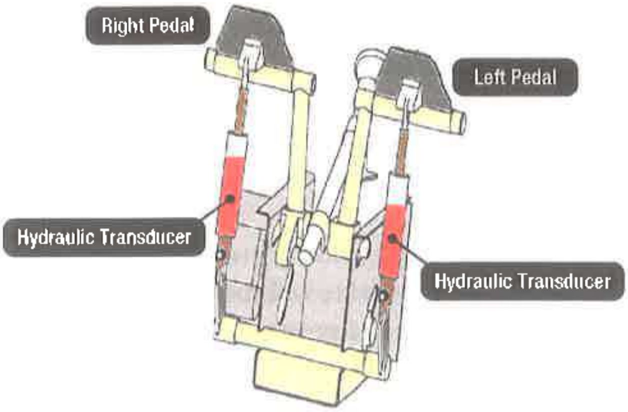


Figure 14-101. Hydraulic transducers.

is supplied to the metering valve when the pedal is operated. On each braking line a pressure transducer is installed. They are fitted in the landing gear bay to allow the monitoring of the pressure either by the pilot or the maintenance crew.

Parking Brakes

All helicopters fitted with wheels have parking brakes which operate by providing constant hydraulic pressure on the brake cylinders. It is done either by pressing the pedals completely down and then locking them with a lever or electrically by opening a valve supplying hydraulic pressure directly to the brakes. If the hydraulic system is switched off, a hydraulic accumulator supplies that pressure to the cylinders. Because of internal leakages of the servo valves, the accumulator becomes empty after some time and the brake pressure becomes minimal. To prevent this, shut off valves are installed in the return lines.

In other instances, to ensure that the braking pressure remains in the system in the case of a leak, a mechanical system, controlled by a handle in the cockpit, allows the brake to be locked on the disc with sufficient pressure to hold the helicopter. The example shown in ***Figure 14-102*** consists of a manual control handle with cable, straight push-pull type, used to actuate a wheel brake parking/emergency valve. This system is used for parking and emergency operations.

Emergency Brakes

Some power brake systems may use an emergency source of brake power that is delivered directly to the brake assemblies and bypasses the remainder of the system completely. A shuttle valve immediately upstream of the brake units shifts to accept this source when pressure is lost from the primary sources. Compressed air or nitrogen is sometimes used. A pre-charged fluid source can also be used as an alternate hydraulic source. An example of a parking/emergency wheel brake schematic is shown in ***Figure 14-103.***

With this, normal braking is done by the utility circuit which provides a differential braking capability to assist the pilot during various operations. Each wheel's brake may then be pressurized, with the applied pressure controlled through the pilot or co-pilot control circuit. The emergency braking mode may be used when the normal operating mode is no longer available as a consequence of failures.

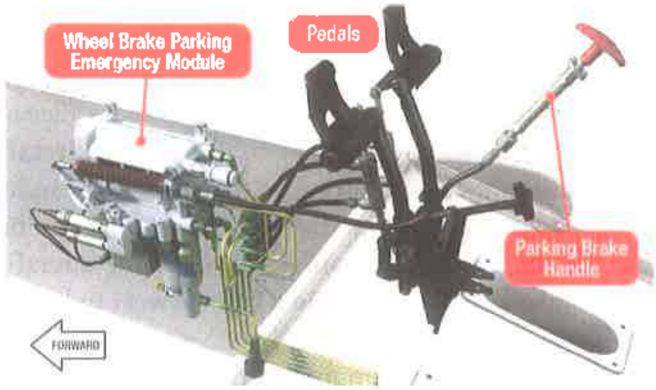


Figure 14-102. Parking brake system.

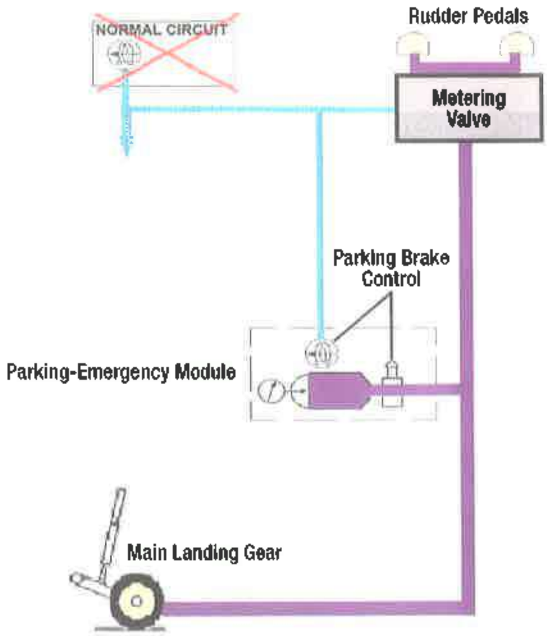


Figure 14-103. Example of parking-emergency wheel brake schematic.

Brake Inspection and Service

Brake inspection and service is important to always keep these critical components fully functional. Brake system maintenance is performed both while the brakes are installed on the aircraft and when the brakes are removed. The manufacturer's instructions must always be followed to ensure proper maintenance.

ON AIRCRAFT SERVICING

Inspection and servicing of aircraft brakes while installed on the aircraft is required. The entire brake system must be inspected in accordance with the manufacturer's instructions. Some common inspection items include brake lining wear, air in the brake system, fluid quantity level, leaks, and proper bolt torque.

Lining Wear

Brake lining material is made to wear as it causes friction during application of the brakes. This wear must be monitored to ensure it is not beyond limits and sufficient lining is available for effective braking. The aircraft manufacturer gives specifications for lining wear in its maintenance information. The amount of wear can be checked while the brakes are installed on the aircraft. Many brake assemblies contain a built in wear indicator pin. Typically, the exposed pin length decreases as the linings wear, and a minimum length is used to indicate the linings must be replaced. Caution must be taken as different assemblies may vary in how the pin is measured. On the brake described in ***Figure 14-104***, the wear pin is measured where it protrudes through the nut of the automatic adjuster on the back side of the cylinder.

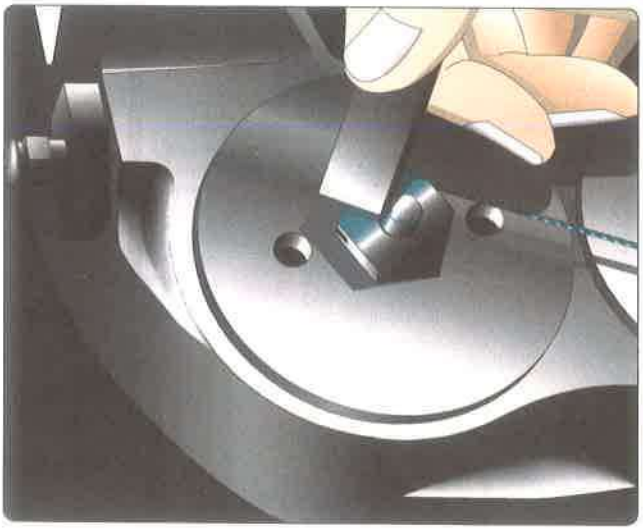


Figure 14-104. Brake lining wear.

On many other brake assemblies, the distance between the disc and a portion of the brake housing when the brakes are applied is used. As the linings wear, this distance increases. The manufacturer specifies at what distance the linings should be changed. (***Figure 14-105***)

Multiple disc brakes typically are checked for lining wear by applying the brakes and measuring the distance between the back of the pressure plate and the brake housing. (***Figure 14-106***)

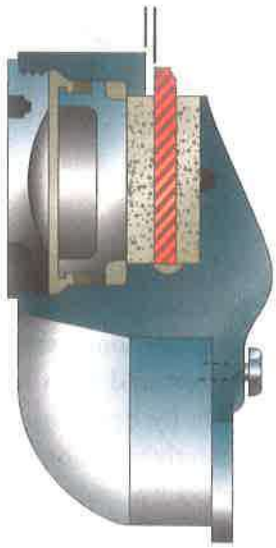


Figure 14-105. The distance between the brake disc and the brake housing.

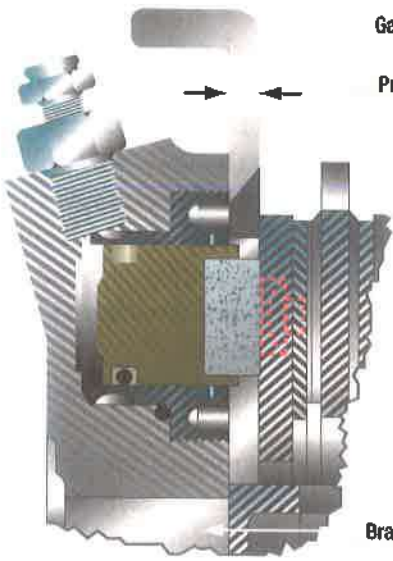


Figure 14-106. The distance between the brake housing and the pressure plate.

Regardless of the method for each brake, regular monitoring and measurement of brake wear ensures linings are replaced as they become unserviceable. Linings worn beyond limits usually require the brake assembly be removed for replacement.

Air In The System

The presence of air in the brake fluid causes the brake pedal to feel spongy. The air can be removed by bleeding to restore a firm feel. The method used is directed by each manufacturer for that type of brake system. Generally, brake systems with master cylinders may be bled by gravity or pressure bleeding methods. Follow the instructions in the aircraft maintenance manual. To pressure bleed a brake system from the bottom up, a pressure pot is used. (***Figure 14-107***)

A pressure pot is a portable tank that contains a supply of brake fluid under pressure. When dispersing fluid from the tank, air-free fluid is forced from near the bottom of the tank by the air pressure above it. The outlet hose that attaches the bleed port on the brake assembly contains a shut-off valve. Note that a similar source of pure, pressurized fluid can be substituted for a pressure tank, such as a hand pump type unit found in some hangars. The typical pressure bleed is accomplished as shown in ***Figure 14-108***.

The hose from the pressure tank is attached to the bleed port on the brake assembly. A clear hose is attached to the vent port on the aircraft brake fluid reservoir or on the master cylinder if it incorporates the reservoir. The other end of this hose is placed in a collection container with a supply of clean brake fluid covering the end of the hose. The brake's bleed port is opened. The valve on the pressure tank hose is then opened allowing air free fluid to enter the brake. Fluid containing trapped air is expelled through the hose attached to the vent port of the reservoir. The clear hose is monitored for air bubbles. When they cease to exist, the bleed port and pressure tank shutoff are closed, and the pressure tank hose is removed. The hose at the reservoir is also removed. Fluid quantity may need to be adjusted to assure the reservoir is not overfilled. Note that it is absolutely necessary that the proper fluid be used to service any brake system including when bleeding air from the brake lines.

Brakes with master cylinders may also be gravity bled from the top down. This is a process like that used

Figure 14-107. A typical brake bleeder pot or tank.

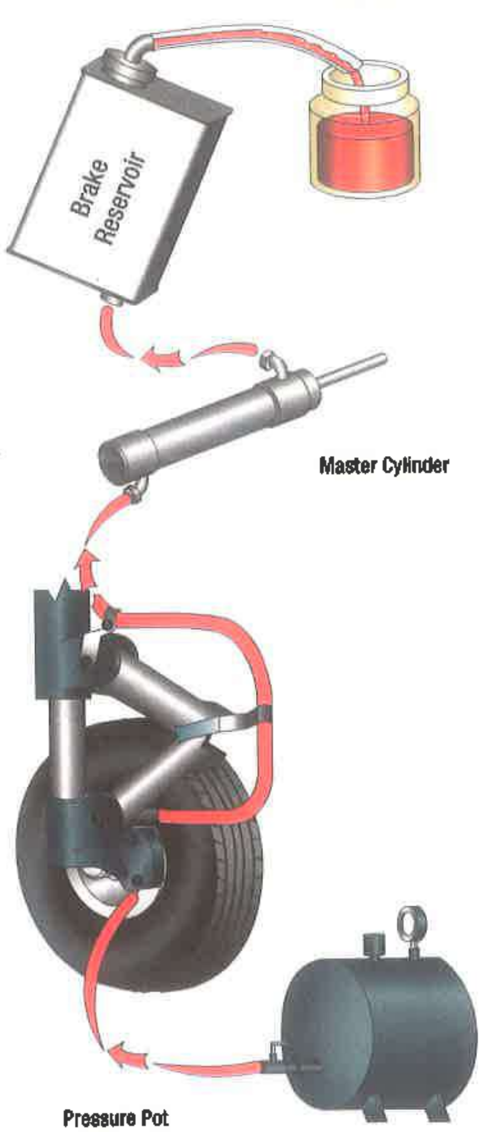


Figure 14-108. Arrangement for bottom-up pressure bleeding.

LANDING GEAR

on automobiles. (***Figure 14-109***) Additional fluid is supplied to the brake reservoir so that the quantity does not exhaust while bleeding, which would cause the reintroduction of air. A clear hose is connected to the bleed port on the brake assembly. The other end is submersed in clean fluid in a container large enough to capture fluid expelled during the bleeding process. Depress the brake pedal and open the brake assembly bleed port. The piston in the master cylinder travels all the way to the end of the cylinder forcing fluid out of the bleed hose and into the container. With the pedal still depressed, close the bleed port. Pump the brake pedal to introduce more fluid from the reservoir ahead of the piston in the master cylinder. Hold the pedal down and open the bleed port on the brake assembly. More fluid and air is expelled through the hose into the container. Repeat this process until the fluid exiting the brake through the hose no longer contains any air. Tighten the bleed port fitting and ensure the reservoir is filled to the proper level. Whenever bleeding the brakes, ensure that reservoirs and bleed tanks remain full during the process. Use only clean, specified fluid. Always check the brakes for proper operation, leaks and that the fluid level is correct when bleeding is complete.

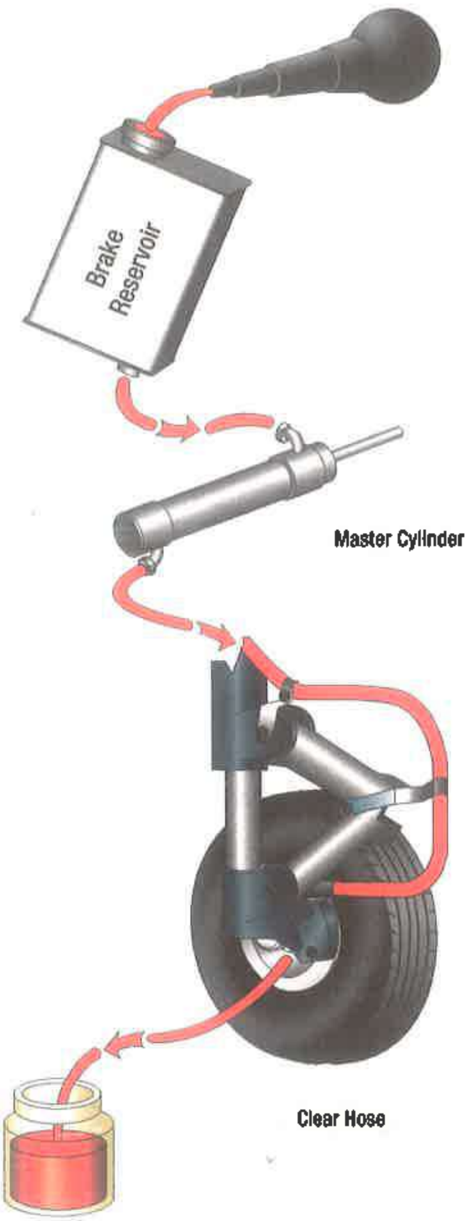


Figure 14-109. Arrangement for top down or gravity bleeding.

Fluid Quantity And Type
It is imperative that the correct hydraulic fluid is used in each brake system. Seals in the brake system are designed for a particular hydraulic fluid. Deterioration and failure occurs when they are exposed to other fluids. Mineral based fluid should never be mixed with phosphate ester based synthetic fluid. Contaminated brake or hydraulic systems must have all the fluid evacuated and all seals replaced before the aircraft is released for flight. Fluid quantity is also important. The technician is responsible for determining the method used to ascertain when the brake and hydraulic systems are fully serviced.

Inspection For Leaks
Aircraft brake systems should maintain all fluid levels and should not leak. Any evidence of a leak must be investigated for its cause. Many leaks are found at brake system fittings. While this type of leak may be fixed by tightening a loose connection, you are cautioned against over tightening fittings. Over tightening of fittings can cause damage and make the leak worse. Replace all fittings suspected of damage. Once any leak is repaired, the brake system must be re-pressurized and tested for function as well as to ensure the leak no longer exists. Occasionally, a brake housing may seep fluid through the housing body. Consult the manufacturer's manual for limits and remove any assembly that seeps excessively.

Proper Bolt Torque
The stresses experienced by the landing gear and brake system requires that all bolts are properly torqued. Bolts used to attach the brakes to the strut typically have the required torque specified in the manufacturer's manual. Check for torque specifications that may exist for all landing gear and brake bolts and ensure they are properly tightened. Whenever applying torque to a bolt on an aircraft, the use of a calibrated torque wrench is required.

OFF-AIRCRAFT BRAKE SERVICING AND MAINTENANCE

Certain servicing and maintenance of an aircraft brake assembly is performed while it has been removed from the aircraft. A close inspection of the assembly and its many parts should be performed at this time. Some of the inspection items on a typical assembly are:

Bolt And Threaded Connections

All bolts and threaded connections are inspected. They should be in good condition without signs of wear. Self locking nuts should still retain their locking feature. The hardware should be what is specified in the brake manufacturer's parts manual. Many aircraft brake bolts, for example, are not standard hardware and may be of closer tolerance or made of a different material. The demands of the high stress environment in which the brakes perform may cause brake failure if improper hardware is used. Be sure to check the condition of all threads and O-ring seating areas machined into the housing. The fittings threaded into the housing must also be checked for condition.

Discs

Both rotating and stationary discs can wear. Uneven wear can indicate that the automatic adjusters may not be pulling the pressure plate back far enough to relieve all pressure on the disc stack. Stationary discs are inspected for cracks. Cracks usually extend from the relief slots if so equipped. On multiple disc brakes, the slots that key the disc to the torque tube must also be inspected for wear and widening. The discs should engage the torque tube without binding. The maximum width of the slots is given in the maintenance manual. Cracks or excessive key slot wear are grounds for rejection.

Brake pads must also be inspected for wear while the brake assembly is removed from the aircraft. Signs of uneven wear should be investigated and the problem corrected. The pads may be replaced if worn beyond limits as long as the stationary disc upon which they mount passes inspection. Rotating discs must be similarly inspected. The general condition of the disc must be observed. Glazing can occur when a disc is overheated. It causes brake squeal and chatter. It is possible to resurface a glazed disc if the manufacturer allows.

Rotating discs must also be inspected in the drive key slot or tang area for wear or deformation. Little damage is allowed before replacement is required. The pressure plate and back plate on multiple disc brakes must be inspected for freedom of movement, cracks, general condition, and warping. New linings may be riveted to the plates if the old linings are worn and the condition of the plate is good.

NOTE: Replacing brake pads and linings by riveting may require specific tools and technique as described in the maintenance manual to ensure secure attachment. Minor warping can be straightened on some brake assemblies.

Automatic Adjuster Pins

A malfunctioning automatic adjuster assembly can cause the brakes to drag on the rotating disc(s) by not fully releasing and pulling the lining away from the disc. This can lead to excessive, uneven lining wear and glazing. The return pin must be straight with no surface damage so it can pass through the grip without binding. Damage under the head can weaken the pin and cause failure. Magnetic inspection is sometimes used to inspect for cracks. The components of the grip and tube assembly must be in good condition. Clean and inspect in accordance with the manufacturer's instructions. The grip must move with the force specified and must move through its full range of travel.

Torque Tube

A sound torque tube is necessary to stabilize the brake assembly on the landing gear. General visual inspection should be made for wear, burrs, and scratches. Magnetic particle inspection is used to check for cracks. The key areas should be checked for dimension and wear. All limits of damage are referenced in the manufacturer's maintenance data. The torque tube should be replaced if a limit is exceeded.

Brake Housing And Piston Condition

The brake housing must be inspected thoroughly. Scratches, gouges, corrosion, or other blemishes may be dressed out and the surface treated to prevent corrosion. Minimal material should be removed when doing so. Most important is that there are no cracks in the housing. Fluorescent dye penetrant is typically used to inspect for cracks. If a crack is found, the housing must be replaced. The cylinder area of the housing must be dimensionally checked for wear. Limits are specified in the manufacturer's maintenance manual.

The pistons that fit into the cylinders in the housing must be checked for corrosion, scratches, burrs, etc. Pistons are also dimensionally checked for wear limits specified in the maintenance data. Some pistons have insulators on the bottom. They should not be cracked and should be of a minimal thickness. A file can be used to smooth out minor irregularities.

Seal Condition

Without properly functioning seals, brake operation will be compromised, or the brakes will fail. Over time, heat and pressure mold a seal into the seal groove and harden the material. Eventually, resilience is reduced and the seal leaks. New seals should be used to replace all seals in the assembly. Acquire seals by part number in a sealed package from a reputable supplier to avoid bogus seals and to ensure the correct seals for the brake assembly in question. Check to ensure the new seals have not exceeded their shelf life, which is typically three years from the cure date. Many brakes use backup rings in the seal groove to support the O-ring seals and reduce the tendency of the seal to extrude into the space which it is meant to seal. These are often made of Teflon or similar material. Backup seals are installed on the side of the O-ring away from the fluid pressure. They are often reusable. (***Figure 14-110***)

BRAKE MALFUNCTIONS AND DAMAGE

A couple of common brake problems are discussed in this section.

Overheating

Excessive heat can damage and distort brake parts, weakening them to the point of failure. When a brake shows signs of overheating, it must be removed from the aircraft and inspected for damage. When an aircraft is involved in an aborted takeoff, the brakes must be removed and inspected to ensure they withstood this high level of use. A typical post overheat inspection involves removal of the brake from the aircraft and disassembly. All the seals must be replaced. The housing must be checked for cracks, warping, and hardness per the maintenance manual. The brake discs must also be inspected. They must not be warped, and the surface treatment must not be damaged or transferred to an adjacent disc. Once reassembled, the brake should be bench tested for leaks and pressure tested for operation before being installed on the aircraft.

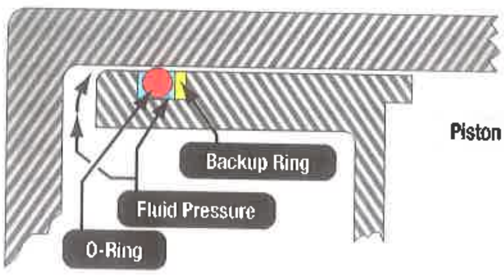


Figure 14-110. Back-up rings.

Dragging

Brake drag is caused by the linings not retracting from the brake disc when the brakes are no longer being applied. Brakes that drag are always essentially partially on. This can cause excessive lining wear and overheating leading to damage to the disc(s). A brake may drag when the return mechanism is not functioning properly. This could be due to a weak return spring, the return pin slipping in the auto adjuster pin grip, or similar malfunction. Inspect the auto adjuster(s) and return units on the brake when dragging is reported.

An overheated brake that has warped the disc also causes brake drag. Remove the brake and perform a complete inspection as discussed in the previous section. Air in the brake fluid line can also cause drag. Heat causes the air to expand, which pushes the brake linings against the disc prematurely. If no damage has been caused when reported, bleed the brakes to remove the air from the system to eliminate the drag. At all times, the technician should perform inspections to ensure the proper parts are used in the assembly. Improper parts, especially in the retraction/adjuster assemblies, can cause the brakes to drag.

Chattering Or Squealing

Brakes may chatter or squeal when the linings do not ride smoothly and evenly along the disc. A warped disc(s) in a multiple brake disc stack produces a condition wherein the brake is applied and removed many times per minute. This causes chattering, and at high frequency causes squealing. Any misalignment of the disc stack out of parallel causes the same. Discs that have been overheated may have damage to the surface layers. Some of this mix may be transferred to the adjacent disc resulting in uneven disc surfaces that also leads to chatter or squeal. In addition to the noise produced by brake chattering and squealing, vibration is caused that may lead to further damage of the brake and the gear system. The technician must investigate all reports of brake chattering and squealing.

STEERING

Nose Wheel Steering

Any wheel type landing gear needs steering, either actively by providing steering jacks in the nose gear, or passively with a freely swiveling nose wheel with yaws by applying tail rotor thrust. Active steering calls for a steering jack on nose(or tail) landing gear, a steering manifold which controls hydraulic supply to the jacks, cockpit controls and indicators.

In the most simplistic of installations, the nose wheel is free castering; meaning it swivels with no mechanism incorporated with which to actively steer the wheel. By using differential braking (the differential braking pressure on each wheel allows the dynamic steering during taxiing) and the main rotor to provide steering, the wheel will automatically spin and allow the helicopter to change direction on the ground. A centering system allows the wheel to be centered after takeoff for the retraction of the landing gear or the next landing.

Figure 14-111 shows a nose landing gear free swiveling without an active steering motor. The rotation capability of the nose landing gear is also used for towing and taxiing procedures. Other steering systems exist that are electrically controlled and hydraulically actuated. The nose gear is steered by an electrically controlled, hydraulic powered cylinder which is mounted on the nose gear recoil strut. The cylinder is connected through mechanical linkage to an eccentrically mounted drive stud on the recoil strut inner cylinder.

In a recent system, sensors measure the position of the nose wheel. A controller receives signals from the steering commands of the pilot. (***Figure 14-112***) The electronic signals from the controller are translated into hydraulic pressure, which in turn steers the nose wheel.

Shimmy Dampers

Nose gear have a tendency to oscillate. Torque links attached from the stationary upper cylinder of a nose wheel strut to the bottom movable cylinder or piston are not sufficient to prevent most nose gear from the tendency to oscillate (or shimmy), at certain speeds. This vibration wheel shimmy must be controlled through hydraulic damping. The damper can be built integrally within the nose gear, but most often it is an external unit attached between the upper and lower shock struts. It is active during all phases of ground operation while permitting the nose gear steering system to function normally.

Piston Type

Aircraft not equipped with hydraulic nose wheel steering utilize an additional external shimmy damper. The case is attached firmly to the upper shock strut cylinder. The shaft is attached to the lower shock strut cylinder and to a piston inside the damper. As the lower strut cylinder tries to shimmy, hydraulic fluid is forced through a bleed hole in the piston. The restricted flow through the bleed hole dampens the oscillation. (***Figure 14-113***)

Figure 14-111. Free swiveling nose landing gear.

Figure 14-112. Nose wheel steer controller.

A piston type shimmy damper may contain a fill port to add fluid, or it may be a sealed unit. Regardless, the unit should be checked for leaks regularly. To ensure proper operation, a piston type hydraulic shimmy damper should be filled to its capacity.

Vane Type

A vane type shimmy damper uses fluid chambers created by the vanes separated by a valve orifice in a center shaft. (***Figure 14-114***) As the nose gear tries to oscillate, vanes rotate to change the size of internal chambers filled with fluid. The chamber size can only change as fast as the fluid can be forced through the orifice. Thus, the gear oscillation is dissipated by the rate of fluid flow. An internal spring loaded replenishing reservoir keeps pressurized fluid in the working chambers. A thermal compensation of the orifice size is included. As with the piston type shimmy damper, the vane damper should be inspected for leaks. A fluid level indicator protrudes from the reservoir end of the unit.

Non-Hydraulic Shimmy Damper

Non-hydraulic shimmy dampers look and fit like piston type shimmy dampers but contain no fluid. In place of the metal piston, a rubber piston presses against the inner diameter of the damper housing when shimmy motion is received through the shaft. The piston rides on a thin film of grease and the rubbing action between the piston and the housing provides the damping. This is known as surface effect damping. The materials used to construct this type of damper provide a long service life without the need to ever add fluid. (***Figure 14-115***)

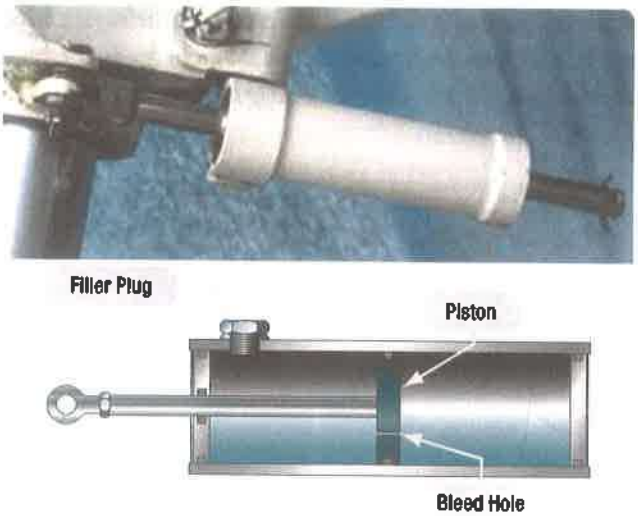


Figure 14-113. A piston-type shimmy damper.

Figure 14-115. A non-hydraulic shimmy damper.

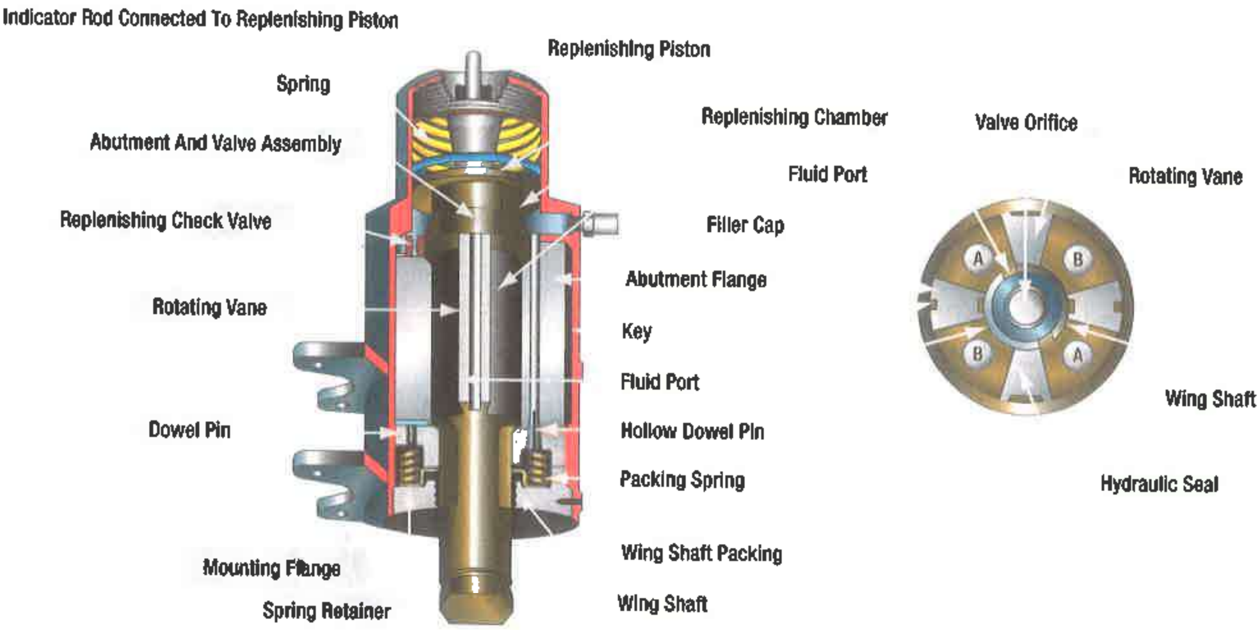


Figure 14-114. A typical vane-type shimmy damper.

LANDING GEAR SYSTEM MAINTENANCE

The moving parts and dirty environment of the landing gear make this an area of regular maintenance. Because of the stresses and pressures acting on the gear, inspection, servicing, and other maintenance becomes a continuous process. The most important job in the maintenance of the aircraft landing gear system is thorough and accurate inspections. To properly perform inspections, all surfaces should be cleaned to ensure that no trouble spots are undetected.

Periodically, it is necessary to inspect shock struts, trunnion and brace assemblies and bearings, shimmy dampers, wheels, wheel bearings, tires, and brakes. Landing gear position indicators, lights, and warning horns must also be checked. During all inspections and visits to the wheel wells, ensure all ground safety locks are installed.

Other landing gear inspection items include checking emergency control handles and systems for proper position and condition. Inspect landing gear wheels for cleanliness, corrosion, and cracks. Check wheel tie bolts for looseness. Check tires for wear, cuts, deterioration, the presence of grease or oil, alignment of slippage marks, and proper inflation. Inspect landing gear mechanisms for condition, operation, and proper adjustment. Lubricate the landing gear, including the nose wheel steering. Check steering system cables for wear, broken strands, alignment, and safetying. Inspect shock struts for cracks, corrosion, breaks, and security. Where applicable, check brake clearances and wear.

Various types of lubricant are required on various points of friction and wear. Specific products to be used are given by the manufacturer in the maintenance manual. Lubrication may be accomplished by hand or with a grease gun. Before applying grease to a pressure grease fitting, be sure the fitting is wiped clean of dirt and debris, as well as old hardened grease. Dust and sand mixed with grease produce a very destructive abrasive compound. Wipe off all excess grease while greasing. The piston rods of all exposed strut cylinders and cylinders should be clean.

Landing Gear Rigging and Adjustment

Occasionally, it becomes necessary to adjust the landing gear switches, doors, linkages, latches, and locks to ensure proper operation. When landing gear actuating cylinders are replaced and when length adjustments are made, over travel must be checked. Over travel is the action of the cylinder piston beyond the movement necessary for landing gear extension and retraction. This additional action operates the landing gear latch mechanisms.

A wide variety of landing gear designs result in procedures for rigging and adjustment that vary from aircraft to aircraft. Uplock and downlock clearances, linkage adjustments, limit switch adjustments, and other adjustments must be confirmed by the technician in the manufacturer's data before acting. The following examples of various adjustments convey concepts, rather than actual procedures for any particular aircraft.

Adjusting Landing Gear Latches

The adjustment of various latches is a primary concern to the aircraft technician. Latches are generally used to hold the gear up or down and/or to hold the gear doors open or closed. Despite variations, all latches are designed to do the same thing. They must operate automatically at the proper time, and they must hold the unit in the desired position.

Many gear up latches operate similarly. Clearances and dimensional measurements of rollers, shafts, bushings, pins, bolts, etc., are common. In ***Figure 14-116***, the landing gear door is held closed by two latches. Both latches must grip and hold the door tightly against the aircraft structure. These components include a hydraulic latch cylinder, a latch hook, spring-loaded crank-and-lever linkage, and the latch hook. When hydraulic pressure is applied, the cylinder operates the linkage to engage or disengage the hook with the roller on the gear door.

In the gear down sequence, the hook is disengaged by the spring load on the linkage. In the gear up sequence, when the closing door is in contact with the latch hook, the cylinder operates the linkage to engage the hook with the door roller. Cables on the emergency extension system are connected to the sector to permit emergency release of the rollers. An uplock switch is actuated by each latch to provide a gear up indication in the flight deck. With the gear up and the door latched, inspect the roller for proper clearance as shown in ***Figure 14-117A***. If the roller is not within tolerance, it may be adjusted by loosening its mounting bolts and raising or lowering

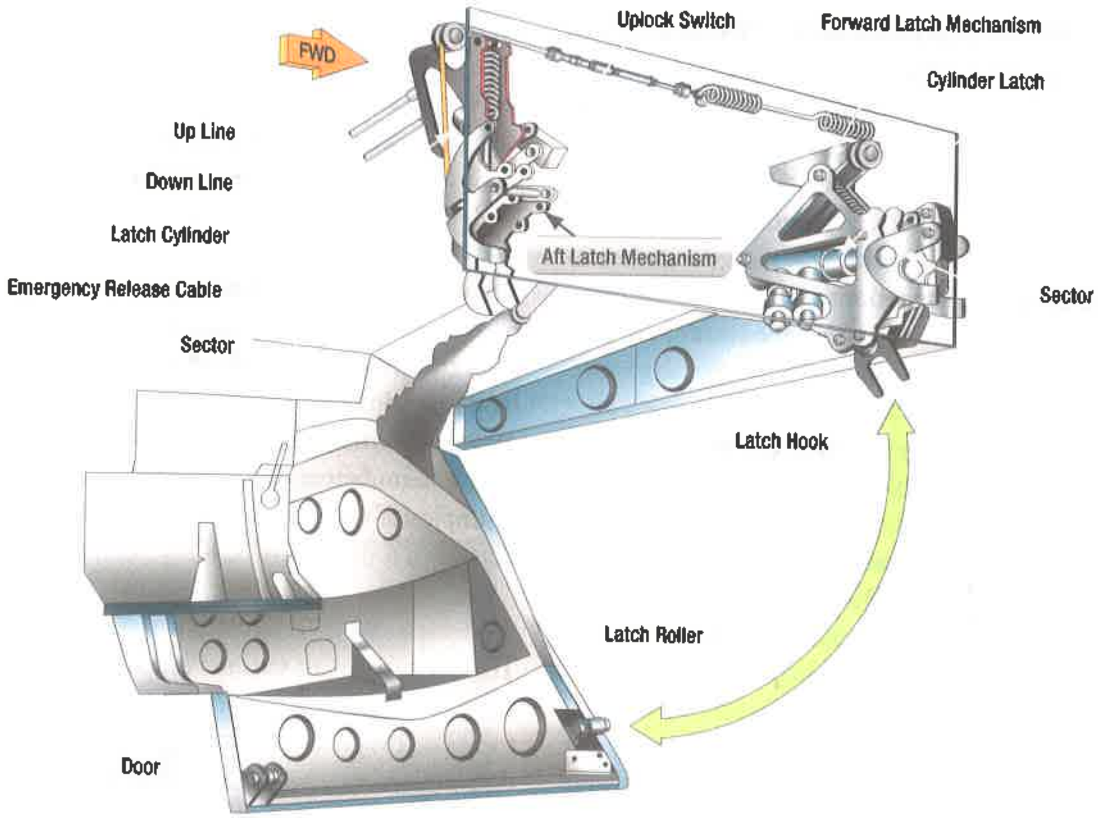


Figure 14-116. An example of a main landing gear door latch mechanism.

the latch roller support. This is accomplished via the elongated holes and serrated locking surfaces of the latch roller support and serrated plate. (*Figure 14-117B*)

Drag And Side Brace Adjustment

Each landing gear has specific adjustments and tolerances per the manufacturer that permit the gear to function as intended. A common geometry used

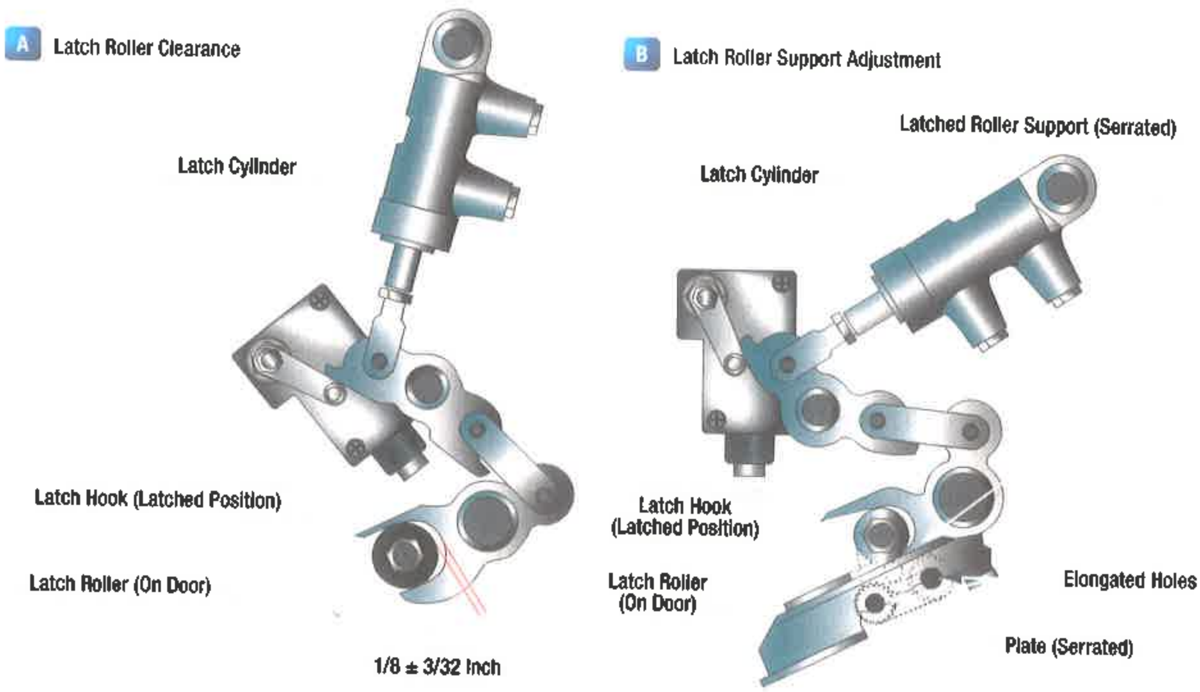


Figure 14-117. Main landing gear door latch roller clearance measurement and adjustment.

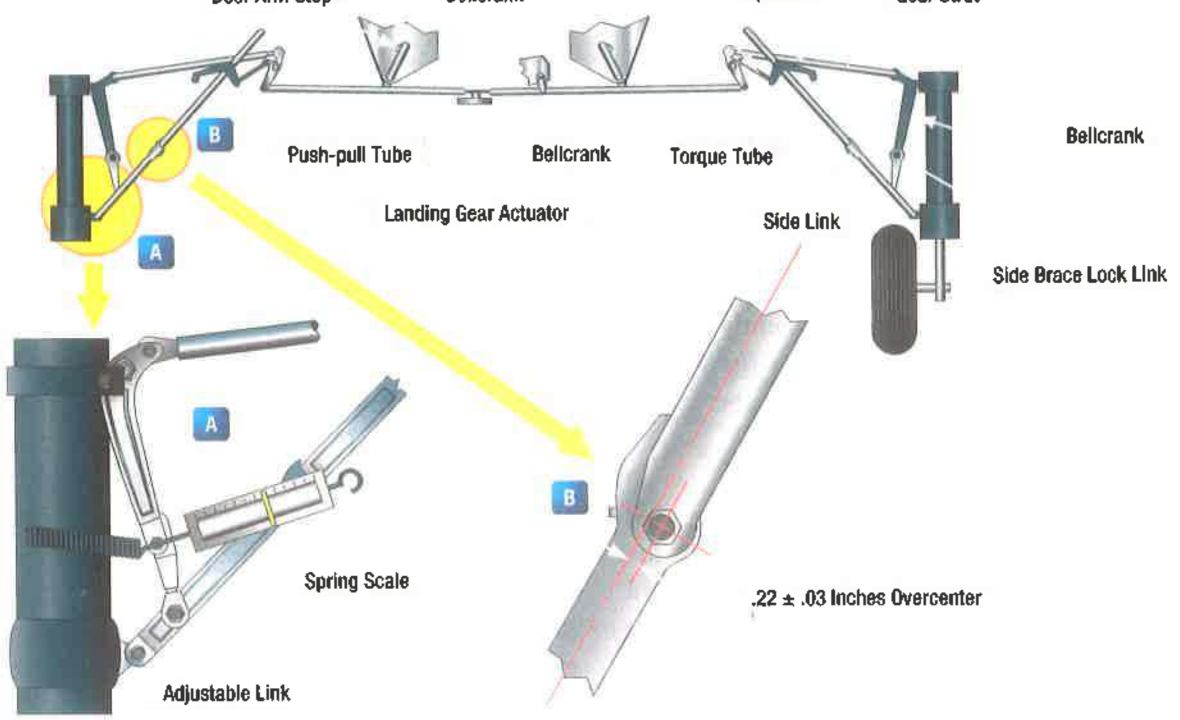


Figure 14-118. Over center adjustments.

to lock a landing gear in the down position involves a collapsible side brace that is extended and held in an over center position by means of a locking link. Springs and actuators may also contribute to the motion of the linkage. Adjustments and tests are needed to ensure proper operation.

Figure 14-118 illustrates a landing gear on a small aircraft with such a side brace. It consists of an upper and lower link hinged at the center that permits the brace to jackknife during retraction of the gear. The upper end pivots on a trunnion attached to the structure in the wheel well overhead.

The lower end is attached to the shock strut. A locking link is incorporated between the upper end of the strut and the lower drag link. It is adjustable to provide the correct amount of over center travel of the side brace links. This locks the gear securely in the down position to prevent its collapse. To adjust the over center position of the side brace locking link, the aircraft must be placed on jacks. With the landing gear in the down position, the lock link end fitting is adjusted so that the side brace links are held firmly over center. When the gear is held inboard from the down and locked position and then released, the gear must free fall into the locked down position.

In addition to the amount that the side brace links are adjusted to travel over center, down lockspring tension must also be checked. This is accomplished with a spring scale. The tension on this gear is between 18 and 27 kg (40 and 60 pounds). Check the manufacturer's maintenance data for each aircraft to ensure correct tensions exist and proper adjustments are made.

AIR-GROUND SENSING

Air/ground sensing is a warning system designed to prevent various systems from operating inappropriately on the ground or in flight. It also ensures systems are enabled or disabled as appropriate to the aircraft's situation.

Air/ground sensing generally includes a detection device on the aircraft to confirm whether the aircraft is on the ground or in the air. One way to detect this is to use a sensor on the wheels to provide information about their position and weight bearing status. Other options are to compute the position of the landing gear from other data. Some systems use a combination of approaches.

Most air/ground sensing is done by means of a switch or sensor on the landing gear. The switch opens or closes whenever the aircraft weight is transferred to, or removed from the gear. For this reason, these switches are referred to as squat switches or proximity sensors, as previously seen in ***Figure 14-49*** and ***Figure 14-50***, or as Weight On Wheels (WOW) sensors. (***Figure 14-119***)

Proximity switches and sensors are considered a safety critical part of landing gear control systems. They provide reliable information on the status of the gear. They can help to ensure that aircraft landing gear remains stowed while the aircraft is in the air and deploys only when it is needed for landing. WOW switches frequently prevent accidents relating to the incorrect position of the landing gear, as well as protect the aircraft from damage. (***Figure 14-120***)

When maintaining an aircraft, it is important to avoid a landing gear retraction. When the aircraft is on the ground, its weight compresses the shock absorber and actuates a sensor whose function is to electrically isolate the up and down lever of the landing gear. On the ground, no command is sent to the hydraulic actuator to avoid errors and dangerous operations. After takeoff, the shock absorber is fully extended and the switch status changes, allowing the landing gear to retract if the front wheel centering sensor authorizes this action. It is important to note that in case of a contact failure the aircraft is grounded by default.

Figure 14-119. Main landing gear with Weight On Wheels (WOW) sensors.

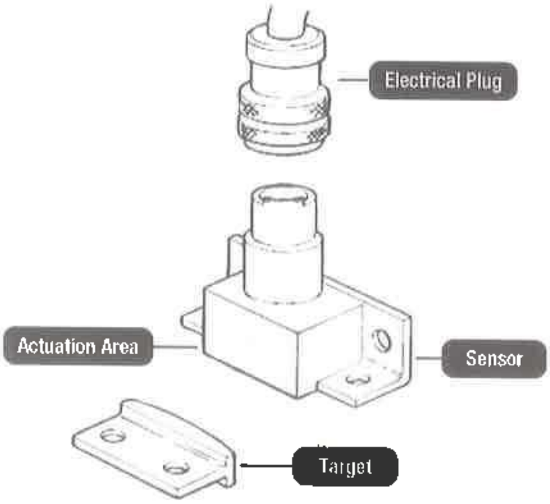


Figure 14-120. Proximity sensor.

SKIDS AND FLOATS

SKIDS

Skids are used mainly because they weigh less than wheels. (***Figure 14-121***) The benefit is to simplify the technology to reduce weight, complexity and maintenance costs and they are easy to replace if damaged. Skids are flexible and can absorb impact of a hard landing or crumple to absorb energy, preventing potentially deadly ground resonance by offering a good ground clearance. They also provide a large and stable support for pinnacle or slope landings. The downsides to skids are:

- Do not allow the helicopter to be easily moved or towed on the ground; hence must be fitted with towing dollies or placed on a towing trailer.
- Cannot be retracted and faired, thus adding more drag in flight.
- Easier to snag on obstacles and wires during surface operations.
- More susceptible to side loads during takeoff and landing, which can cause rollovers.

Helicopter skids are typically constructed of tubular steel and are secured either with bolts or a combination of bolts and tipped pins and clevis pins. (***Figure 14-122***)

Figure 14-121. Helicopter with skids.

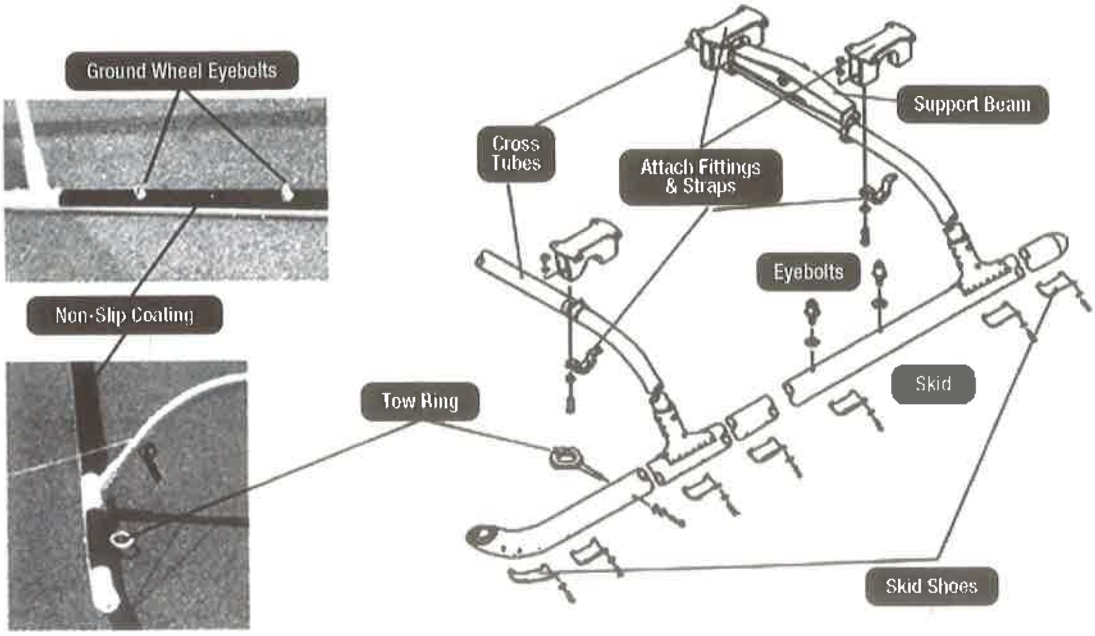


Figure 14-122. Skid gear attachment.

Skids And Shock Absorbers

To increase the landing shock absorption, hydraulic shock absorbers are often connected between the cross tubes and the structure. Wear plates are fitted under the skids to absorb the wearing due to the friction. (***Figure 14-123***)

The second function of the shock absorber is to avoid ground resonance. A flexible steel blade "stretched" downwards and extending to the rear of the skids, increases the flexibility of the skid and locates the proper frequency of the assembly in such a way that under no circumstances can ground resonance occur. In addition, the shock absorbers, interposed between the soft front leg of the undercarriage and the structure, have the role of absorbing vibratory energy and thus avoid any divergence of the oscillations. The cross tubes, skids and steel blade reduce the vertical deceleration of the helicopter during landing. The impact energy is absorbed by the shock absorbers as well as by the friction of the shoes on the ground. (***Figure 14-124***)

Skid Dampeners

Hydraulic dampers are attached from the cross tubes to the front bulkhead of the body assembly to control the rate of leg distortion. The skids are equipped with skid shoes at the front and rear. In addition to the skid shoes, a long steel strip attached to the skids is bent downward. This is a vibration dampener and is used to eliminate the possibility of ground resonance upon landing.

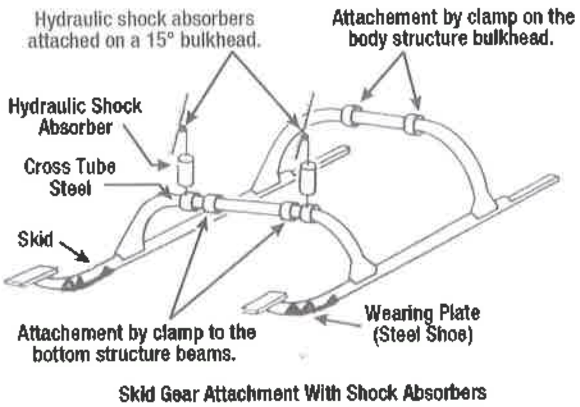


Figure 14-123. Skids with shock absorbers.

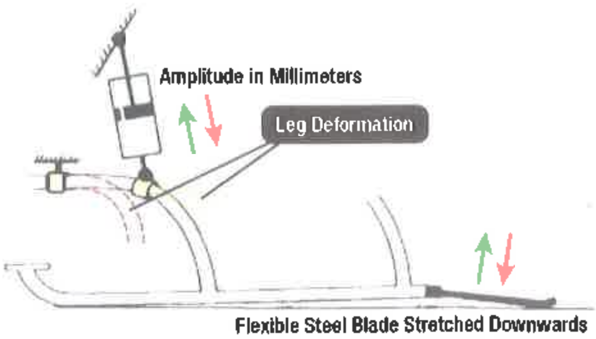


Figure 14-124. Skids with energy absorption.

Skid and Wheels

To move the helicopter on the ground, additional wheels can be installed and used by the technician, or the helicopter can land on a cart that can be moved electrically to move the helicopter in and out of the hangar. (***Figure 14-125***)

Figure 14-125. Wheels on skids.

FLOATS

There are two main types of flotation systems available:

- Fixed Utility Floats
- Emergency Pop-Out Floats

Each type has a very different purpose and depending on what the helicopter operator requires will depend on which type of flotation system is installed.

Fixed Utility Floats

Utility floats are fitted to an aircraft when work over water or swampy areas requires the temporary use of floats. These floats are designed to allow the aircraft to land and shut down without the chance of the helicopter overturning. These floats are easily filled using a simple air pump with pressure relief valves to prevent over pressurization when altitude or temperature changes.

Although fixed utility floats are great for specialized work, depending on the model they can limit the speed of the helicopter and the weight it can carry inside. Sling loads are prohibited from being carried on the belly hook while they are fitted. Fixed floats may be of the skid-on-float or the float-on-skid design. (***Figure 14-126***)

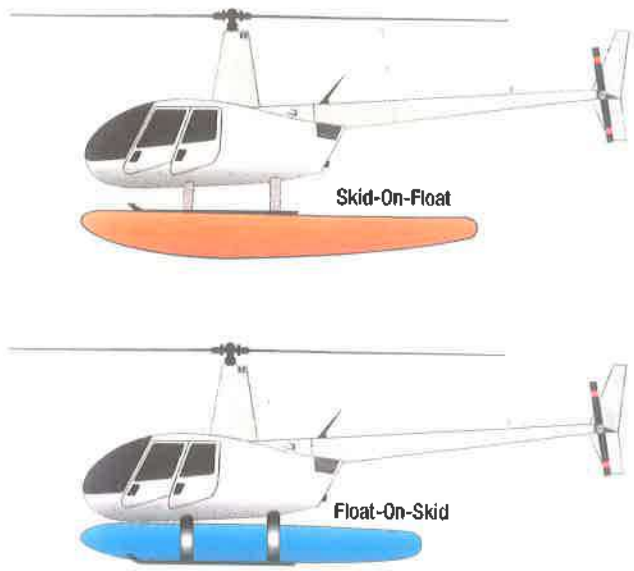


Figure 14-126. Fixed floats on a helicopter.

A skid-on-float landing gear has no rigid structure in or around the float. The float rests on the hard surface and supports the weight of the helicopter. With this type, be aware of differences in float pressure. While the pressures are usually low, a substantial difference can cause the helicopter to lean while on a hard surface making it more susceptible to dynamic rollover.

A float-on-skid landing gear has modified skids that support the weight of the helicopter on hard surfaces. The floats are attached to the top of the skid and only support the weight of the helicopter in water. A float with low pressure or one that is completely deflated will not cause any stability problems on a hard surface.

Emergency Pop-Out Floats

The second kind of system that allows helicopters to land on water is the emergency pop-out flotation kits. These kits are like airbags in a car. The deflated airbags are tucked away in interior compartments or exterior pouches and are connected to a gas bottle (generally compressed nitrogen or helium) and a firing system. The system consists of two or more floats with one or more individual compartments per float, depending on the size of the helicopter. (***Figure 14-127***)

In case of an emergency landing on water the pilot can either arm the system so that the detection of water automatically inflates the airbags, or activate the inflation via a button on the collective or cyclic controls. When the system activates, a valve opens on the pressurization bottle and the gas rapidly fills the airbags.

Figure 14-127. Emergency pop-out floats on a helicopter.

Just like with fixed utility floats, each individual airbag is made up of two chambers. If one chamber develops a leak, the other chamber will prevent that part of the helicopter from sinking. This system provides enough time for the occupants to evacuate into a life raft.

Because of the narrow width between the floats, it is not uncommon for a helicopter to overturn, especially in rough water. The whole purpose of the pop-out float system is to provide time for the occupants to escape. With luck and in calm water, and if the helicopter remains upright it may be possible for a recovery of the aircraft to take place.

Question: 14-1
Name the two ways in which a wheeled helicopter can steer on the ground.

Question: 14-2
A centering cam on a nose gear causes the ______________ to line up with each other.

Question: 14-3
______________ plating is a type of electrolytic finish frequently applied to landing gear components for the purpose of ______________.

Question: 14-4
What is the purpose of a metering pin in a shock strut?

Question: 14-5
What information is always given on the attached instruction plate of a shock strut?

Question: 14-6
What type of hydraulic component insures that the retractable gear is in its fully up position before the landing gear doors close?

Question: 14-7
What is meant when a flight deck landing gear position indicator flashes the color amber?

Question: 14-8
What is the first step always required prior to removing a wheel assembly from an aircraft?

ANSWERS

Answer: 14-1
Steerable nose or tail wheel, or by differential braking.

Answer: 14-2
inner and outer struts

Answer: 14-3
Cadmium; corrosion protection.

Answer: 14-4
It controls the rate of fluid flow from the lower to the upper chamber of the strut.

Answer: 14-5
The correct hydraulic fluid type and the pressure to which it is inflated.

Answer: 14-6
Sequence valve.

Answer: 14-7
The gear is in transit, extending or retracting.

Answer: 14-8
Deflate the tires.

HELICOPTER AERODYNAMICS, STRUCTURES AND SYSTEMS

LIGHTS (ATA 33)

SUB-MODULE 15

PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY → B1.3 B1.4

Sub-Module 15

LIGHTS (ATA 33)

Knowledge Requirements

12.15 - Lights (ATA 33)	
Light sources; External: navigation, landing, taxiing, ice; Internal: cabin, cockpit, cargo; Emergency.	3

12.15 - LIGHTS

Aircraft lighting systems provide illumination for both external and internal use. Exterior lights provide illumination for such operations as landing at night, inspection of icing conditions, and safety from midair collision.

Internal lighting provides illumination for instruments, cockpits, cargo, and other sections occupied by crew members and passengers. Certain special lights, such as indicator and warning lights, indicate the operational status of that equipment.

LIGHT SOURCES

Older aircraft lighting made use mostly of incandescent and halogen lighting. Incandescent lamps produce light with a filament wire heated to a high temperature by an electric current until it glows. The hot filament is protected from oxidation in the air with a glass bulb that is filled with inert gas or evacuated. In a halogen lamp, the filament is protected by a chemical process that redeposits metal vapor onto the filament, extending its life. Incandescent bulbs are less efficient than modern types of light bulbs, converting less than 5% of its energy into visible light with the remainder being converted into heat. On modern aircraft, there are three types of light sources primarily used for external lighting:

- High Intensity Discharge (HID) Bulbs
- Light Emitting Diodes (LED) Lamps
- Xenon Flash Tubes

HIGH INTENSITY DISCHARGE BULBS (HID)

HID lamps have become common as replacements for traditional landing and taxi lights. (***Figure 15-1***) In an HID lamp, instead of heating a filament as in an incandescent lamp, a gas is heated to produce light.

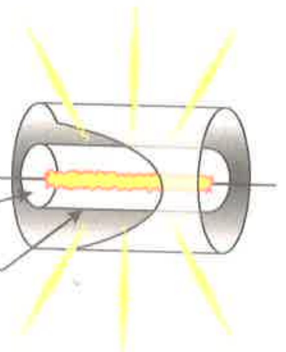


Figure 15-1. The operating principle of a high intensity discharge bulb.

They are a type of electrical gas discharge lamp which produces light by creating an electrical arc between tungsten electrodes placed inside a translucent or transparent tube which is filled with both gas and metal salts. Once the arc forms, it evaporates the metal salts forming a plasma, which produces high intensity light with a low power consumption. (***Figure 15-2***)

HID Precautions

HID lamps emit small amounts of UV rays. While most of them are blocked by the outer tube, some precautions should be taken:

- Parts of the body closely exposed to direct light for long periods of time should be covered because of sunburn. Avoid direct eye contact, and when working closely, always wear UV blocking glasses.
- Because the temperature of the outer tube will range from 240°C - 800°C at its base, do not touch the lamp when in operation. The bulb should be allowed to cool for three minutes after switching off.
- Operating pressures inside HID bulbs are approximately 50 bars in its hot state and remain high when cooled. Thus to guard against explosion wear safety glasses and gloves even when bulbs are switched off.
- While less than fluorescent lamps, HID lamps contain mercury. Mishandling can expose the operator through the skin causing neurological problems in the long run. Always wear safety gloves.
- Because of the mercury content, spare and discarded lamps are carefully packed to prevent breakage and then discarded as special waste.

LIGHT EMITTING DIODES (LED)

LED lamps emit light due to photons emitted by the recombining of electrical charges between its diode layers after the application of current. LED lamps produce light into a single color, depending on the

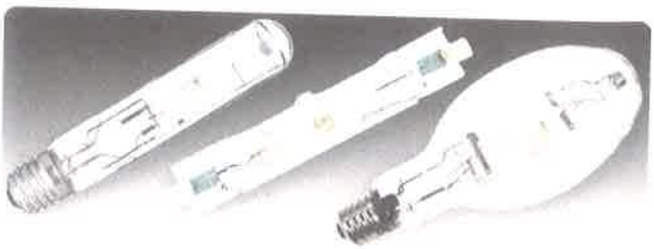


Figure 15-2. Three configurations of HID lamps.

semiconductor material used. (*Figure 15-3*) Inside the semiconductor chip, there are three different layers:

- The "n" layer where the material has an excess of electrons.
- The "p" layer where material has a deficit of electrons, thus an excess of positively charged particles.
- The active layer (the junction of "n" and "p" layers), which is the barrier to the flow of electrons.

When voltage is applied, the flow of electrons across the active layer is reduced, leading to the emission of single color photons. This effect is called electroluminescence. As diodes work with only one voltage direction, LEDs are used only with DC power.

LED technology is a cold light generation technology with almost no heat generation. As such, the lifespan of LED bulbs are significantly longer and so require fewer maintenance actions.

The semiconductor diode is mounted in a reflector cup on one electrode, normally the cathode. The other connection is made by a wire connection from the chip to the second electrode, the anode. This assembly is covered with a lens that helps focus the light and holds the electrical connections in place for power supply.

LEDs are able to produce either red, yellow, orange, amber, green, blue/green, or blue. White LEDs can be produced two ways:

- By combining red, green, and blue chips in one package known as a RGB LED.
- By the use of phosphor coatings, which will be excited by colored LED photons. This method is simpler and has better color integrity but its energy efficiency is less.

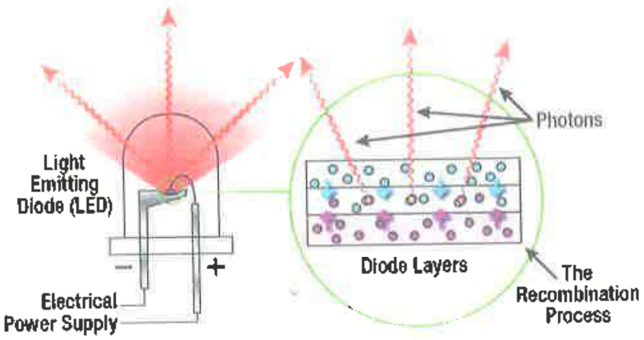


Figure 15-3. The LED light uses the recombination of electrical charges between diode layers.

XENON FLASH TUBES

Xenon flash tubes are used extensively for beacon and strobe lighting. A flash tube is an electric arc lamp designed to produce intense white light for very short durations.

Flash tubes are made of a length of glass tubing with electrodes at either end and filled with a noble gas, typically xenon, which when triggered ionizes and conducts a high voltage pulse to produce light. As a high voltage power source is needed to energize the gas, a charged capacitor is used to allow very speedy delivery of current when the lamp is triggered.

The electrodes protrude into each end of the tube and are sealed to the glass using a few different methods. Ribbon seals are thin strips of molybdenum foil bonded directly to the glass, which are durable but are limited in the amount of current that can pass through. Solder seals bond the glass to the electrode for a very strong mechanical seal, but are limited to low temperature operation. Rod seals are used in laser applications, where the rod of the electrode is wetted with another type of glass and then bonded to a quartz tube. This seal is very durable and capable of withstanding very high temperatures and currents.

The capacitor, which is charged between 250 and 5 000 volts, uses a step-up transformer and a rectifier. The bulb's gas exhibits very high resistance and the lamp will not conduct electricity until the gas is ionized. Once ionized, or "triggered", a spark will form between the electrodes, allowing the capacitor to discharge. The sudden surge of current quickly heats the gas to a plasma state where electrical resistance becomes low. As the current pulse travels through the tube, it ionizes the atoms, causing them to jump to higher energy levels. As the ionized atoms recombine with their lost electrons they immediately drop back to a lower energy state, releasing photons in the process. (*Figure 15-4*)

EXTERNAL: NAVIGATION, LANDING, TAXIING, ICE

Navigation lights (or otherwise known as position lights), anti-collision lights, landing, and taxi lights are common examples of aircraft exterior lights. Some lights are required by regulation for night operations. (*Figure 15-5*)

NAVIGATION/POSITION LIGHTS

Aircraft operating at night must be equipped with position lights that meet minimum requirements as set forth in CS-29.1385-1397. A set of position lights consist of one red, one green, and one white light.

Red and green lighting units are always mounted on the sides of the fuselage. The green unit is mounted on the right side. The red unit is mounted in a similar position on the left side. Each must provide a horizontal beam parallel with the centerline of the helicopter to 110° outboard with 180° vertical coverage.

A white unit is usually located on the vertical stabilizer in a position where it is clearly visible through a wide angle from the rear of the aircraft. It must provide a horizontal beam parallel with the centerline of the helicopter to 140° outboard with 180° vertical coverage. (*Figure 15-6*)

Figure 15-4. A Xenon flash tube as used with a strobe light.

Some lighting units contain a single lamp mounted on the surface of the aircraft. Others contain two lamps and are often streamlined into the aircraft structure. On some types of installations, a switch in the cockpit allows a choice of either steady or flashing operation.

There are many variations of the position light circuits used on different helicopters. All circuits are protected by fuses or circuit breakers and many circuits include flashing and dimming equipment. *Figure 15-7* illustrates a schematic diagram of a position light circuit.

Figure 15-6. Navigation/Position lights.

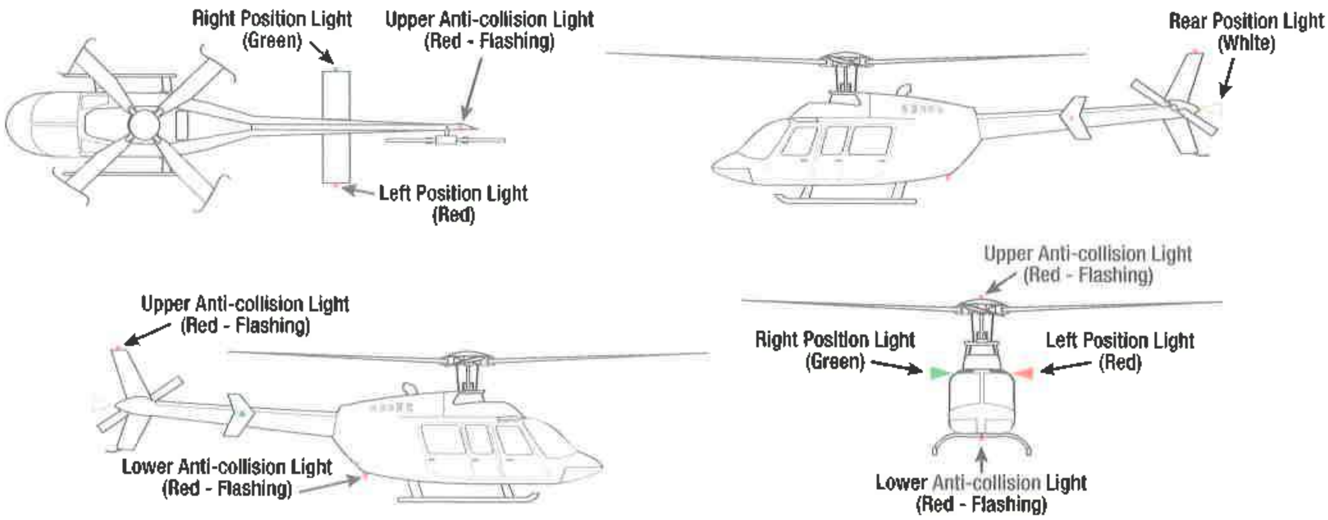


Figure 15-5. A typical external lighting array showing position lights and top and bottom anti-collision lights.

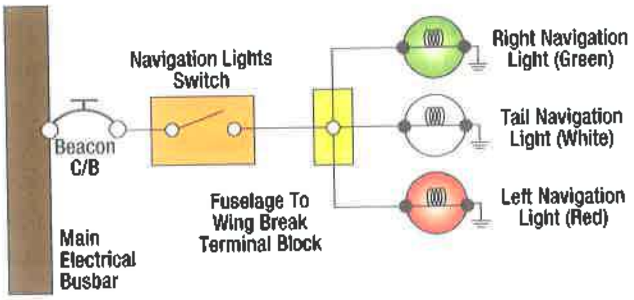


Figure 15-7. Navigation light system schematic.

Figure 15-8. A forward position LED position light from Devore Aviation.

Small aircraft are usually equipped with a simplified control switch and circuitry. In some cases, one control knob or switch is used to turn on several sets of lights. For example, one system utilizes a control knob, the first movement of which turns on the position lights and the instrument panel lights. Further rotation of the control knob increases the intensity of only the panel lights.

A flasher unit is generally included in the position light circuitry of all aircraft. Traditional position lights use incandescent light bulbs. LED lights have been introduced on modern aircraft because of their good visibility, high reliability, and low power consumption. (***Figure 15-8***)

ANTI-COLLISION LIGHTS

Anti-collision lights are safety lights to warn other aircraft, especially in congested areas. Anti-collision lights may be of two types:

- Rotating Beacons
- Flashing Strobe Lights

Rotating Beacons

Rotating beacons are usually installed on top of the fuselage or tail and/or under the fuselage. Each is installed in such a location that the light does not affect the vision of the crew member or detract from the visibility of the position lights. ***Figure 15-9*** shows a typical anti-collision light installation in a vertical stabilizer.

Figure 15-9. A typical anti-collision light installation in a vertical stabilizer.

The beacon illustrated in (***Figure 15-10A***) employs a V-shaped reflector over the axis of a single sealed lamp. One half of the rotating reflector is flat and emits a narrow high intensity beam towards the horizon, while the other half is curved to increase the up and down spread of its beam to 30° above and below the horizon. The gear and pinion type motor turns the reflector at a constant speed of 40-45 RPM giving an observed flash frequency of 80-90 flashes per minute.

Another type of beacon illustrated in (***Figure 15-10B***) employs two filament lamps, mounted in tandem, each of which is pivoted on its own axis. One half of each

Figure 15-10A. A V-shaped reflector over the axis of a single sealed lamp.

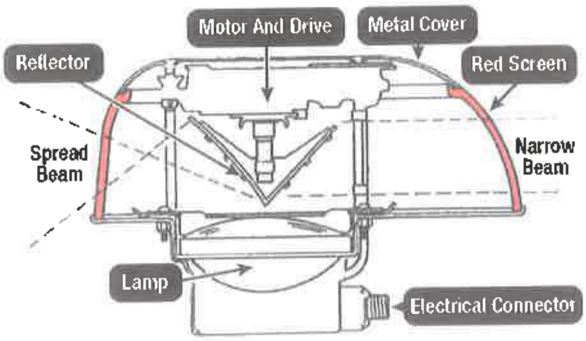


Figure 15-10B. Two filament lamps mounted in tandem.

LIGHTS

lamp forms a reflector by being silvered. The drive from the motor is so arranged that the lamps oscillate through 180° so that the two beams are 180° apart at any instant.

A rotating beacon circuit is shown in ***Figure 15-11.***

Strobe Lighting

A white strobe light is a second common type of anti-collision light. Strobe lights produce an extremely bright intermittent flash of white light that is highly visible. The light is produced by a high voltage (400 volt) discharge of a capacitor which converts an input power of either 28V DC or 115V AC into a high DC output. The capacitor is charged to periodically discharged between two electrodes in the xenon filled tube, producing a high intensity flash having a characteristic blue-white color. (***Figure 15-12***)

High intensity strobe lights can be dangerous to service, as the capacitors are charged to voltages which can be lethal. Accordingly, a minimum of two minutes should be allowed for the capacitors to discharge after the circuit has been deenergized. In addition, damage to the eyes may result from looking directly into the high intensity light. A flashing strobe circuit is shown in ***Figure 15-13.***

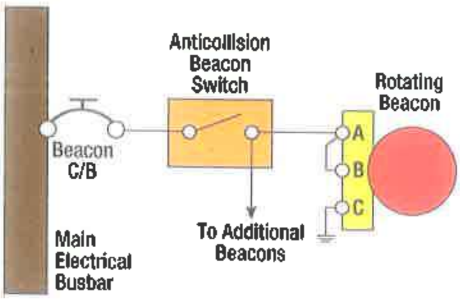


Figure 15-11. A rotating beacon circuit.

Figure 15-12. An LED anti-collision light from Devore Aviation.

LANDING LIGHTS

Landing lights are installed in aircraft to illuminate runways and landing pads during night landings. These lights are very powerful and are directed by a parabolic reflector at an angle providing a maximum range of illumination. Landing lights are usually located under the cockpit within the aircraft surface. Each light may be controlled by a relay, or it may be connected directly into the electric circuit. A sealed beam, halogen, or high intensity xenon discharge lamp is used. (***Figure 15-14***)

TAXI LIGHTS

Taxi lights are designed to provide illumination on the ground while taxiing or towing the aircraft to or from a runway, taxi strip, or hangar area. Taxiing lights are not designed to provide the greater degree of illumination necessary for landing lights. On aircraft with tricycle landing gear, either single or multiple taxiing lights are often mounted on the non steerable part of the nose landing gear. They are positioned at an oblique angle

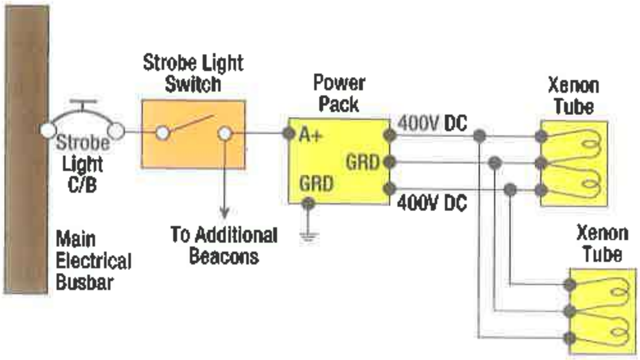


Figure 15-13. A flashing strobe circuit.

Figure 15-14. A combination landing and rotatable search light on an airbus BK 117.

to the center line of the aircraft to provide illumination directly in front of the aircraft and to an angle to the right and left of the aircraft path.

Some small helicopters are not equipped with any type of taxiing light but instead rely on the intermittent use of a landing light to illuminate the taxiing route. Still others utilize a dimming resistor in the landing light circuit to provide reduced illumination for taxiing. Another possibility is to have a high beam type adjustment to use the same light to land and to taxi. These lights, operated by a separate switch from the main taxiing lights, illuminate the area immediately in front of and below the aircraft nose. A typical circuit for taxiing lights is shown in ***Figure 15-15***.

ICE INSPECTION LIGHTS

Some aircraft are equipped with inspection lights to illuminate the external structure to allow observation of icing and the general condition of these areas during flight. These lights allow visual detection of ice formation on the skate legs or on parts visible to the pilot during night flight. They are usually controlled via a relay by a toggle switch in the cockpit.

INTERNAL: CABIN, COCKPIT, CARGO

A variety of interior lights are found on modern helicopters, including instrument lights, overhead lights, and step lights to mention a few. These lights can be divided into incandescent and fluorescent types.

FLUORESCENT LIGHTS

Fluorescent lights are made of a glass tube filled with noble gases and mercury vapor which glows when an AC voltage is applied to the electrodes at each end. The electrodes then emit electrons which strike the atoms of mercury vapor in the tube to produce ultraviolet light. The invisible ultraviolet light strikes a phosphorous coating on the inside of the tube which glows white.

Fluorescent lamps are more efficient than incandescent lamps, however because they require the use of a ballast transformer and AC voltage, they are typically found only on large aircraft. Fluorescent lamps can operate in a bright or dimmed position. For the dim position, reduced voltage is applied to the transformer. (***Figure 15-16***)

Fluorescent Lamp Operating Circuit

Fluorescent lamps are negative differential resistance devices. As more current flows through them, the

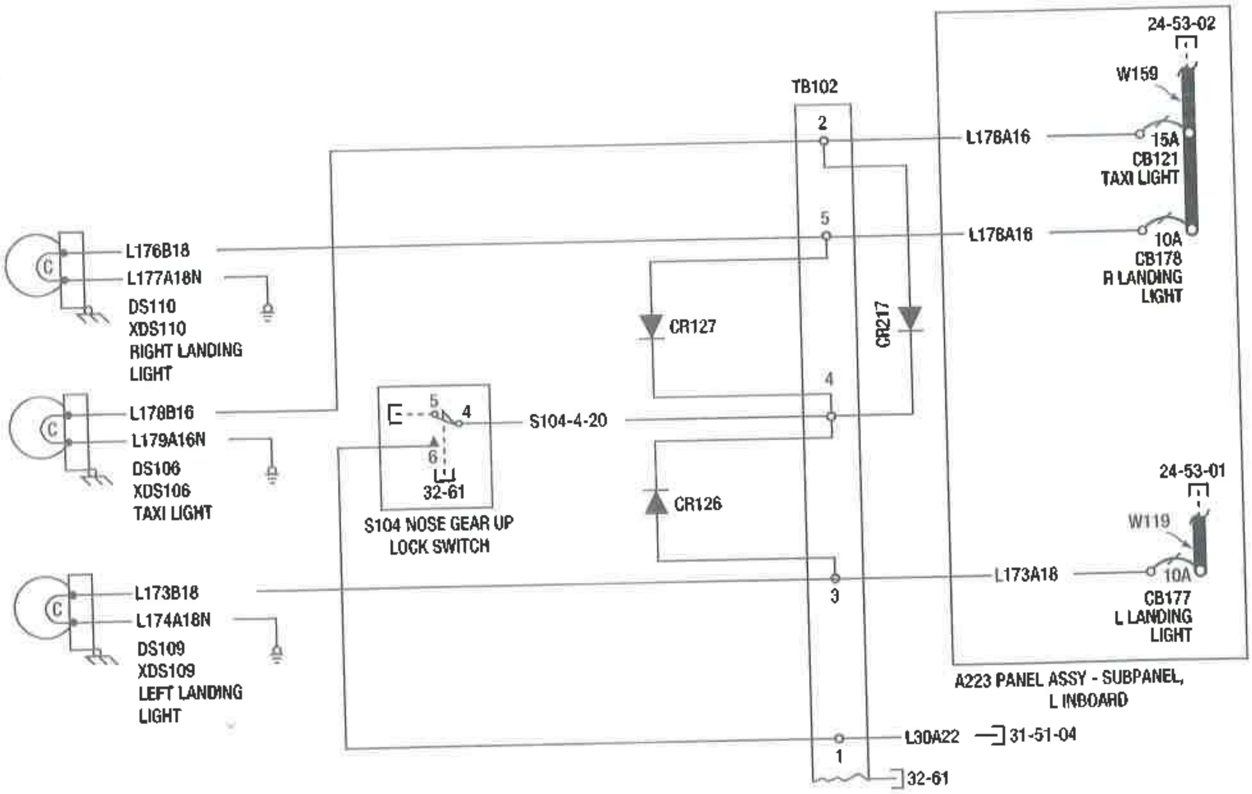


Figure 15-15. A typical circuit for taxiing lights.

Figure 15-16. (A) A fluorescent tube, (B) electrodes and ballast.

electrical resistance of the lamp drops allowing even more current to flow. If connected directly to a constant voltage power supply, the lamp would rapidly self destruct due to the uncontrolled flow. To prevent this, fluorescent lamps must use an auxiliary device called a ballast to regulate flow through the tube.

The simplest ballast is an inductor placed in series, and consisting of a winding on a laminated magnetic core. The inductance of this winding limits the current flow. Ballasts are rated for the size of the lamp and the power frequency. Where the main voltage is insufficient to start long fluorescent lamps, the ballast acts as a step up autotransformer to limit the current flow. A ballast may also include a capacitor for power correction.

FLIGHT DECK LIGHTING

On a flight deck, it is normal to have lighting for general illumination as well as local lighting for panels, instruments, and controls. Often red lighting is provided on the flight deck which helps to maintain the pilot's night vision of objects outside the aircraft. Fluorescent background lights are also used.

A central panel, (typically overhead), houses the controls for most internal and external lights. Additional light controls may also be located on appropriate panels. General lighting requirements are met using 28V AC power with certain key lights positioned to operate in partial or no power conditions from a 28V DC bus.

Dome lights provide overhead shadowless general lighting. The 28V DC battery bus usually supplies the operating power. The operation is typically from a three position (DIM-OFF-ON) toggle switch on the overhead panel. A resistor controls the voltage drop for the dimming circuit.

Map holder lighting is provided at the captain and first officer stations. Each is controlled independently and typically via the 28V AC bus. Side console and floor lighting is provided at the captain and first officer stations. The center instrument panel is illuminated by a set of lights below the glare shield. The standby compass is provided with its own integral lighting. Individual reading lights are provided at the captain and first officer stations. A flood light provides illumination of the center pedestal. Each of these is typically powered by the 28V AC bus.

Storm lights are sometimes fitted in the cockpit. These extremely bright lights are switched on during a storm. They raise the ambient light level and cause the iris in the eye to reduce the amount of light entering the eye, thus reducing the chance that a pilot may be temporarily blinded by a bright lightning strike. (***Figure 15-17***)

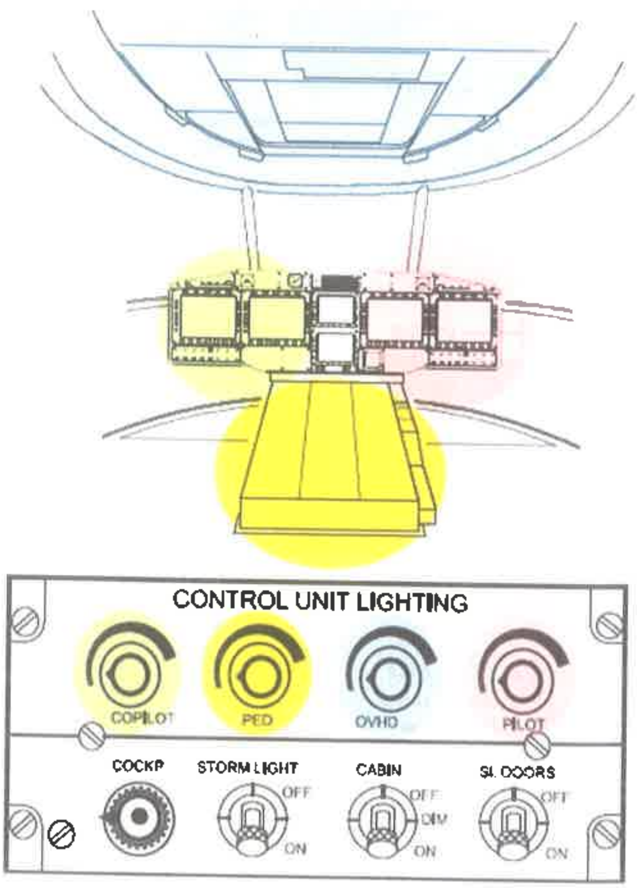


Figure 15-17. Typical cockpit lighting zones.

Integral Instrument Lighting

A common form of integral lighting for instruments is known as 'wedge' lighting; deriving its name from the two wedge shaped portions of glass which make up the instrument cover. Wedge lighting relies on the physical law that the angle at which light leaves a reflecting surface equals the angle at which it strikes that surface. The two wedges are mounted opposite each other with a narrow space separating them. (***Figure 15-18***) Light is introduced into inner wedge (A) from a lamp in its wide end. Some light the passes through the wedge and onto the face of the dial while the remainder is reflected into the wedge. The angle at which the light strikes the wedge surface governs the amount of light reflected; the lower the angle, the more light is reflected. The double wedge mechanically changes the angle at which the light rays strike one of the reflecting surfaces, thus distributing the light evenly across the dial and limiting the amount given off by the dial face.

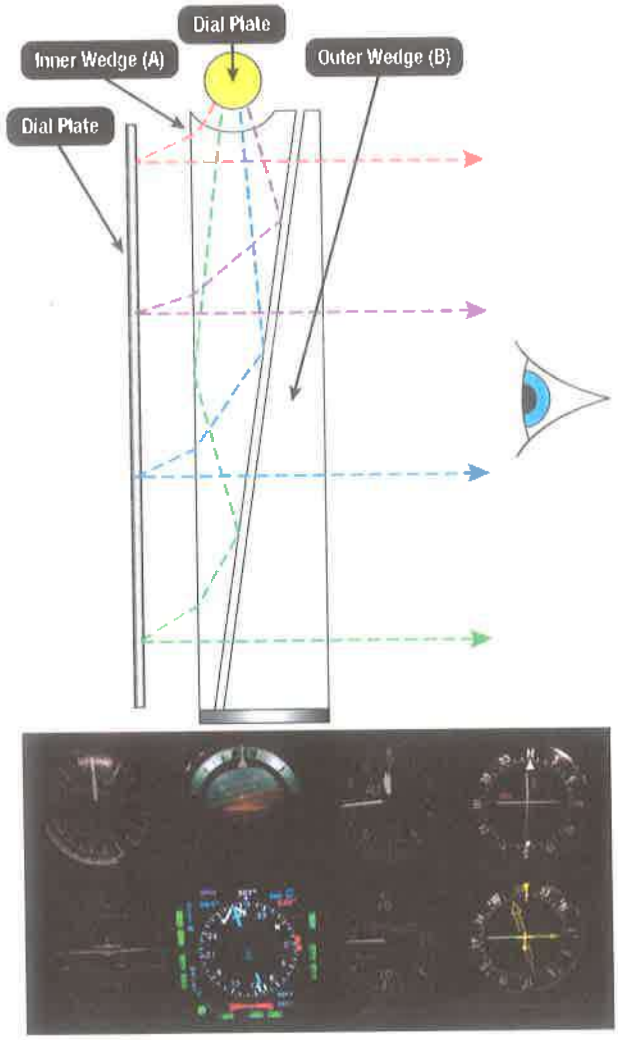


Figure 15-18. Wedge lighting.

CABIN LIGHTING

The extent to which lighting is used in a passenger compartment is dependent on the decor of that aircraft. It can vary from a small number of incandescent lamps to several fluorescent fittings located in the ceiling to combine pleasing, and functional effects. (***Figure 15-19***)

Each fluorescent tube requires a ballast to provide the momentary high voltage enabling the tube to illuminate. In all commercial passenger transport helicopters, the lights are controlled from panels at the cabin attendant's stations. Cabin lighting is typically provided by The 115V AC bus.

In addition to the general lighting, lights are also provided at passenger service units and for essential information signs, such as "Fasten Seat Belts/No Smoking". The lights for these signs may be incandescent or of the electro-luminescent type. Lights for signs conveying essential information are usually controlled from the cockpit. Passenger compartment lighting also includes those for entry ways, attendant work areas, lavatories and galleys. The typical circuitry for essential information lighting is shown in (***Figure 15-20***)

CARGO AND BAGGAGE COMPARTMENTS LIGHTS

Cargo and service compartments also have lighting. Dome lights, flood lights and explosion proof lights as required are installed with independent circuits protected by circuit breakers. The lights are controlled by switches near the entrance to each area or inside the compartments. Often, a control panel for a cargo area includes light switches in addition to door and cargo system operating controls.

Figure 15-19. Cabin lighting.

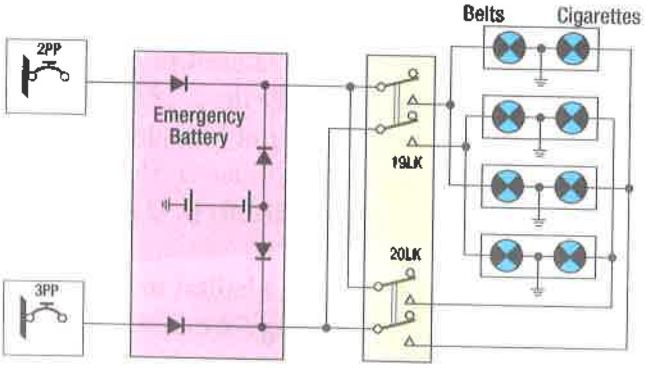


Figure 15-20. Typical circuitry for essential information lighting.

EMERGENCY LIGHTING

Emergency lights are used to illuminate all escape doors. In case of complete or partial electrical failure, various configurations exist for automatic switching of certain emergency lights to the hot DC battery bus which is an integral part of the lighting installation. Some interior lighting is designed to always be connected to a DC bus, so no switching is required.

Emergency exit area lights for example, may contain a battery in the assembly that includes the lamps, cover lens, solid state switching logic and battery charging control circuits. In some cases, the light/battery assembly can be removed from its mounted location and used as a portable flashlight. NiCad batteries are typical. In other configurations, the dedicated emergency light battery is remotely located in the same area as the light.

Regardless, emergency lighting is armed by a switch on the flight deck or in the cabin lighting control panel. All-in-one emergency light assemblies also have a switch that must be set to ARMED when the unit is installed.

Inspection of an aircraft emergency lighting system normally includes checking the condition and security of all visible wiring, connections, terminals, fuses, and switches and light units. A continuity light or meter can be used in making these checks, since the cause of many troubles can often be located by systematically testing each circuit for continuity.

ELECTROLUMINESCENCE

Electroluminescent strip lighting eliminates the need for bulbs, sockets, diffusers and reflectors. Without filaments to break, the lighting can withstand extreme shock, vibration and high or low temperatures without failure. Numerous tests and experience has proven that the system will continue operating under very high G forces and/or after considerable structural damage. Electroluminescent lighting is more easily seen through smoke than incandescent or other point sources.

Thin film electroluminescent displays comprise a solid-state glass panel, an electronic control circuit and a power supply. The glass panel consists of a luminescent phosphorous layer sandwiched between transparent dielectric layers and a matrix of electrodes. The circuit board, which contains the drive and control electronics, is connected directly to the back of the glass panel. A pixel on the display is lit by applying voltage to the rows and columns of electrodes, thus causing the area of intersection to emit light. (***Figure 15-21***)

SELF ILLUMINATING SIGNS

Self illuminating signs are entirely self powered. Their brightness is such that they are instantly seen by persons that are not dark adapted, and present no direct radiation hazard.

Each is a small sealed glass envelope internally coated with a layer of phosphor and containing tritium gas. Tritium is an isotope of hydrogen and emits electrons which, on striking the layer of phosphor powder, causes it to emit visible light. Placing the light element behind a diffusing panel provides a good means of conveying emergency instructions in darkened areas.

The lighting for self illuminated signs comes from a radioactive material. They are always on and can not be shut off. Each is a plastic container that holds capsules of radioactive tritium gas. The sign is safe unless it is broken. However, if it does have a hole or a crack, the radioactive gas can be inhaled or absorbed causing injury. Should breakage occur, the aircraft should be evacuated and all doors left open to allow maximum ventilation.

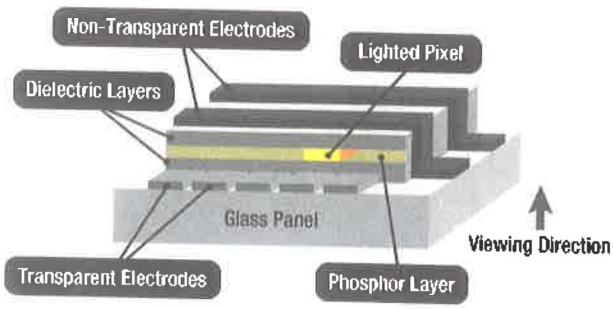


Figure 15-21. A thin film electroluminescence system by the Lumineq Company.

Disposal of signs are subject to the applicable regulations regarding radioactive substances. All self illuminating signs should be checked for luminosity level on initial fitting and at periods specified in its maintenance schedule. Such signs usually have a scrap life of 5 years and should then be returned to the manufacturer for disposal.

FLOOR MOUNTED ESCAPE PATH LIGHTING

The floor emergency lighting system is an additional system to show passengers where the emergency exits are in case there is smoke in the compartment and the emergency lights are not visible. Electroluminescent strips with blocks and arrows pointing toward the exits are installed on the floor. Electroluminescent EXIT sign panels are on the wall just above the floor near each emergency exit.

The same batteries used for the emergency lighting, supply power to this system. Inverters convert the DC to 115V 400 Hz and are installed below the floor panels. The color of the floor lighting or exit signs can be changed by using different filters to match with the color of the carpet or side panels. If floor proximity lighting is inoperative, the aircraft is allowed to fly with passengers, but only during daylight hours. (***Figure 15-22***)

EMERGENCY EXIT LIGHTING ACTIVATION

In principle, at least two separate means of activation should be provided:

- By flight crew action, to switch all exit light systems simultaneously.
- Automatically, when the cabin becomes more than half submerged in water, each emergency exit being provided with its own automatic switch.
- Cockpit emergency exit lights should activate automatically, unless it can be shown that reflections or dazzle would be a hazard to the flight crew.

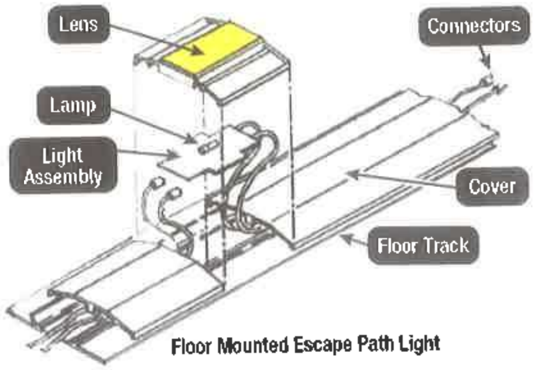


Figure 15-22. A floor mounted escape path light.

LIGHTS

Question: 15-1
Which type of aircraft light requires a capacitor for its operation?

Question: 15-2
What determines the color of light produced by an LED?

Question: 15-3
Which type(s) of external lighting is/are required for night flights by EASA regulation?

Question: 15-4
On a rotating beacon, how many flashes of light per minute are observed by a person on the ground, if the beacon rotates at a speed of 40 RPM?

Question: 15-5
Two dangers to the technician exist when servicing strobe lights. Name them.

Question: 15-6
A ______________ is essentially an inductor placed in series consisting of a winding on a laminated magnetic core.

Question: 15-7
Why is lighting on the flight deck often colored red?

Question: 15-8
Why do special handling precautions exist for self illuminating emergency lighting?

ANSWERS

Answer: 15-1
Strobe lights; which are typically Xenon flash Tubes.

Answer: 15-2
The material used in the semiconductor.

Answer: 15-3
Three position lights; green on the right side, red on the left side, white on the tail.

Answer: 15-4
80 flashes will be observed per minute as each rotating beacon contains 2 lamps at 180° opposition.

Answer: 15-5
1. The lamp's capacitor may still be charged casing a severe electrical shock.
2. The bright light from a discharge may cause blindness if looking directly at it.

Answer: 15-6
Fluorescent lamp ballast.

Answer: 15-7
To preserve the flight crew's night vision for objects outside the aircraft.

Answer: 15-8
They contain radioactive tritium gas.

HELICOPTER AERODYNAMICS, STRUCTURES AND SYSTEMS

PNEUMATIC/VACUUM (ATA 36)

SUB-MODULE 16

PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY → B1.3 B1.4

Sub-Module 16

PNEUMATIC/VACUUM (ATA 36)

Knowledge Requirements

12.16 - Pneumatic/Vacuum (ATA 36) High pressure systems; Medium pressure systems; System layout; Sources: engine/APU, compressors, reservoirs, ground supply; Pressure control; Distribution; Indications and warnings; Interfaces with other systems.	3

12.16 - PNEUMATIC/VACUUM (ATA 36)

Pneumatic and hydraulic systems are similar in that they use confined fluids. Since liquids and gases flow, they are both considered fluids; however, there is a great difference in the characteristics of the two. Liquids are practically incompressible; a quart of water still occupies about a quart of space regardless of how hard it is compressed. But gases are highly compressible; a quart of air can be compressed into a thimbleful of space. In spite of this difference, gases and liquids are both fluids which are confined and made to transmit power. The type of unit used to provide pressurized air for pneumatic systems is determined by the system's air pressure requirements.

HIGH PRESSURE SYSTEMS

In the past, some aircraft manufacturers equipped their aircraft with a high pressure pneumatic system (3 000 psi). Such systems operate a great deal like hydraulic systems, except they employ air instead of a liquid for transmitting power. High pressure pneumatic systems are sometimes used for:

- Opening and closing doors.
- Driving hydraulic pumps, alternators, starters, water injection pumps, etc.
- Operating emergency devices (landing gear).

SYSTEM LAYOUT

Many high pressure pneumatic systems are installed for one time emergency or back-up use and are completely discharged when used. They use pressurized air or nitrogen containers with no on-board means provided to repressurize the system once deployed. Other high pressure pneumatic systems use pressurized containers that are recharged during flight through the action of compressors installed in the system. This type of installation allows the pneumatic system to operate components repeatedly rather than just once in a manner similar to a hydraulic system. ***Figure 16-1*** shows a typical layout of a high pressure pneumatic system on an aircraft equipped with onboard compressors.

SOURCES

Sources for high pressure pneumatic systems include engine driven and other on-board compressors, ground air, and ground nitrogen sources.

As stated, some aircraft employ permanently installed air compressors which recharge air bottles whenever pressure is used for operation of a unit. Several types of compressors are used for this purpose. Some have two stages of compression, while others have three,

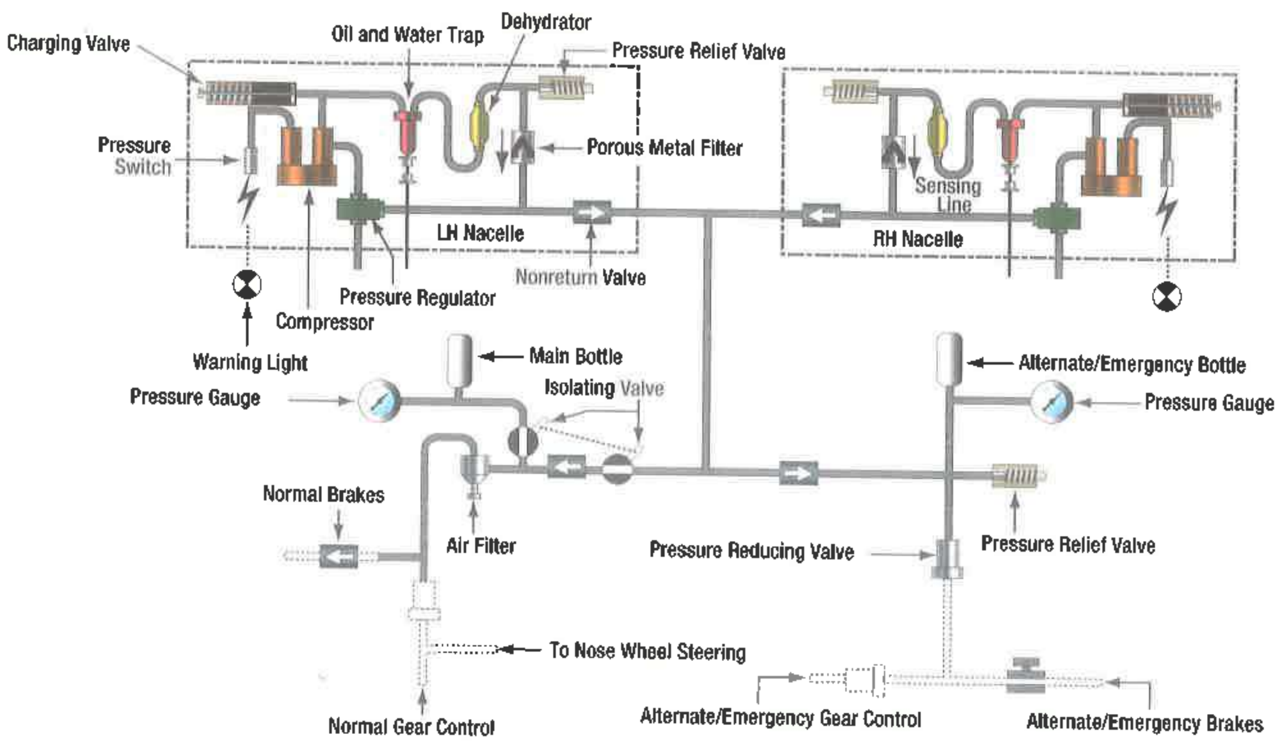


Figure 16-1. High-pressure pneumatic system.

depending on the maximum desired operating pressure. Details on compressor operation are found in the aircraft maintenance manual. They are typically oil lubricated thus the system plumbing may contain an oil separator of some type as well as a means for removing moisture in the system.

Alternately, air and nitrogen storage containers for pneumatic systems are filled on the ground with either a ground based compressor or a high pressure bottle transfer for nitrogen.

STORAGE

For high-pressure systems, air is usually stored in metal bottles at pressures ranging from 1 000 to 3 000 psi, depending on the particular system. The high pressure storage cylinder is typically a light weight, wire-wrapped, steel-constructed reservoir. The date of manufacture and safe working pressure should be on the reservoir as well as a date stamped for the performance of the last hydrostatic test. It is common practice for these high pressure containers to be inspected often and removed periodically for hydrostatic checks.

A standpipe is commonly used at the discharge port to prevent any water that has collected inside the container from entering the system. Air flowing out of the container must go through the end of the standpipe which is elevated above any conceivable water level. This type of container is used in both one-time and multi-deployment systems.

CHARGING

Charging of high pressure bottles is done with either an on-board compressor or a ground source. The typical high-pressure storage bottle has two ports, one of which is equipped with a charging valve. A ground operated compressor or air bottle can be connected to this valve to add air. Nitrogen may also be introduce through this valve.

For onboard charging, the charging valve is plumbed to the compressor outlet. The other valve on a typical high pressure pneumatic reservoir is a control valve. It acts as a shutoff valve, keeping air trapped inside the bottle until the system is operated. This valve must be opened when installed in a chargeable system. Reservoir contents stay held in the bottle with system pressure. A pressure switch is used for flight deck warnings.

DISTRIBUTION

Pneumatic power is distributed through high pressure steel or stainless steel lines. The use of check valves is common to prevent back flow. The lines are routed in the same manner as hydraulic lines to reach the components. In systems that operate one time and emergency systems, a shuttle valve is often used to close off the normal system flow and allow flow of high pressure pneumatic air to operate the component.

SUPPLY REGULATION

A pressure regulator maintains system pressure with a relief valve to limit pressure in case of regulator failure. Check valves are used to prevent back flow to the compressor. In addition to the use of a selector valve or control valve to direct the air to the portion of the system through which it must be distributed, isolation valves are often installed in the distribution system to isolate working components from those that are inoperative or to isolate part of the system that has a leak.

NOTE: All components in a high pressure pneumatic system do not necessarily operate at full system pressure. Pressure reducing valves are used to lower the system pressure to that required by a particular component or sub system. Restrictors and variable restrictors are used to control the speed of the component(s) operated by pneumatics. (***Figure 16-2***)

The few high pressure pneumatic systems on aircraft that the technician encounters are either one time use or multi-use or both. ***Figure 16-3*** illustrates part of a pneumatic system that uses a rechargeable system for normal operation of gear extension and retraction as well as brake operation. For emergency brake application, a completely redundant distribution system supplies high pressure air from a reservoir (not shown) independent of the normal system.

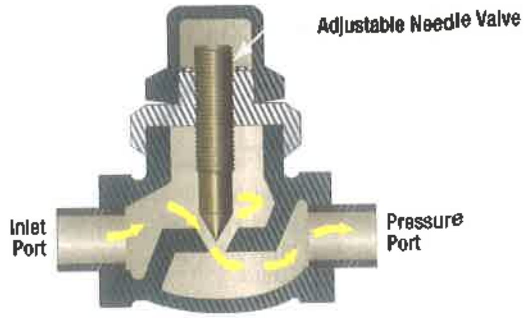


Figure 16-2. Variable pneumatic restrictor.

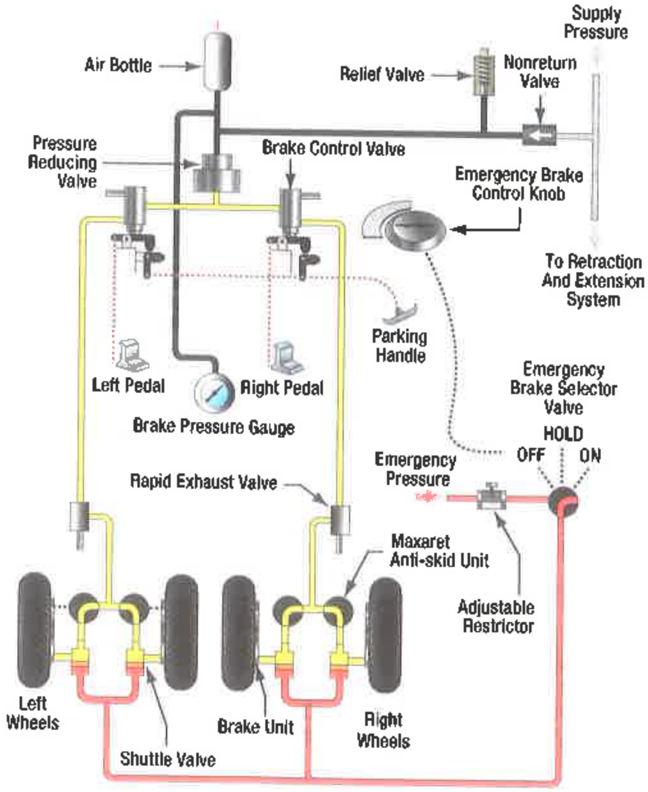


Figure 16-3. Normal rechargeable and emergency non-rechargeable pneumatic brake systems on the same aircraft.

EMERGENCY BACK-UP SYSTEMS AND PNEUDRAULICS

Many aircraft use a high pressure pneumatic back-up source of power to extend the landing gear or actuate the brakes if the main hydraulic braking system fails. High pressure nitrogen is not directly used to actuate the landing gear actuators or brake units but, instead, it applies the pressurized nitrogen to move hydraulic fluid to the actuator. This process is called pneudraulics. The following paragraph discusses the components and operation of an emergency pneumatic landing gear extension system used on a business jet. (***Figure 16-4***)

Nitrogen Bottles

Nitrogen used for emergency landing gear extension is stored in two bottles, one bottle located on each side of the nose wheel well. Nitrogen from the bottles is released by actuation of an outlet valve. Once depleted, the bottles must be recharged by maintenance personnel. Fully serviced pressure is approximately 3 100 psi at 21°C (70°F) enough for only one extension of the landing gear.

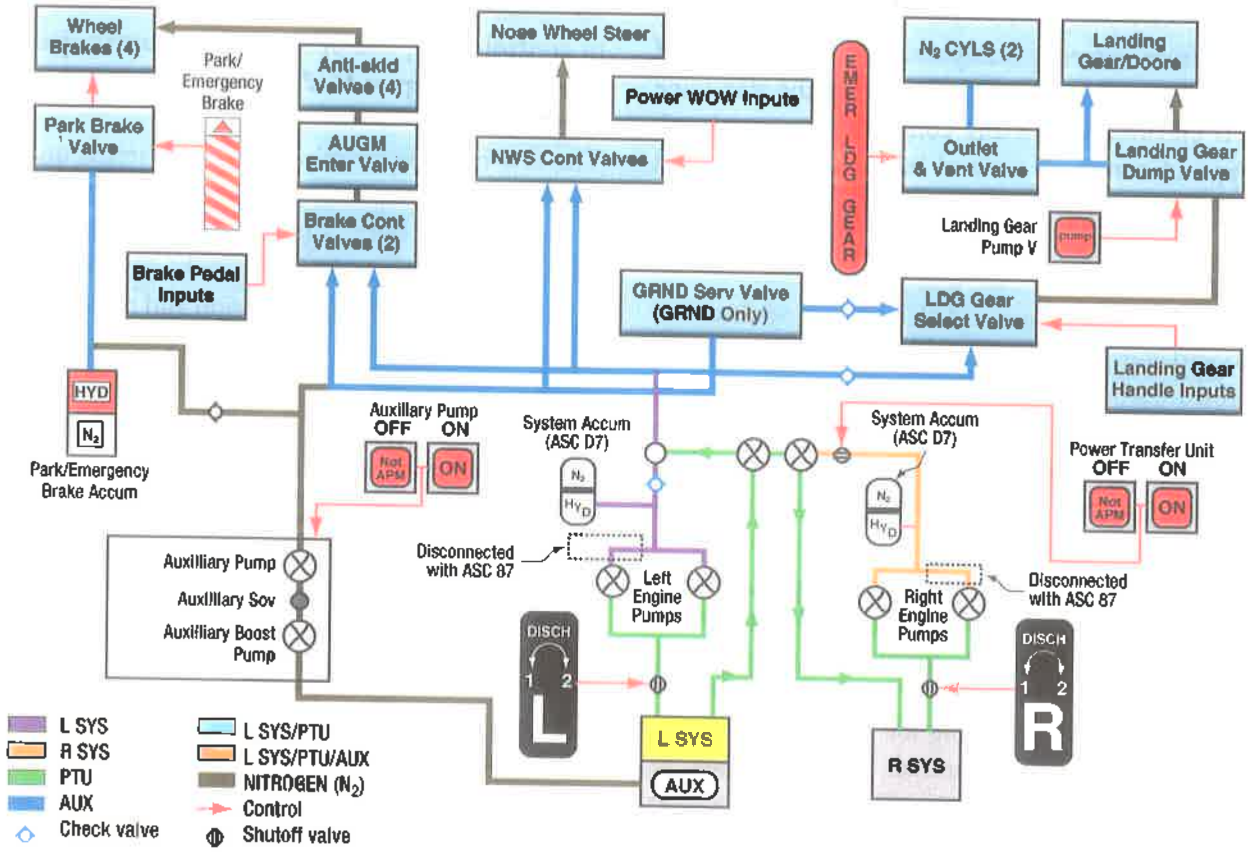


Figure 16-4. Pneumatic emergency landing gear extension system.

Gear Emergency Extension Cable and Handle

The outlet valve is connected to a cable and handle assembly. The handle is located on the side of the copilot's console and is labeled EMER LDG GEAR. Pulling the handle fully upward opens the outlet valve, releasing compressed nitrogen into the landing gear extension system. Pushing the handle fully downward closes the outlet valve and allows any nitrogen present in the emergency landing gear extension system to be vented overboard. The venting process takes approximately 30 seconds.

Dump Valve

As compressed nitrogen is released to the landing gear selector/dump valve during emergency extension, the pneudraulic pressure actuates the dump valve portion of the landing gear selector/dump valve to isolate the landing gear system from the remainder of the hydraulic system. When activated, a blue DUMP legend is illuminated on the LDG GR DUMP V switch, located on the panel. A dump valve reset switch is used to reset the dump valve after the system has been serviced.

Emergency Extension Sequence

1. Landing gear handle is placed in the DOWN position.
2. Illuminated Red light in the landing gear control handle.
3. EMER LDG GEAR handle is pulled fully outward.
4. Compressed nitrogen is released to the landing gear selector/dump valve.
5. Pneudraulic pressure actuates the dump valve portion of the landing gear selector/dump valve.
6. Blue DUMP legend is illuminated on the LDG GR DUMP switch.
7. Landing gear system is isolated from the remainder of hydraulic system.
8. Pneudraulic pressure is routed to the OPEN side of the landing gear door actuators, the UNLOCK side of the landing gear uplock actuators, and the EXTEND side of the main landing gear sidebrace actuators and nose landing gear extend/retract actuator.
9. Landing gear doors open.
10. Uplock actuators unlock.
11. Landing gear extends down and locks.
12. Three green DOWN AND LOCKED lights on the landing gear control panel are illuminated.
13. Landing gear doors remain open.

HIGH PRESSURE PNEUMATIC POWER SYSTEM MAINTENANCE

Maintenance of high pressure pneumatic power systems consists of servicing, troubleshooting, removal, and installation of components, and operational testing.

The air compressor's lubricating oil level should be checked daily in accordance with the applicable manufacturer's instructions. The oil level is indicated by means of a sight gauge or dipstick. When refilling the compressor oil tank, the oil (type specified in the applicable instructions manual) is added until the specified level. After the oil is added, ensure that the filler plug is torqued and safety wire is properly installed.

The pneumatic system should be purged periodically to remove the contamination, moisture, or oil from the components and lines. Purging the system is accomplished by pressurizing it and removing the plumbing from various components throughout the system. Removal of the pressurized lines causes a high rate of airflow through the system, causing foreign matter to be exhausted from the system. If an excessive amount of foreign matter, particularly oil, is exhausted from any one system, the lines and components should be removed and cleaned or replaced. Upon completion of pneumatic system purging and after reconnecting all the system components, the system air bottles should be drained to exhaust any moisture or impurities that may have accumulated there.

After draining the air bottles, service the system with nitrogen or clean, dry, compressed air. The system should then be given a thorough operational check and an inspection for leaks and security.

MEDIUM PRESSURE PNEUMATIC SYSTEMS

SYSTEM LAYOUT

Medium pressure pneumatic systems on large passenger aircraft are typically designed around the sources for pneumatic air that feed a common manifold. Each engine contains an independent bleed air subsystem that is designed to extract and regulate pneumatic bleed air from the engine. It is then forwarded to the pneumatic manifold for use. The pneumatic manifold contains the control valves that are operated to supply the systems that require pneumatic power.

An isolation valve separates the pneumatic manifold from each engine bleed air supply and regulation subsystem so as to be able to turn the supply ON and OFF from that engine. The APU is similarly designed although the APU may turn a dedicated load compressor to supply the air rather than tapping bleed air off the compressor section of the engine. A pneumatic power supply cart provides already regulated air pressure. When it is used to supply the manifold, the aircraft engines are not operated. A ground pneumatic air supply adapter with check valve is located directly in the pneumatic manifold. Closing the engine and APU isolation valves isolates the ground air supply. The supply cart must be powered down to deenergize the pneumatic manifold and remove the hose.

SOURCES

A medium pressure pneumatic system (35-150 psi) does not include an air bottle/storage reservoir. Instead, it draws air from the compressor section of a turbine engine. This is known as bleed air and is used to provide pneumatic power for engine starts, engine de-icing, wing de-icing, air conditioning and more. In some cases, it provides hydraulic power to the aircraft systems (if the hydraulic system is equipped with an air driven hydraulic pump). Engine bleed air is also used to pressurize the aircraft's hydraulic reservoirs, anti-ice the Total Air Pressure probe and other applications specific to particular aircraft.

Ground sources of pneumatic air also are used. Fixed and portable cart type units containing engine-driven air supply compressors are connected to the pneumatic manifold to power the pneumatic system without running the engines. A ground air supply connector and check valve is provided in the manifold for the duct-diameter sized hose from the ground source.

STORAGE

Bleed air pneumatic systems normally do not store pneumatic air in any particular container like the reservoir bottles of a high pressure pneumatic system. Each turbine engine and the APU supply the bleed air. A shutoff or regulating and shutoff valve is typically located between the engine bleed air tap-offs and the pneumatic ducting that makes up the pneumatic manifold. A shutoff type valve is also used to control the flow of pneumatic air from the APU.

Thus, the pneumatic manifold, which is typically 4 inch diameter ducting, may be considered a storage location. It is located downstream of the pneumatic shutoff valves from the engines and APU. Control valves allow pneumatic air to be routed from the manifold into pneumatically powered components such as engine starters, pneumatically driven hydraulic pumps, and into the wing anti-ice ducts and air conditioning packages.

PRESSURE CONTROL

Turbine aircraft pneumatic system pressure control begins with control of engine compressor bleed air. Intermediate-stage compressor bleed air normally supplies the bulk of the pneumatic system demand. However, in times of high demand or reduced engine throttle, a second, and sometimes a third tap off of high stage compressor bleed air is combined with intermediate stage air to supply sufficient volume for operating pneumatic system component demands. Various pressure regulating and sourcing valves are used to deliver the optimum volume of air into the pneumatic manifold at any given time. On the most modern aircraft, regulation is maintained electronically. Digital data buses supply inputs to central pneumatic system control computers. The computers set the position of the various valves in the system to meet demand. The dominant use of pneumatic air is cabin air conditioning.

Without the benefit of computer control found on the most modern aircraft, many aircraft control and operate bleed air pneumatic system pressure regulating valves solely with pneumatic pressure. No electricity is needed. Pneumatic pressure in the pneumatic manifold is routed to a pressure regulator that also receives pneumatic/bleed air input from other locations. Pneumatic temperature inputs are also used. The regulator's internal mechanism balances the air inputs as required by system demands. This is largely done with springs and chambers with diaphragms for comparing pressures. Pneumatic lines then run from the regulator output to pressure regulating and shutoff valve mechanism(s) which modulate valve position accordingly using the pressurized air signal from the regulator.

DISTRIBUTION

The engine bleed air distribution system interconnects the engine bleeds of the engines and APU and contains the necessary valves to shut off bleed air at each engine and isolate various ducts. The medium pressure pneumatic system is generally characterized by the use of 3-4 inch diameter ducting. The pneumatic manifold, which is itself ducting, distributes the air through the use of control valves leading to various pneumatic system components and sub-systems. The ducts into which the control valves direct the air are of various sizes. High volume ducting (3-4 inch diameter) is used for engine starting and anti-ice and air conditioning. Smaller diameter ducting is used for many other components such as windscreen anti-fogging and total air temp gauge anti-ice.

The temperature of pneumatic air is controlled within an acceptable range. Typically, some sort of heat exchanger is located in the bleed air portion of the pneumatic system for this purpose. A working temperature in the pneumatic manifold of close to 200°C is normal. By controlling the quantity of the overall volume of air that passes through the heat exchanger, the pneumatic manifold temperature is regulated.

Air conditioning systems accept air from the pneumatic manifold that is too hot to be released directly into the cabin. Air conditioning packages use heat exchangers and well as an air cycle machine to adjust the temperature of the pneumatic air so that it is comfortable in the cabin. Anti-ice and engine starter air temperatures are not as critical and make use of pneumatic manifold air without further temperature adjustment.

There is a limit to the amount of ozone present in the pneumatic air processed by the air conditioning system that is sent to the cabin. Above the limit, passengers experience symptoms such as headache, respiratory problems, and even cancer with long term exposure. An ozone converter in the distribution line to the cabin reduces the ozone concentration in the pneumatic air. The ozone converter uses special, active, oxide coated surfaces to remove the ozone. Some converters are constructed to remove hydrocarbons and carbon monoxide contaminants/odors as well. The process involves a chemical reaction between the coated metal plates and the hot air. ***Figure 16-5*** illustrates a typical ozone converter in a pneumatic duct.

Note that the air cleaners shown in ***Figure 16-5*** do not affect ozone levels. Instead this type of cleaner in a pneumatic system is used to remove airborne particles. A swirling motion is induced such that the

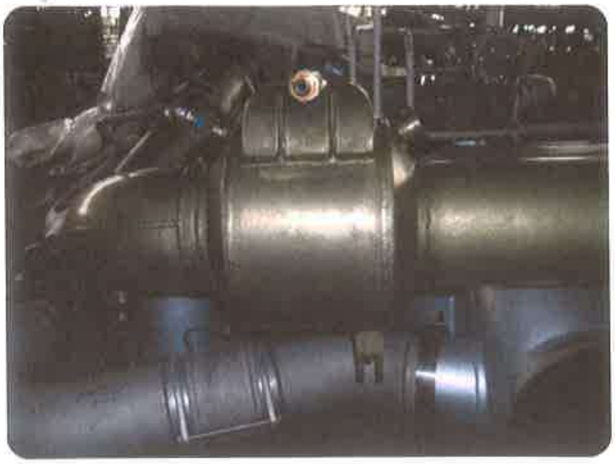


Figure 16-5. Catalytic ozone converter.

heavier particulates are separated. They are moved to the outside of the cleaner housing by centrifugal force. An electrically controlled pneumatically operated purge valve removes the particles.

PRESSURE AND VACUUM PUMPS

OVERVIEW

Pumps used in hydraulic power and those used in pneumatic power, have a lot in common; as do pumps used in engine oil and aircraft fuel systems. The biggest difference with pneumatic pumps is the fluid they handle is a gas and not a liquid.

The purpose of the pump in each of these systems is the same; to increase the pressure of the fluid and to use that pressure to accomplish a task such as move a flight control, raise the landing gear, pressurize the airplane, start a turbine engine, or countless other possibilities. When studying a large aircraft, one discovers that a greater variety of things are accomplished using pneumatic power than with hydraulic power.

All pumps, whether mechanical in nature (gear type, gerotor type, vane type, piston type, diaphragm type, centrifugal type, Roots blower type) or those based on Bernoulli's principle (ejector type), operate under the same basic concept. By forcing a fluid out at the discharge side of the pump, a partial vacuum is created at the inlet side of the pump. The supply of fluid the pump is designed to handle is at normal atmospheric pressure, or possibly higher, and this higher pressure forces the fluid into the pump's inlet. This concept can be viewed when someone sucks on a straw inserted into a glass of water and creates a partial vacuum in the straw.

The liquid in the glass is at normal atmospheric pressure, which forces the liquid into the straw and we are able to drink.

Pumps are generally viewed as devices that add pressure to a fluid to accomplish some type of work. Most pressure pumps can also be viewed as vacuum pumps. The discharge side of the pump is at a pressure higher than ambient, but the inlet side of the pump is at a pressure lower than ambient (partial vacuum). So pressure pumps and vacuum pumps are not different types, the classification is simply based on what the pump is being used for. In a small airplane, for example, the inlet side of a vane type pump can create the vacuum that powers the gyros in attitude instruments and at the same time, the discharge side can send pressurized air to the cabin. In this case, the device is acting as both a vacuum and a pressure pump.

CLASSIFICATION OF PUMPS

The pumps used in pneumatic systems can be classified as positive or non-positive displacement. Positive displacement means that in one complete revolution of the pump, a fixed amount of fluid will be discharged. If one revolution produces 2 ounces of fluid, then five revolutions displaces 10 ounces. A positive displacement pump is often called a constant-displacement, constant-volume or constant-delivery type of pump. As long as there is no slippage of fluid within the pump, the output per revolution will remain the same. A non-positive displacement pump, such as a centrifugal impeller type, experiences a lot of fluid slippage as the pressure increases and it cannot maintain a consistent output per revolution.

TYPES OF PUMPS

When it comes to the pneumatic system of the aircraft, the types of pumps typically used are the wet-type or dry-type vacuum pump, the multiple piston pump, the centrifugal impeller pump and the roots blower pump. A very common source of pneumatic power is the compressor of an aircraft's turbine engine, which in addition to pumping air into the engine's combustion chamber can also pump air into the aircraft.

Vane Pump, Wet Type and Dry Type

The vane type power pump is a constant-displacement pump. It consists of a housing containing four vanes (blades), a hollow steel rotor with slots for the vanes, and a coupling to turn the rotor. (***Figure 16-6***) The only difference between the wet or dry type pump is the method used to lubricate the rotating vanes.

The rotor is positioned off center within the sleeve. The vanes, which are mounted in the slots in the rotor, together with the rotor, divide the bore of the sleeve into four sections. As the rotor turns, each section passes one point where its volume is at a minimum and another point where its volume is at a maximum. The volume increases from minimum to maximum during the first half of a revolution and decreases from maximum to minimum during the second half. As the volume of a given section increases, that section is connected to the pump inlet through a slot in the sleeve. Since a partial vacuum is produced by the increase in volume of the section, fluid is drawn into the section through the pump inlet and the slot in the sleeve. As the rotor turns through the second half of the revolution and the volume of the given section is decreasing, fluid is displaced out of the section through the slot in the sleeve aligned with the outlet port, and out of the pump. This type of pump moves four pockets of fluid in one revolution.

Piston Pump

For many turbine engine powered aircraft, a high pressure bottle filled with compressed gas is carried on board. In an emergency, the high pressure gas can be used to apply the brakes or force the landing gear to extend. The source to fill these high pressure bottles comes from a multi-stage piston pump, which is typically land based so the bottle cannot be refilled in flight.

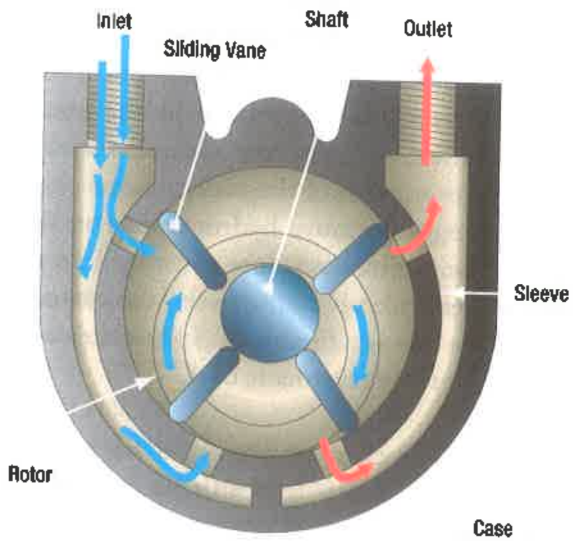


Figure 16-6. Vane-type pneumatic pump.

Centrifugal Impeller Pump

Pressurized airplanes that have turbocharged reciprocating engines often use the centrifugal compressor (impeller) in the turbocharger to both pressurize the airplane and to supercharge the engine. The compressor takes in outside air and directs it into the center of the rotating impeller. (***Figure 16-7***) The impeller throws the air to the outside (centripetal acceleration), increasing both the velocity and the pressure of the air. The fins on the impeller form diverging passages when flowing from the center to the outside, and this diverging shape is why the pressure increases in addition to the velocity increasing. The increased velocity is converted to additional pressure as the air leaves the impeller and flows through another diverging passageway. This high-pressure air is used to increase the engine's power and to pressurize the aircraft.

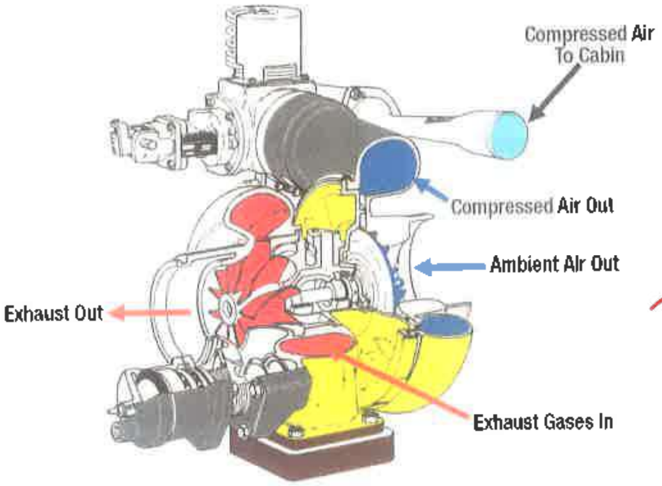


Figure 16-7. Turbocharger for pressurization (pneumatic system pump).

Turbine Engine Compressor

Most large aircraft rely on the compressor found in the turbine engines as a source of pneumatic power. For large airplanes the compressor would be an axial flow type, utilizing airfoil shaped blades and vanes to pump the air from front to back while steadily increasing its pressure. Some of this air can be bled away and sent to the airframe to be used for a variety of purposes, to include cabin pressurization, wing anti-icing, pressurizing hydraulic reservoirs and pressurizing water tanks. The turbine engines in smaller aircraft might use a centrifugal impeller type of compressor, which was discussed earlier.

Roots Type Blower

Some small turbine powered airplanes, have used a roots type blower as a source of pneumatic power. This device is made up of two lobed rotors that rotate very close to each other, without touching, with air entering the space between the lobes and being compressed. (***Figure 16-8***)

PNEUMATIC/VACUUM

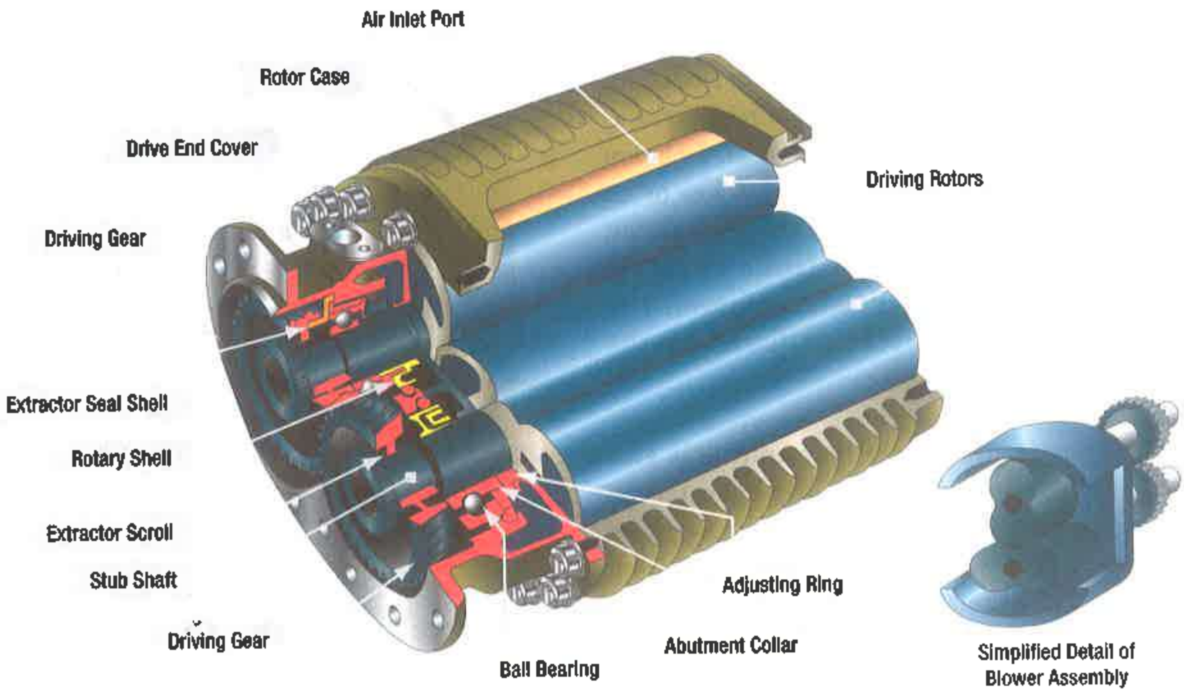


Figure 16-8. Roots type blower (pneumatic system pump).

Vacuum Pumps

Almost all pressure pumps are capable of functioning as a vacuum pump, because the inlet side of a pressure pump is under a partial vacuum when it is operating. The vane pump, (***Figure 16-6***), is the most common type. The suction side of this pump is typically used to pull air through the gyros in attitude indicating instruments. The output of the pump, which is at an increased pressure, can either be dumped overboard or used to perform tasks like inflating de-icer boots or pressurizing a small cabin.

FLIGHT DECK INDICATIONS

On the flight deck of an aircraft with a pneumatic power system, the following indications might be seen:

1. A gauge showing the pressure in the pneumatic manifold.
2. A gauge showing the temperature of the air in the pneumatic manifold.
3. An advisory light to indicate if the pressure or temperature in the pneumatic manifold is not within limits.
4. If the pneumatic system includes a filter, there might be an advisory light to indicate if the filter becomes clogged and starts bypassing.

INDICATIONS AND WARNINGS

There are few indications and warnings associated with the pneumatic system. Pneumatic manifold pressure is a key parameter monitored on the flight deck. Large aircraft typically have a pressure transmitter mounted in each section of the pneumatic manifold associated with an engine. These transducers send an electric signal to a dual gauge on the pneumatic control panel. Isolation valve control switches are located nearby. A low or no pressure situation can be handled by closing an isolation valve and using the remaining pressure to supply all pneumatic requirements. Pressure transmitters are connected to a dual pressure indicator on the overhead panel.

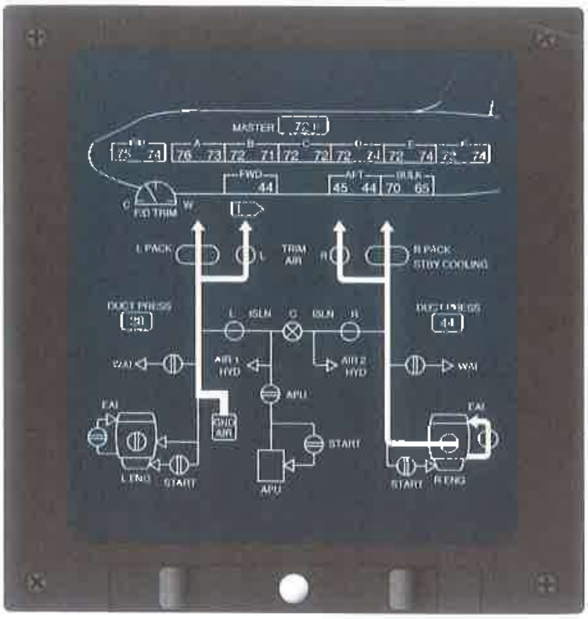
Engine bleed pneumatic system temperature is also monitored. The switches are wired to trip lights on the flight deck overhead panel. The temperature switches illuminate the trip lights and the corresponding engine bleed valve closes automatically when the bleed air temperature exceeds approximately 250°C.

On digital aircraft, pneumatic system operation is completely automatic. Redundant computer controlled systems factor all supply and demand parameters including pneumatic pressure and temperature as in a non-digital aircraft. Valve positions and flow data are also factored. Over pressure and over temperature bleed air and duct leak conditions cause protective shutdown of the affected part of the system. BITE equipped components provide self monitoring information to the central maintenance computer system. Synoptic display of pneumatic air user system parameters and those of the pneumatic system itself is given via a multi-functional display panel. (***Figure 16-9***)

For the pneumatic system, the air synoptic display shows the following information:

- Ground air in use.
- Duct pressures.
- Engine bleed air pressure regulating and shutoff valve position.
- Isolation valve position.
- APU shutoff valve position.

An × on an isolation valve symbol or the APU shutoff valve symbol shows the valve has failed or the switch on the bleed air/pressurization panel for the valve is in the non-normal position. An air supply maintenance page viewable on the multi-functional display is shown in ***Figure 16-10.***



Synoptic Display

Figure 16-9. Synoptic display of a modern pneumatic system.

Pneumatic Air Supply Display

Figure 16-10. Pneumatic air supply maintenance page.

INTERFACE WITH OTHER SYSTEMS

Pneumatic systems interface with other types of systems on many aircraft. The most common interface is with portions of the aircraft hydraulic system. As mentioned in the discussion on high pressure pneumatic systems, interface of emergency high pressure pneumatic system air with a normally hydraulic braking system is common.

Use of a shuttle valve prioritizes the flow of air and directs it into the brake actuating mechanisms. The seals installed for hydraulic use are sufficient for use in the one type deployment of emergency brakes by pressurized air. Hydraulic system actuators are designed primarily for use with hydraulic fluid. Use with air is limited to one time emergency operations.

However, pneumatic power may be used to supplement and backup hydraulic system components without loss of performance. This is done by turning a hydraulic pump with pneumatic power. The pneumatically driven hydraulic pump then supplies the hydraulic system components with fluid in the usual engineered manner. Cross utilizing hydraulic components with pneumatic air is eliminated. Traditional benefits of hydraulic power are retained such as those from the incompressibility of the fluid.

Control of a pneumatically driven hydraulic pump is through the use of a control valve in the pneumatic manifold. Selection of the pump via a switch on the flight deck causes the control valve to open and supply pneumatic air to drive to the pump. All hydraulic system controls are then operated normally either by the flight crew or automatically by computer.

Question: 16-1
In what major way is a pneumatic system different from a hydraulic system?

Question: 16-2
In what circumstance is an oil separator required in an aircraft pneumatic system?

Question: 16-3
What three notifications are always stamped on the outside of a high pressure storage tank?

Question: 16-4
What is the purpose of a standpipe in a high pressure pneumatic system?

Question: 16-5
In what circumstance is a restrictor valve used in a pneumatic system?

Question: 16-6
What is the purpose of a check valve in a pneumatic system?

Question: 16-7
What is the typical pressure source for a medium pressure pneumatic system?

Question: 16-8
In what way is the temperature of compressed air controlled in a medium pressure hydraulic system?

ANSWERS

Answer: 16-1
Gaseous fluids are compressible; liquid fluids are incompressible.

Answer: 16-2
When storage bottles are recharged with on board compressors.

Answer: 16-3
Date of manufacture; maximum safe pressure; date of last hydrostatic test.

Answer: 16-4
To purge moisture from the system as air exits the storage bottles.

Answer: 16-5
When a component to be actuated requires less pressure then what exists in the storage bottle.

Answer: 16-6
To prevent backflow to the gas to the compressor.

Answer: 16-7
Turbine engine compressor bleed air.

Answer: 16-8
By regulating the volume of air flowing into a heat exchanger.

INTEGRATED MODULAR AVIONICS (ATA 42)

SUB-MODULE 17

PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY →	B1.3 B1.4

Sub-Module 17

INTEGRATED MODULAR AVIONICS (ATA 42)

Knowledge Requirements

12.17 - Integrated Modular Avionics (ATA 42)	
Functions that may be typically integrated in the Integrated Modular Avionic (IMA) modules are, among others: Bleed Management, Air Pressure Control, Air Ventilation and Control, Avionics and Cockpit Ventilation Control, Temperature Control, Air Traffic Communication, Avionics Communication Router, Electrical Load Management, Circuit Breaker Monitoring, Electrical System BITE, Fuel Management, Braking Control, Steering Control, Landing Gear Extension and Retraction, Tire Pressure Indication, Oleo Pressure Indication, Brake Temperature Monitoring, etc. Core System; Network Components.	2

12.17 - INTEGRATED MODULAR AVIONICS (ATA 42)

Avionics encompasses a wide range of systems that are used for navigation, communication, aircraft control and other purposes. On a modern aircraft, there are dozens of systems that can be considered as avionics systems. Integrated Modular Avionics (IMA) is a design methodology rather than a system as such. In other words, it describes how various avionics systems are assembled together rather than the specific functions of individual avionics devices. IMA advances avionics technology. Aircraft with IMA can achieve reductions in the size and weight of their avionics systems. In addition, due to the simplification, the overall reliability of the avionics can be improved. The following is just a partial list of functions that may be integrated into modularized systems:

- Bleed Management
- Air Pressure Control
- Air Ventilation and Control
- Avionics and Cockpit Ventilation Control
- Air Traffic Communication
- Avionics Communication Router
- Electrical Load Management
- Circuit Breaker Monitoring
- Electrical System Built In Test Equipment (BITE)
- Fuel Management
- Braking Control
- Steering Control
- Landing Gear Extension and Retraction
- Tire Pressure Indication
- Brake Temperature Monitoring

INTEGRATION OF AVIONICS

Design methods for avionics systems have evolved over time. Initially, avionics systems were individual stand alone systems. This means that each device was separate. For example, an aircraft compass system might have consisted of its own flux detector and its own gyroscope feeding data to a heading indicator on the instrument panel. These components were used and wired only within the compass system rather than being shared by other systems throughout the aircraft even though the needed data of those other systems might have been the same.

Traditionally, on non-IMA aircraft, each avionics system had its own separate indicator and its own separate controls. As more and more avionics systems were developed and installed in aircraft, more indicators and controls had to be installed. Instrument panels became more complex and crowded. ***Figure 17-1*** shows such a non-IMS panel on an older Bell 222 helicopter.

In addition, as more avionics systems were developed and installed, more Line Replaceable Units (LRUs) or "black boxes" were installed in avionics compartments. With this, more wiring was needed to interconnect these LRUs with their associated cockpit controls and indicators. And due to that, more electrical power was needed to operate these ever increasing numbers of systems.

Each additional indicator, LRU, and wire that is installed on an aircraft takes up space and adds weight. Because both space and weight carrying capability are at a premium, it is desirable to keep the number of indicators, LRUs, and wires to a minimum.

In the case of indicators, engineers began to develop designs that used the same indicator to display information from more than one system. For example, older designs had separate indicators for the compass system, the radio navigation system, and the weather radar system.

In newer designs, these systems are all connected to single, "integrated" indicators such as a navigation display. The use of integrated indicators saves space and weight, and it streamlines pilot workload by reducing the number of indicators that must be scanned during flight. (***Figure 17-2***)

Figure 17-1. Non-IMS panel on older Bell 222.

Figure 17-2. IMA display panel on a newer Bell 407.

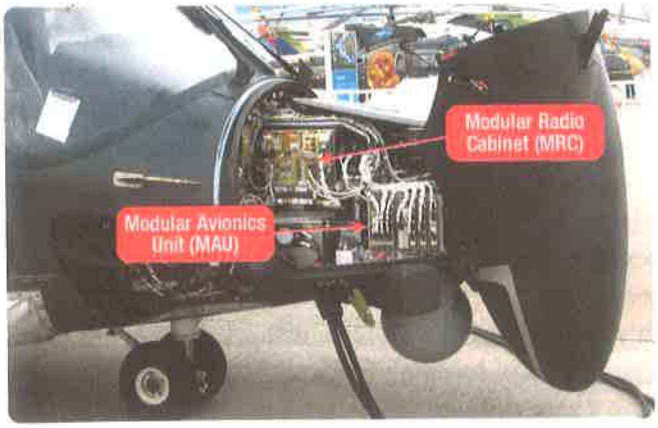


Figure 17-3. An IMA avionics bay located in the nose of an Agusta AW139.

Traditional (non-IMA) avionics suites have many separate LRUs located in an avionics compartment. In such systems, there is a considerable amount of duplication. For example, each box typically contains its own power supply. These power supplies receive aircraft power to provide the various voltages needed by the circuits within that box. Also, each power supply is connected to the aircraft electrical power system by separate wiring. These power supplies might be functionally identical to each other. In a non-IMA aircraft, there could be twenty identical power supplies in twenty separate boxes. However, if the power supply in one of the black boxes fails, that system fails because that box cannot use power from another box.

In an aircraft with IMA, some of the self contained boxes are replaced by common modules. The modules form the integrated system because they are plugged into a mainframe or rack which is a single piece of equipment. This results in a bulk and weight reduction because some circuitry is now shared among the various modules. (***Figure 17-3***)

For example, instead of having twenty power supplies, each serving a separate device, an IMA aircraft might have just three (1 primary and 2 redundant) power supplies that are each capable of supplying all twenty modules. If one of the IMA power supplies fails, there are still two redundant power supplies available. A failure of one, or even two, of the power supplies does not result in the failure of any of the avionics modules. Thus, the IMA aircraft is more fault-tolerant, resulting in better reliability.

As another example, often the same kind of data processing circuitry is required for various avionics functions. In non-IMA designs, this data processing circuitry had to be duplicated in separate LRUs for each separate system. With IMA, the data processing circuitry can be contained in fewer LRUs, and it is shared among the systems requiring it.

In addition less wiring is needed in an IMA aircraft. Using the example of the power supplies, with IMA there is no need to provide twenty separate wires from the aircraft electrical system for power supplies because there are only three power supplies for all IMA modules. In addition to this, the fact that IMA aircraft make extensive use of digital data buses also reduces the amount of wiring needed.

DIGITAL DATA BUS USE REDUCES WIRING

Several different digital data bus systems are used in various aircraft. Some of the more important ones are ARINC 429, ARINC 629, and Avionics Full Duplex (AFDX). Wires are used to transfer information from one piece of equipment to another. The use of digital data buses can also result in a tremendous reduction in the amount of wiring used.

An example that illustrates this reduction is radio tuning. When a radio is tuned, frequency information must be transferred from a tuning unit to the radio receiver or transceiver being tuned. This frequency information might consist of four or more digits. Say a pilot wishes to tune a VHF communications transceiver to the frequency 128.35 MHz. The pilot enters this frequency into the tuner. From there, it must be carried to the VHF transceiver located in the avionics compartment.

With avionics systems without digital data buses, each piece of information to be transferred from one location to another requires at least one separate wire. Often, a far larger number of wires are needed. Using our example of tuning to 128.35 MHz, each digit of the selected frequency must be transferred to the transceiver. Because there are five digits in this frequency, it might seem that five wires would be needed to transfer this data. However, because the information being transferred is complex (each digit might be anything from a zero through a nine), still more wires are needed. The previous method of accomplishing this was the ARINC 2×5 tuning scheme as shown in ***Figure 17-4.***

Two out of five (2×5) tuning requires the use of five wires for each digit of information being transferred. Of these five wires, two are connected to ground and three are not. The wires which are grounded determine whether the digit transferred is a zero, a one, a two, etc.

Since all VHF communication frequencies begin with the number "1," it is not necessary to transfer that digit to the transceiver. However, each of the other four digits in the selected frequency must transfer. Four digits at five wires per digit results in 20 wires. A ground wire is also needed, bringing the total number of tuning wires to 21. ***Figure 17-5*** illustrates this.

When using a digital data bus such as ARINC 429, this same tuning information can be transferred by using only two wires. With ARINC 429, the two information carrying wires are twisted together and covered with a braided shield. The shield is kept grounded and protects the two inner wires from electromagnetic interference. These two inner wires are referred to as a "twisted pair". (***Figure 17-6***)

The reason it is possible to transfer so much information on only two wires, is that the information is sent serially. This means that the same two wires carry each bit of information one bit at a time. One bit is sent, then another, and another. Soon, all the required information has been transferred from the tuning unit to the radio. The information transfer is done at a rapid rate. An ARINC 429 system can transfer up to 100 000 bits of information per second. ARINC 629 can transfer up to two million bits per second. Avionics Full Duplex Switched Ethernet (AFDX) is a newer digital data bus system able to transfer data at rates up to 100 million bits per second. Because of these speeds, the same two wires can be used to carry out multiple tasks simultaneously, and thus far fewer wires are needed offering a substantial weight savings.

2 × 5 Tuning Scheme

	0	1	2	3	4	5	6	7	8	9
A		✚	✚						✚	✚
B	✚	✚		✚	✚					
C			✚	✚		✚	✚			
D					✚	✚		✚	✚	
E	✚						✚	✚		✚

*Wires marked with ✚ are grounded.

Figure 17-4. Two out of five tuning scheme.

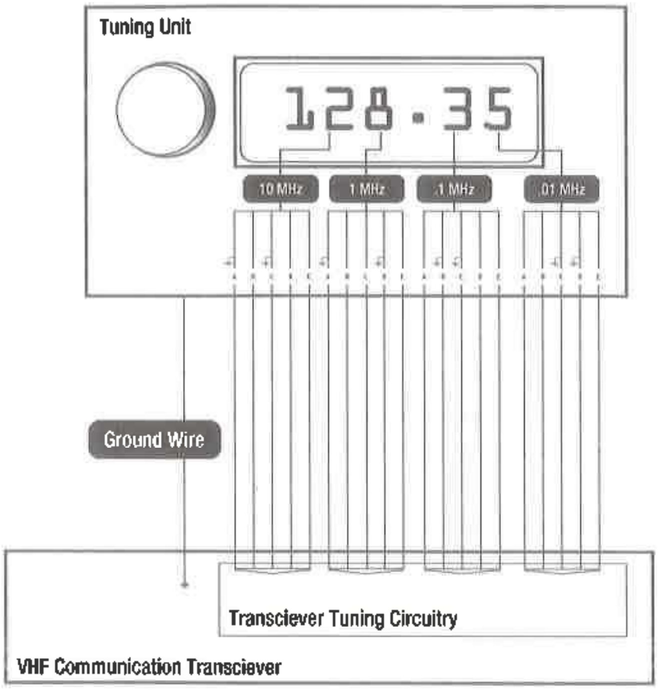


Figure 17-5. VHF Comm system tuning wires using two out of five tuning.

CORE SYSTEMS

IMA systems consist of a core system and network components. The core system contains the data processing circuitry that processes many kinds of information. This processing circuitry is shared by the various avionics systems that have been integrated. Information from various sensors, controls, and LRUs is brought into the core system for processing, then sent out to displays, actuators, and other places in the aircraft where that information is used.

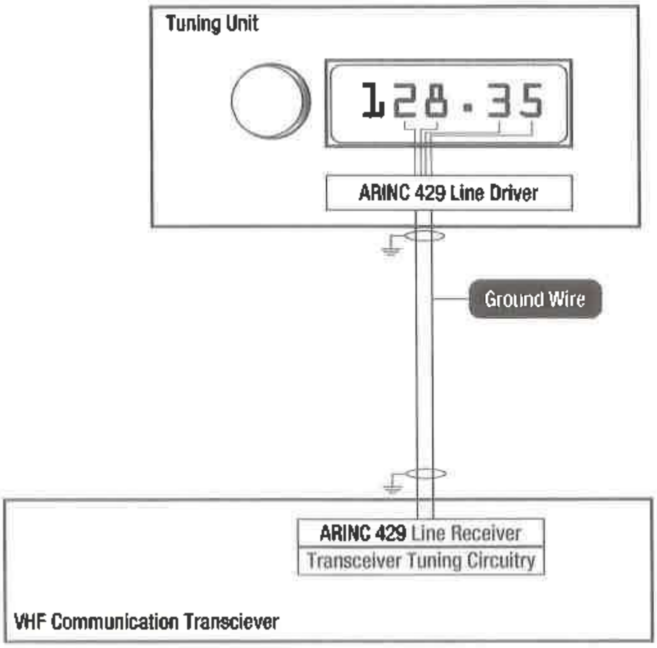


Figure 17-6. VHF Comm system tuning wires using ARINC 429 digital data bus.

The core system uses the same computer processors for many different purposes. For example, a core system can use the same processor for such tasks as calculating throttle settings for best fuel economy, calculating the amount of rudder deflection needed to coordinate a turn, monitoring an instrument landing system receiver for malfunctions, determining if the stall warning system needs to be activated, and many others. This sharing of data eliminates the need to have a unique processor in each component and for every system.

COMMON CORE SYSTEMS CONCEPTS

This IMA system consists of a common core with network components. The core system contains a central and backup computer and data processing circuitry. This processing circuitry is shared by the various integrated avionics systems. Information from various sensors, controls, and LRUs are brought into the core system from data concentration modules. Data is then sent out from the core system to displays, actuators, and other places on the aircraft where that information is used.

CORE PROCESSOR INPUT/OUTPUT MODULE (CPIOM)

The CPIOM is a standard hardware platform designed to host several independent functions. For example, CPIOMs for the utilities domain perform fuel management, measurement, and display, and can also control the landing gear extension systems, braking, and nose wheel steering software. CPIOMs in the energy domain use the standardized architecture to control electrical power distribution. CPIOMs in the cabin domain host interrelated functions such as cabin pressure control, air conditioning and ventilation, and can also integrate such systems as fire and smoke detection as well as the monitoring of aircraft doors and evacuation slides. Today, the CPIOM operative system is based on the latest version of the ARINC 653 standard. ARINC 653 manages time and space between avionics systems and normal operative systems.

NETWORK COMPONENTS

Networking defines the aircraft systems into specific functional areas, related to:

- Flight Controls and Auto Flight
- Cockpit
- Engine Control
- Energy (Electrical Power)
- Pneumatic and Cabin
- Fuel
- Landing Gear

These functional areas group computers like LRUs, hosting modules and/or LRUs with an AFDX interface that share a common interest or characteristics. These computers exchange operational and maintenance data between each other. For most of them, this exchange is completed through the ARINC 664 network. The network uses AFDX technology which is an Ethernet protocol adapted for in-flight use to make them ARINC 664 compliant.

The ARINC 664 standard provides a means to adapt commercial off-the-shelf networking protocols and hardware to an aircraft environment. It refers to devices such as bridges, switches, routers and hubs and their use in an aircraft environment. Manufacturers have developed their own proprietary systems to comply with ARINC 664 specification. For example, the Airbus system is called Avionics Full-Duplex Switched Ethernet (AFDX), and the Augusta Westland system is called the Avionics Standard Communication Bus, version-D (ASCB-D).

On the Augusta Westland AW139 helicopter, modular avionics units, the display units and the modular radio cabinets are directly connected to the ASCB-D. The ASCB-D network lets these units transmit and receive data at the highest possible speed. Each ASCB-D bus uses tapped twin axial wire. Two data buses are provided for redundancy and failure protection. Each data bus is made of sets of twisted wires that are electrically isolated with shield and resistor terminations.

The Local Area Network (LAN) data bus is available to all units that connect to the ASCB-D. The LAN is a high speed, non-essential bus without redundancy that is used for the maintenance and test functions. It is also used to upload the data and software in the modular avionics units and the modular radio cabinets. The maintenance is done on the ground by connection of a standard laptop computer to the LAN.

ARINC 664 is used for the exchange of operational, maintenance and loading data between subscribers, which are LRUs with an AFDX interface. The subscribing device can communicate at speeds of 10 or 100 Mbits per second. In case of total network failure, all essential data transmission is backed up using an ARINC 429 databus system. (***Figure 17-7***)

DATA TRANSMISSION ON THE NETWORK

Aircraft system data is sent simultaneously from a network subscriber to other network subscriber(s) on both redundant sub-networks A and B through AFDX switches according to predefined paths called Virtual Links. (***Figure 17-8***)

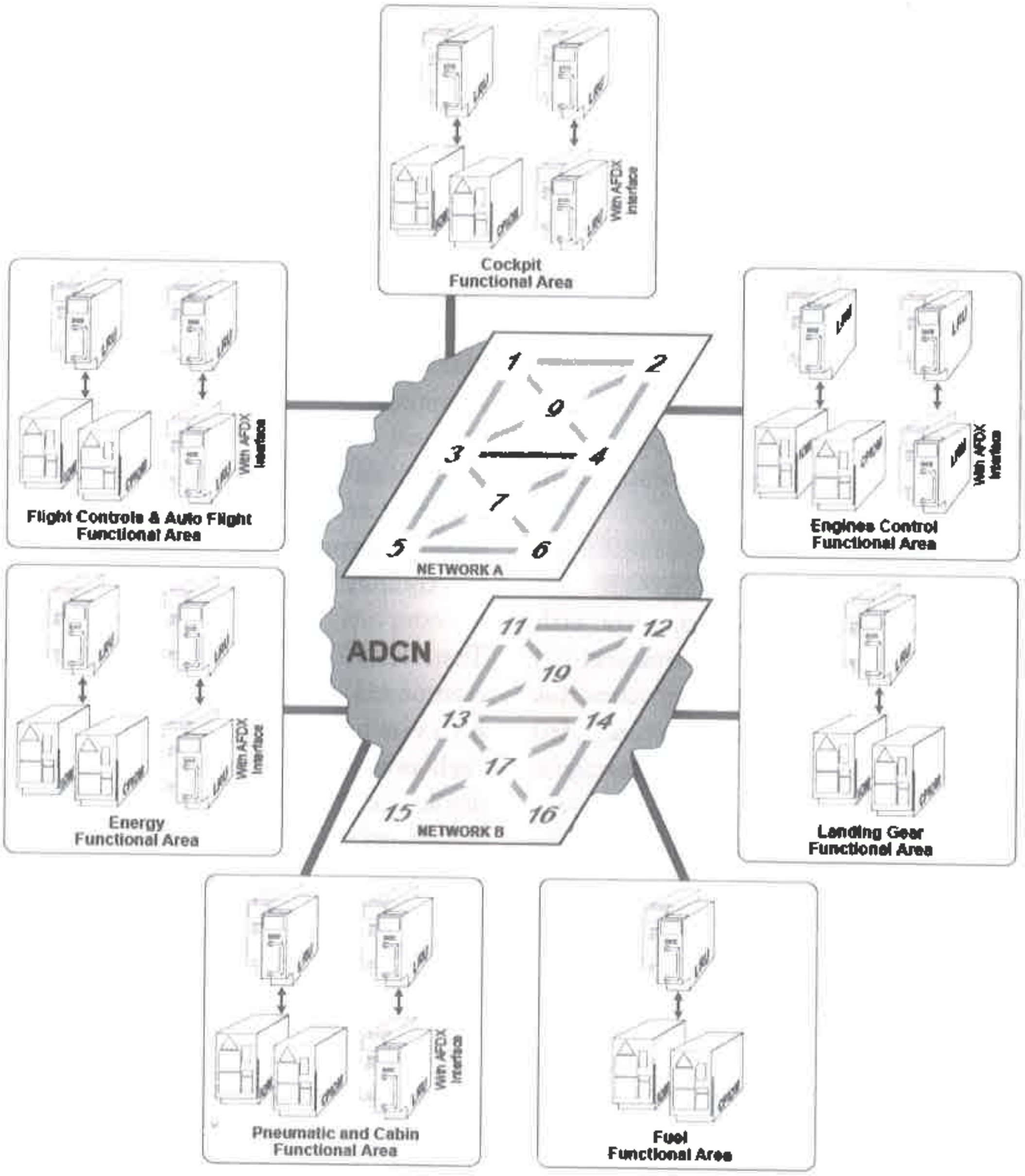


Figure 17-7. Avionics Data Communication Network interfaces.

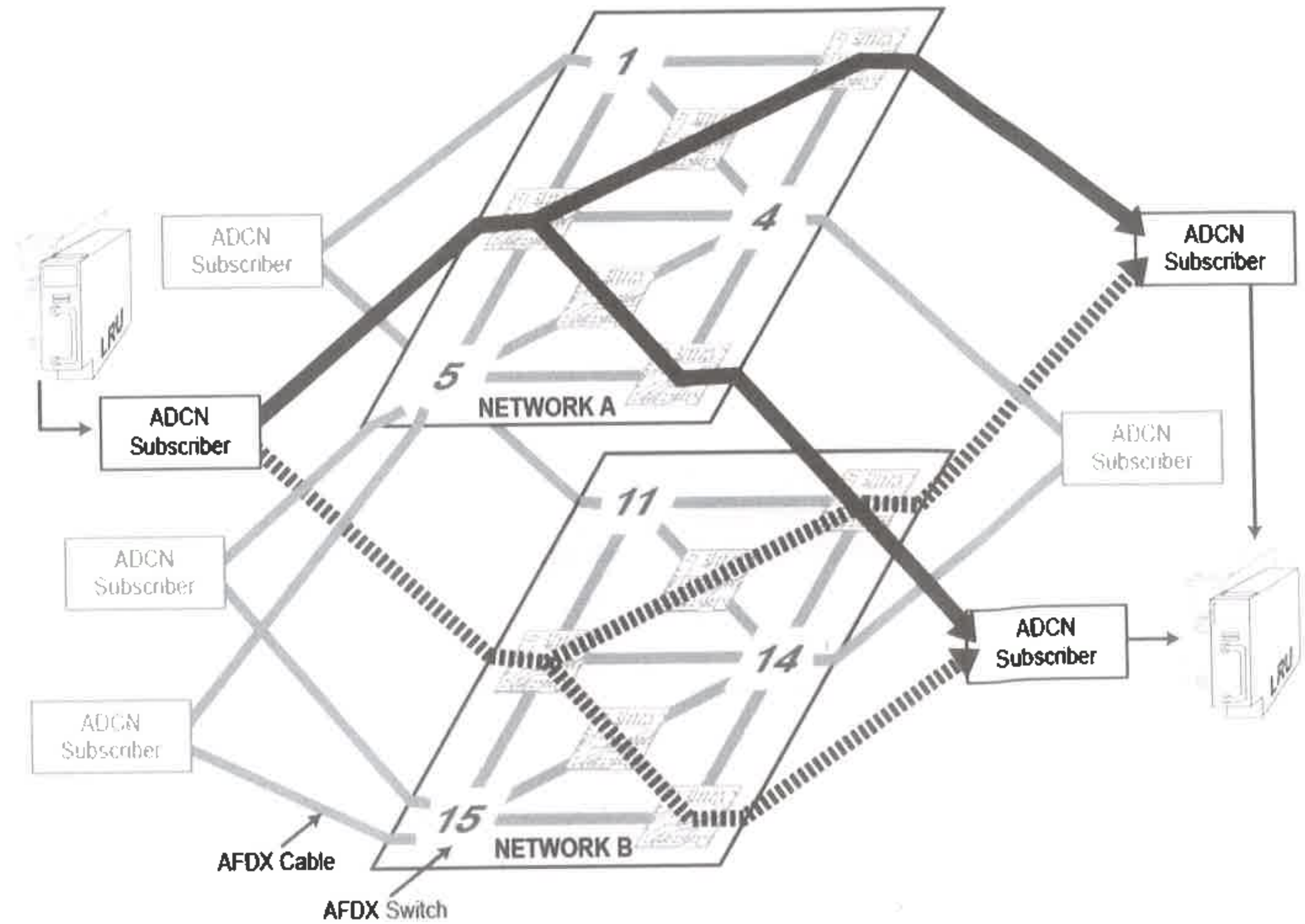


Figure 17-8. Data transmission using ADCN.

AFDX SWITCH

An AFDX switch is composed of various components, including the following:

- Hardware boards.
- A power supply board connected to the 28 volt DC bus.
- A switching board, which routes the AFDX frames according to a configuration table.
- An inputs/outputs board connected to other AFDX switches and network subscribers.

Field Loadable Module Software includes:

- One AFDX SW operational program software that operates the module.
- One AFDX SW configuration table software that provides the AFDX switch with configuration data. (Such as a switching board configuration, etc.).

AFDX Switch Software

AFDX switches host the following software:

- The AFDX SW operational program software.
- The AFDX SW configuration table software.

All AFDX switches are interchangeable but may require software reconfiguration. (***Figure 17-9***)

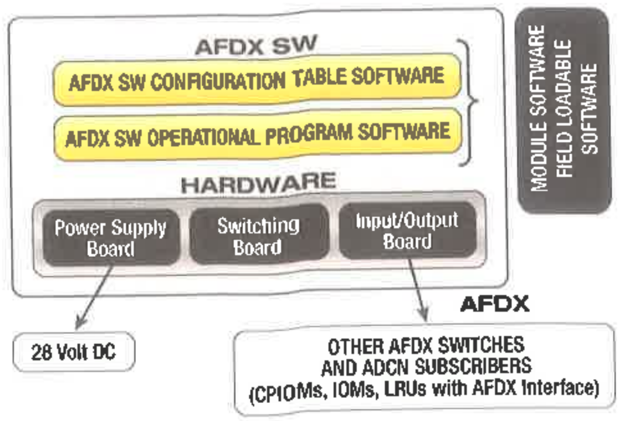


Figure 17-9. AFDX Switch internal components.

Question: 17-1
What is the primary purpose of Integrated Module Avionics?

Question: 17-2
State an example of an avionics component which can be shared with other components.

Question: 17-3
In what way is redundancy provided in an IMA system?

Question: 17-4
What is the general method by which an ARINC system replaces 2×5 wiring?

Question: 17-5
In a non-digital system, how many wires are required to change the tuning of a radio from 128.35 MHz to 129.35 MHz?

Question: 17-6
What are the 5 components of a "twisted pair" wire?

Question: 17-7
If an AFDX network fails, what is its primary backup?

Question: 17-8
What is the primary purpose of ARINC 664?

ANSWERS

Answer: 17-1
A reduction of components and inter-component wiring saving on weight and space requirements.

Answer: 17-2
Power supplies, data processors, indicators and display screens, wiring.

Answer: 17-3
Each shared component has one or two backups, each capable of taking over for the entire system should the primary fail.

Answer: 17-4
A same wire can serve multiple functions because of the speed in which that wire can transfer information.

Answer: 17-5
One digit was changed from an 8 to a nine. Thus, 5 wires were required.

Answer: 17-6
Two wires, one braided shield, two terminals.

Answer: 17-7
An ARINC 429 databus system.

Answer: 17-8
Provides a standardized method for AFDX networking protocols.

HELICOPTER AERODYNAMICS, STRUCTURES AND SYSTEMS

ON BOARD MAINTENANCE SYSTEMS (ATA 45)

SUB-MODULE 18

PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY →	B1.3 B1.4

Sub-Module 18

ON BOARD MAINTENANCE SYSTEMS (ATA 45)

Knowledge Requirements

12.18 - On Board Maintenance Systems (ATA 45)	
Central maintenance computers; Data loading system; Electronic library system; Printing; Structure monitoring (damage tolerance monitoring).	2

12.18 - ON BOARD MAINTENANCE SYSTEMS (ATA 45)

On board maintenance systems (also called central maintenance systems) are electronic systems used to facilitate the maintenance of modern aircraft. The exact configuration of on board maintenance systems varies from one aircraft model to another. However, the core functions of these systems are the same. They monitor the aircraft for faults, record and store the fault data, and provide information about these faults to flight crews and maintenance personnel.

The data collected by on board maintenance systems can be accessed both in flight and on the ground. In flight, the system advises the flight crew of faults that may affect aircraft operation. On the ground, maintenance crews use the system for testing and troubleshooting purposes. In some applications, the aircraft can radio fault information to the ground while in flight.

Moreover, on board maintenance systems can store data contained in maintenance manuals, flight manuals, and other publications. The systems allow maintenance personnel to access these publications without having to carry books and papers to the aircraft. On board maintenance systems allow such technical data to be uploaded, downloaded, viewed, and printed by maintenance personnel. The systems are used for both line and base maintenance. For the most part, the information provided here is general. When aircraft specific information is given, it should be noted that system details and terminology differ somewhat among the various aircraft manufacturers.

On Board Maintenance Systems are composed of several applications including:

- The Central Maintenance System
- The Data Loading and Configuration System
- The Aircraft Condition Monitoring System
- The Electronic Logbook (eLogbook)
- Minimum Equipment List (MEL) and Configuration Deviation List

CENTRAL MAINTENANCE COMPUTERS

The Central Maintenance Computer (CMC) is the main processing unit for an on board maintenance system. Like all computers, the CMC has inputs and outputs and is programmable. Inputs to the CMC come from the various systems being monitored, which are located all over the aircraft. Outputs from the CMC are provided in the form of visual displays, printed text, and digital data that may be downloaded. ***Figure 18-1*** shows a block diagram of the Airbus central maintenance system.

It is common for an aircraft on board maintenance system to have two redundant CMCs. With dual CMCs, all data inputs are available to both units. One CMC will be "active" at any given time. The other CMC is on "standby". The active CMC is the one providing outputs. If the active CMC should fail, the standby can immediately replace it. In some systems this changeover occurs automatically. The system monitors itself, and when it senses the failure of the active CMC, it automatically switches to the standby CMC. The switching can also be done manually.

CMCs are controlled by control units in the cockpit, such as within the Multipurpose Control Display Units (MCDU). The MCDUs enable the user to navigate the on board maintenance system by selecting from various on screen menus. These menus allow the user to view both current faults and fault history. Current faults are, of course, important for determining the aircraft current status. Fault history can be useful for monitoring trends such as recurring failures of a particular component. In addition a CMC menu permits the user to check the status of individual systems, even if no fault condition is present.

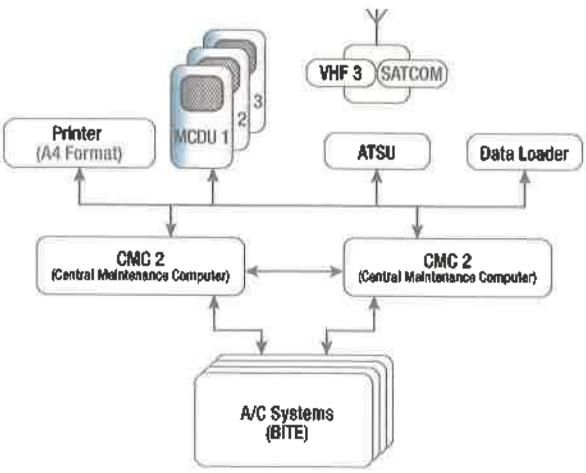


Figure 18-1. Airbus central maintenance system.

Figure 18-2 shows an example of navigating a central maintenance system menu on a MCDU. In this example, there are three faults: A bleed air system fault, and anti ice system fault, and an electrical system fault. The menu allows the user to access specific information about each fault that was sensed, such as the date and time the fault occurred.

In addition to the cockpit mounted control units, some on board maintenance systems allow for the connection of a personal computer. When the personal computer is connected, it can be used to access the data stored in the CMC. Reports of current faults and fault history can be downloaded to the personal computer.

Some types of system failures will immediately affect the operational capability of the aircraft, while other failures have no immediate impact. Because of the redundancy designed into aircraft for safety purposes, some faults can be tolerated. The Minimum Equipment List (MEL) determines which faults must be corrected before further flight, and which faults may be deferred for correction later. The fault indications provided by the CMC should be compared with the MEL to determine whether the aircraft can be dispatched.

The CMC classifies faults according to their severity. More severe faults will trigger cockpit indications for the flight crew so that their effect on operations can be evaluated. Less severe faults will not be displayed to the flight crew but will simply be stored and dealt with by maintenance personnel after the flight.

As an example, a CMC divides faults into three classes: Class 1, Class 2, and Class 3.

Class 1 faults are the most serious and involve something listed in the MEL. A Class 1 fault may ground the aircraft (a NO GO condition), or it may limit the conditions under which the aircraft may fly (a GO IF condition). An example of a GO IF, Class 1 fault would be the failure of a pressurization system component that limits the aircraft to unpressurized operation only. Class 1 faults are indicated in the cockpit by warning or caution lights, by failure messages on indicator screens, or by flags on the flight instruments. These indications are referred to as "flight deck effects."

Class 2 faults are less serious. The aircraft can be dispatched with a Class 2 fault because operational capability is not compromised (a GO condition). When a Class 2 fault exists in a system, the system is still functioning normally, although it may not have full redundancy. The flight crew is provided with a notification that the fault has occurred, in the form of a "maintenance status" message. Repair of Class 2 faults may be deferred for a period in accordance with the operator approved maintenance program.

Class 3 faults are minor discrepancies within a monitored system. They do not affect the operation of the aircraft. The flight crew is not notified of Class 3 faults. Maintenance personnel can access the CMC's record of Class 3 faults and make the repair when convenient.

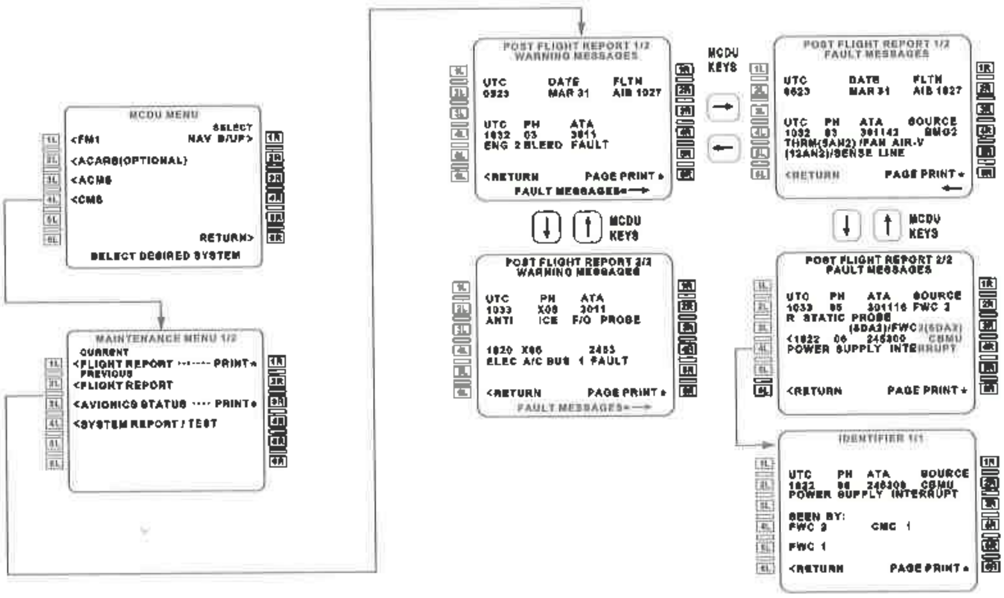


Figure 18-2. Example of Maintenance Menus.

Figure 18-3. The ACARS system.

The information gathered by central maintenance computers can be relayed to the ground through the Aircraft Communication Addressing and Reporting System (ACARS). ACARS is a data link system that uses the aircraft VHF communication radio, and in some aircraft the Satellite Communication (SATCOM) radio. Worldwide, a network of ground stations can communicate digitally with aircraft using this system. *Figure 18-3* shows the ACARS system.

Although the on board maintenance system can be connected to the ACARS, the latter is a stand alone system. It can be manually used by the flight crew to send messages, and it automatically sends reports about occurrences not associated with the on board maintenance system. ACARS sends data automatically when the on board maintenance system detects a serious fault. This alerts the maintenance personnel about the faults before the aircraft arrives, allowing them more time to prepare for dealing with the faults. Parts or LRUs can be pulled from stock and are available as soon as the aircraft lands. This can reduce ground time, helping to keep the aircraft on schedule.

BUILT-IN TEST EQUIPMENT (BITE)

Units that are monitored by the CMC contain special circuits known as Built In Test Equipment. BITE is installed in many systems throughout the aircraft including navigation systems, flight control systems, environmental control systems, and others. Within each system, the BITE circuitry tests numerous individual parameters to determine whether the system is functioning properly. The individual system's BITE circuits are connected to the CMC by a digital data bus. ARINC 429 buses are used for this purpose in many aircraft. Other data buses, such as ARINC 629, may also be used.

Whenever a system that contains BITE is first powered on, the BITE automatically performs a test of that system. This is referred to as an initialization test or a power-up check. If any fault is detected by the BITE during this test, an output is generated and sent to the CMC. If the system passes the initial test, BITE begins its regular monitoring of the system parameters. This monitoring is sometimes referred to as a "watchdog" function. During operation, the monitoring process is continuous. If anything that is being monitored fails, BITE will alert the CMC automatically.

In some aircraft, the user can run the BITE power-up check for a given system from the CMC control unit at any time. This capability is provided as a CMC menu item. This function can be useful when troubleshooting the system. Some LRUs containing BITE have indicator lights that indicate the status of the LRU. Green lights indicate a normal condition, red lights indicate that the BITE detected a fault in the LRU. ***Figure 18-4*** shows an LRU with BITE indicators.

BITE systems also have the capability of storing fault history. The history is kept in non-volatile memory. Non-volatile memory holds the stored information even after the system has been powered off.

DATA LOADING SYSTEMS

An aircraft data loading system provides a means to upload and download data to and from the on board maintenance system. The data loading system connects to other on board systems, as well. It can be used with any digital system that requires data uploads and downloads while installed in the aircraft.

Early data loading systems used floppy disks as the data storage medium. An example of this is the Multipurpose Disk Drive Unit (MDDU) used on many Airbus models. The MDDU uses 3.5-inch floppy disks for uploading, downloading, and data storage. In the Airbus system, a data loader selector switches the MDDU to the various systems that require a data upload or download. On other aircraft, data loading is accomplished through a Maintenance Access Terminal (MAT) on the flight deck. ***Figure 18-5*** shows a typical MAT.

Data loading systems also allow for the use of other forms of storage media. Newer systems can be connected to a laptop via a USB cable. A CD-ROM disk, or a USB memory stick or "flash drive" may also be used. In some aircraft, there are multiple locations to connect external devices to the data loading system. ***Figure 18-6*** shows various types of data loading panels.

The primary uses for the data loading system are for the uploading of program updates, the uploading of database updates, and the downloading of reports. An example of a unit requiring program updates is the central maintenance computer which contains an operating program that is upgraded from time to time. The same is true for other aircraft systems with internal programming. The number of systems requiring program updating varies from aircraft to aircraft.

An example of a database requiring updating is the navigation database which forms a part of the Flight Management System. The navigation database contains a great deal of information used by the flight crew. This includes the location of airports, airways, way points, and intersections, the location and frequencies of radio navigation aids, and other information needed to create and follow a flight plan. Because changes to this information occur from time to time, the navigation

Figure 18-4. Built In Test Equipment (BITE) Indicators.

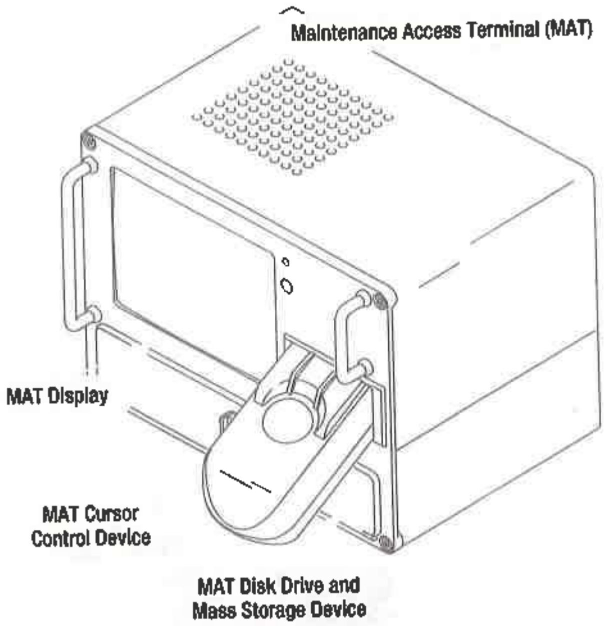


Figure 18-5. Maintenance Access Terminal (MAT).

Figure 18-6. Various types of data loading panels.

database requires periodic updates. These updates are uploaded through the data loading system. The standard frequency for navigation database updates is every 28 days. *Figure 18-7* shows examples of navigation database update software.

ELECTRONIC LIBRARY SYSTEM

An Electronic Library System (ELS) consists of databases containing information used by flight crews and maintenance personnel. These databases can include maintenance manuals, illustrated parts catalogs, wiring diagram manuals, flight manuals, service bulletins, and many other kinds of documentation from the manufacturer or the aircraft operator. The ELS takes the place of paper manuals. This results in a weight savings and can make the information access in the manuals quicker and easier. The system has the capability of storing the equivalent of hundreds of pounds of paper manuals in its computer memory.

Figure 18-7. Navigation system update software.

The databases in an electronic library system can be accessible through an on board display terminal and keyboard. They can also be accessed by an external personal computer, or through another digital device such as a tablet or a smart phone. The laptop or other external device is typically connected to the system by means of a serial bus cable.

The databases in an ELS must be periodically updated, as revisions are made to the technical data contained in the manuals. These revisions can be input through the data loading system.

PRINTING

Many helicopters have the capability to print out paper copies of reports from the on board maintenance system, as well as other documents. Aircraft printers typically conform to ARINC Standard 744A, which standardizes the technical requirements for such printers. Most aircraft printers are the thermal type, with some having the capability to print high resolution alphanumeric text, as well as graphical images on paper up to 21 cm (8.5 inches) wide. *Figure 18-8* shows an example of a basic thermal aircraft printer.

The speeds of aircraft printers vary, depending on the specific model of printer, and on what is being printed. Text generally prints faster. Images take longer. Print resolution also varies. A standard resolution is 300 dots per inch (dpi), but some printers are capable of better. The paper supply for aircraft printers comes in the form of 150 foot long rolls and may be perforated or non-perforated.

Figure 18-8. An ACARS compatible flight deck thermal printer.

Inside the printer, an electric motor is used to advance the paper. The printer uses a thermal print head, and the paper is heat sensitive. For this reason, care must be taken to keep the paper away from heat sources and out of direct sunlight while it is being stored. Exposure to heat can darken the paper, making it unusable for printing.

Aircraft printers receive input from CMCs, the ACARS system, and other sources by means of data lines, which may be ARINC 429 buses or Ethernet cables. Some printers can receive input wirelessly and operate as part of a wireless Local Area Network (LAN).

A typical aircraft printer is equipped with an indicator light to show whether the power is on or off. It will also give an alert when the paper supply is running low. Some printers perform a self test on power-up and will provide an indication if a fault is found during the test.

STRUCTURE MONITORING

Structure monitoring, also known as damage tolerance monitoring, has been recognized as an important function in aircraft maintenance. As aircraft age, their structure becomes more susceptible to damage caused by fatigue. Repairs and alterations can change the structural characteristics of an aircraft, introducing different stresses that were not existing with the original design. Corrosion can seriously weaken an aircraft structure. Also, events such as hard landings can lead to structural damage which may be difficult to detect.

Certification regulations require aircraft manufacturers to identify critical areas of the aircraft structure. These areas are known as Fatigue Critical Structures (FCS). These critical structures are identified by performing fatigue testing on test articles which are subjected to repeated load cycles until they fail. The results of this testing are analyzed to determine the FCS for the aircraft. Operators are required to monitor all FCS on their aircraft. This monitoring is intended to detect cracks and other structural deformations before they reach critical proportions, resulting in catastrophic failure.

The FCS monitoring process is accomplished by performing Damage Tolerance Inspections (DTIs) which specifically focused on fatigue critical structures. The aircraft DTI program will state where to inspect, how to inspect, and how often to repeat the required inspections. DTI inspections may be accomplished using visual inspection, eddy current, penetrant, X-ray, or other methods.

In addition to these, strain sensors may be used for structure monitoring. A strain sensor is a device that is bonded to a critical point on the structure. If the structure at that point becomes deformed, the strain sensor also becomes deformed. This deformation changes the electrical characteristics (typically the resistance) of the sensor. When the electrical characteristics of the sensor changes, a structural deformation is indicated.

Effective structure monitoring is crucial for preventing accidents caused by structural failure. For this reason, all data gathered during damage tolerance inspections must be recorded and carefully evaluated to ensure that the aircraft remains structurally sound.

Question: 18-1
What will happen if an active Central Maintenance Computer (CMC) senses a failure within itself?

Question: 18-2
What class fault will be declared by the CMC if a failure is detected in an anti-icing system?

Question: 18-3
What action must the flight crew take if a Class 3 fault is detected?

Question: 18-4
What will be the purpose of the ACARS system in the event a class 1 or 2 fault is detected?

Question: 18-5
What is an example of information stored in the non-volatile memory of a BITE system?

Question: 18-6
What three maintenance related documents are always stored in the aircraft's Electronic Library System?

Question: 18-7
What piece of equipment on board a helicopter is built in accordance to ARINC 744A?

Question: 18-8
What is the basic operating principle of a strain detector as related to airframe structural monitoring?

ANSWERS

Answer: 18-1
It will automatically switch to the standby CMC.

Answer: 18-2
Class 1 GO-IF. It is OK to operate so long as icing conditions can be avoided.

Answer: 18-3
None. The flight crew likely will not even be notified of a Class 3 fault.

Answer: 18-4
To notify maintenance personnel on the ground that a particular service will be required upon arrival.

Answer: 18-5
A history of earlier faults.

Answer: 18-6
Manufacturer's maintenance manual, Wiring diagrams, Illustrated parts catalog.

Answer: 18-7
The on board printer..

Answer: 18-8
If an object on which a strain detector deforms, its electrical resistance will change.

HELICOPTER AERODYNAMICS, STRUCTURES AND SYSTEMS

INFORMATION SYSTEMS (ATA 46)

SUB-MODULE 19

PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY – B1.3 B1.4

Sub-Module 19

INFORMATION SYSTEMS (ATA 46)

Knowledge Requirements

12.19 - Information Systems (ATA 46)

Information Systems consist of the units and components which furnish a means of storing, updating and retrieving digital information traditionally provided on paper, microfilm or microfiche. They include units that are dedicated to the information storage and retrieval function such as the electronic library mass storage and controller. They do not include units or components installed for other uses and shared with other systems, such as the flight deck printer or general use display. 2

Typical examples include:
Air Traffic and Information Management Systems and Network Server Systems.
Aircraft General Information System;
Flight Deck Information System;
Maintenance Information System;
Passenger Cabin Information System;
Miscellaneous Information System.

ADOPTION BY THE HELICOPTER OPERATORS

While the airline industry and similar fixed wing operators have been quick to adopt mobile technologies, many helicopter operators have been moving slowly. Typically this is due to small fleet sizes which can make it difficult to justify the transition to a paperless cockpit. For a private pilot, joining the digital revolution can be as simple as downloading an app, but for a commercial operator, complying with the relevant regulatory guidance is time consuming and expensive.

However, it is becoming increasingly difficult for operators to ignore the advantages that electronic systems can offer, not simply as e-readers for charts and documents, but as essential tools for improving organizational efficiency and safety. When used to their full potential, these systems can enable operators to streamline and automate any number of difficult systems including billing, record keeping, and document distribution, in addition to improving situational awareness in the cockpit. Thus, much of the content provided below is based on current and in use technology common in modern airliners, which is being gradually adopted by helicopter operators as well.

AIRCRAFT GENERAL INFORMATION SYSTEMS

Aircraft operations and maintenance involve dealing with large quantities of information. This information must be stored in some manner, and accessed as efficiently as possible. Also, because the information changes from time to time, there must be an efficient way to update the information stored in each system.

Aircraft information systems have evolved over the years. Initially, paper was the storage medium used. Manuals, drawings, charts, and other publications were available only in printed form. Information was retrieved by physically locating and by reading the pages containing the information needed. Paper resources have the advantage of being self contained. No special equipment is needed to access the information. When revisions are required, new pages are printed and distributed, and the old pages are removed and discarded.

Because paper manuals are bulky and heavy, other methods for storing large quantities of information were developed. Microfilms and microfiches were methods that saved space and weight. Microfiche involved using tiny photographic images of each page contained in manuals, including drawings, charts, etc. A disadvantage of microfilms and microfiches was that special equipment was needed to magnify the images so that they could be viewed. Microfilm and microfiche was useless if the viewing equipment malfunctioned or if it was not available when the information was needed. Updates to microfilm and microfiche came in the form of new rolls of microfilm, or new microfiche sheets. After the new rolls or sheets were received, the old media was removed and discarded.

Digital computers represent a major advance in information technology. Modern computers can store and process large amounts of information. They are compact and very lightweight compared to other storage media, such as printed books. Computers store this information in a type of memory known as Read Only Memory (ROM).

ROM can be internal to the computer, as in a hard drive or can be in a portable format such as in compact disks, flash drives, and other forms. The information contained in the digital memory can be accessed through display screens. It can be transferred in and out of the computer using wired data buses or wirelessly, and it can be printed. When updates are needed to the information contained in ROM, the memory can be electronically erased and written over. Thus there is no need to physically remove and replace paper or film.

Aircraft information systems can be used to store and retrieve many kinds of information for various users. Examples are flight deck information systems, maintenance information systems, and passenger cabin information systems.

Memory systems used on modern aircraft are solid state drives that have completely replaced the older hard disk drive systems. Applications for these functions are hosted on three sub-networks or "domains" of a network server system known as avionics domain, flight operation domain, and communication and cabin domain.

THE AIRBUS SYSTEM

The architecture selected by Airbus is based on a system of networked real time servers and routers combined with the centralized acquisition of secure digital communications. Although open to the world via digital radio links, the entire onboard system is designed to be highly secure both from the point of view of computer security and operational availability.

The Airbus system collects, centralizes, and compiles all the data related to the flight and provides external communication, data calculation and storage capacities. This modular central system also hosts applications unique to Airbus aircraft and its users that deal with the actual operation of the aircraft from services to passengers to on board documentation, navigation, performance calculations, flight logs, and more. The information system is made up of four components that operate in an integrated way.

Network Server System

The network server is the backbone of the system. One of its subsections is highly secured and strictly devoted to avionics. Another subsection, containing information and documents related to flight operations is connected to the outside world.

Secure Communication Interface

The secure communication interface is a link between the avionics and open world. As a basic component for the whole network security, it guarantees the security of information exchanged between the in-flight entertainment system and the avionics systems, as well as the security of the ground-to-air and air-to-ground communications.

Central Data Acquisition Module

The Central Data Acquisition Module is a maintenance monitoring system capable of recording and analyzing up to a million parameters. It can generate over a hundred different maintenance reports concerning the condition of the aircraft and possible technical failures. Operators can program and configure the module based on their needs, but also decode and display reports generated on board using ground programming and reading tools.

Data Loading and Configuration Systems

Airbus aircraft are also equipped with a unique data loading and configuration system. This is an application software for downloading and managing the configuration of various onboard computer software.

FLIGHT DECK INFORMATION SYSTEM

An example of a flight deck information system is the Electronic Flight Bag (EFB), an optional system used by many aircraft. (***Figure 19-1***)

The flight crew uses the EFB to access information that would traditionally have been found in various printed publications and carried aboard in a flight bag. Such publications include navigational information such as sectional charts and approach plates. The system also provides advanced capabilities beyond those available in printed publications.

On an aircraft equipped with the EFB, the system displays information on two display units that are installed on the flight deck. One display is for the pilot and one is for the copilot or navigator. These two displays are typically touchscreen devices and operate independently of each other. The display units are connected to two electronics units located on the Aircraft Information Management System (AIMS) rack in the main equipment center. The AIMS is an integrated system for processing information from many sources in the aircraft.

In this system, the electronic units send information stored in databases to the display units through fiber optic cables. The two displays are then connected to each other through wired connections. Databases accessible through the EFB include aeronautical charts, airport maps and charts with real time position monitoring, manuals, minimum equipment lists, and logbooks.

The system can also display video. The EFB system receives data from the AIMS and from Multi-Mode Receivers (MMRs). MMRs are GPS receivers designed for use in instrument landings. The MMRs provide aircraft position information that is extremely precise, allowing the EFB to pinpoint the aircraft position on an airport map.

The system also receives video from a camera interface. The EFB databases are normally updated through the aircraft data loader, but can also be updated wirelessly. For wireless updates, the system uses a terminal wireless

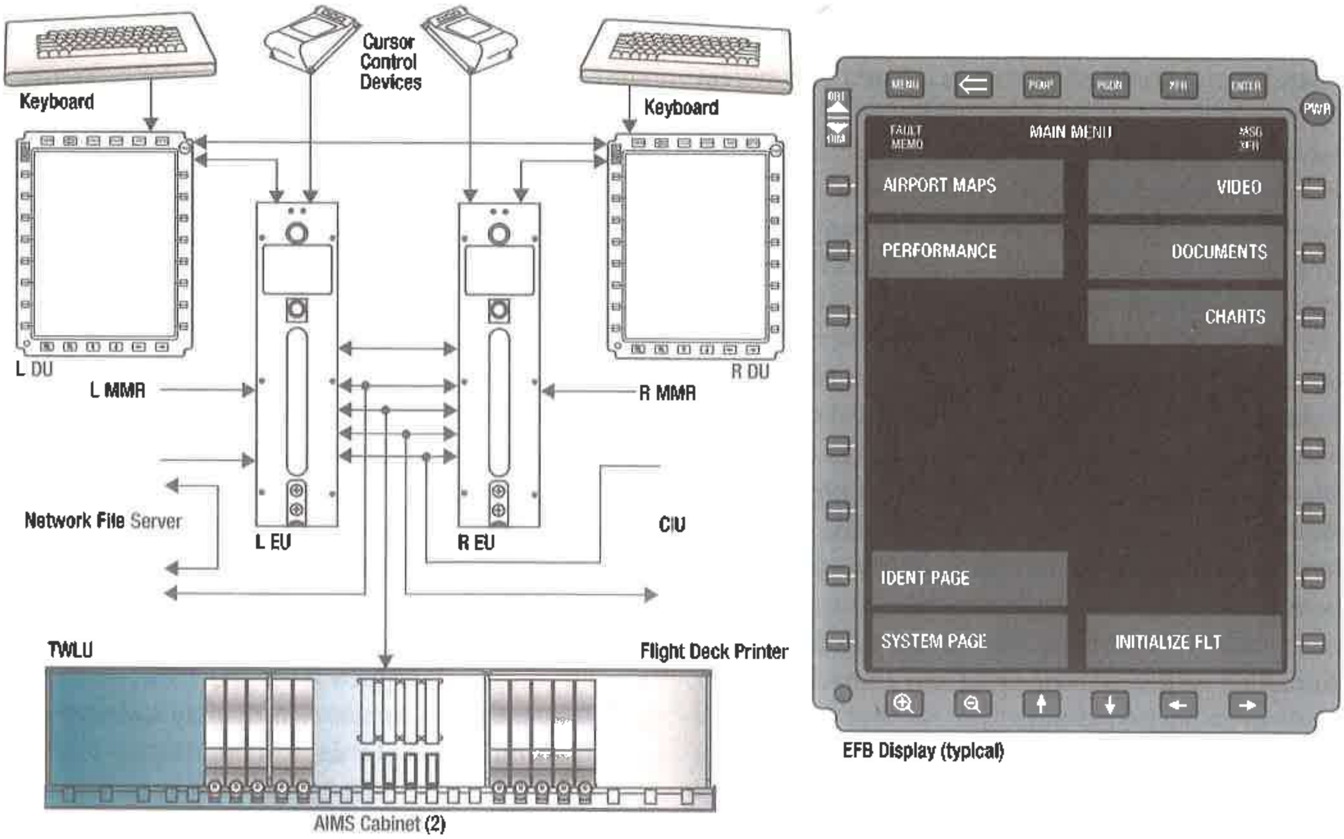


Figure 19-1. Electronic flight bag.

Local Area Network (LAN) unit containing a radio transmitter and receiver that creates a between the airplane and a ground based network, thus allowing the databases to be updated wirelessly.

MAINTENANCE INFORMATION SYSTEM

This EFB for the flight crew is one example of an aircraft information system. However, the same kind of technology is also used by maintenance personnel who need to access different kinds of information while the aircraft is on the ground.

Maintenance information systems work along similar lines as a flight deck information system, but the information being stored is different. Instead of maps and charts, maintenance personnel use maintenance manuals, illustrated parts catalogs, wiring diagram manuals, service bulletins, and other technical data. Maintenance information systems electronically provide technicians with access to these publications.

An advantage of using this electronic format, in addition to the space and weight savings it provides, is the ability to quickly locate the desired information. Instead of leafing through large paper manuals, a technician using a maintenance information system uses hyperlinks which allow an easy navigation within the system.

A typical method of achieving this is to use a menu containing links to each of the ATA 100 chapters within the manual. Within each chapter, the table of contents contains links that will quickly access a particular page and so allowing the technician to locate the desired page with a few "clicks."

Aircraft manufacturers, which previously published their manuals only on paper or microfilm/microfiche, now offer their manuals in electronic format. Laptops are very well suited for storing and retrieving maintenance information in the aviation maintenance environment. Laptops can be taken practically anywhere on the aircraft that the technician might need to go while performing maintenance, and are widely used to access maintenance information. (*Figure 19-2*)

ELECTRONIC LOGBOOKS

The electronic logbook replaces paper logbooks with computer based logs that can be easily stored and shared, and connecting flight data with ground based technicians

Figure 19-2. Laptop computers.

and equipment. The application feeds relevant data into a central repository where it is combined with maintenance and engineering information. This allows operators to better understand and diagnose issues within the context of multiple aircraft systems. The e-logbook maintenance application has the same function as the paper logbook, being used for:

- Defect reporting.
- Maintenance action reporting.
- Aircraft release after maintenance.

Airbus Helicopter's Fleet Keeper is an example of an e-logbook system. This system helps operators anticipate their maintenance activities and efficiently plan their operations. Moreover, pilots, airworthiness managers and technicians can easily share data among themselves in real time, removing the need for duplicating hand written reports. All flight and technical information recorded before flight, during flight and after flight will be uploaded automatically to a "cloud storage" from where it can be downloaded to a ground based computerized system.

PASSENGER CABIN INFORMATION SYSTEM

Flight attendants also need to access information to do their jobs. In paper form, a typical flight attendant manual weighs about 2.5 kg containing vital information such as checklists, procedures, security information, and information about safety devices and systems on the aircraft. The use of the digital electronics allows flight attendants to carry a small device, such as an iPad or tablet that contains the manuals they are required to have on hand. The device weighs much less than a paper manual, saving the operator money, as even small reductions in weight affect fuel economy.

CABIN WIRELESS CONNECTIVITY

In addition, cabin information systems on corporate helicopters include the provision for passenger's internet connectivity. An internal wireless data link system provides the resources necessary for wireless connections in the cabin and in the cockpit. (***Figure 19-3***)

Leaky Line Antennas

Provision of a wireless signal throughout the cabin is achieved by calibrated slots along the length of an antenna cable. The leaky line antenna is normally installed in the ceiling of the passenger cabin, from front to rear. (***Figure 19-4***)

Wireless Media Streamer

A wireless media streamer is designed for aircraft cabins using the aircraft's wireless LAN unit to create a wireless network server.

Figure 19-3. An internal wireless data link system for wireless connections in the cabin and cockpit.

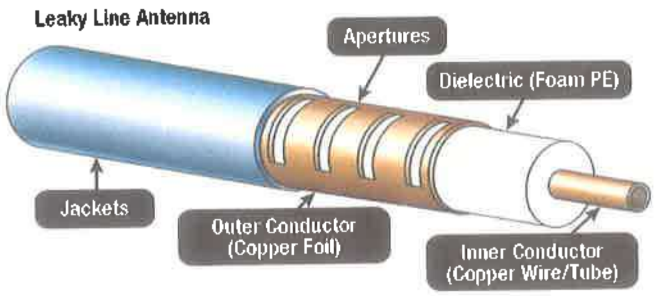


Figure 19-4. A leaky line antenna configuration used for cabin connectivity.

Question: 19-1
In which subsection of a computer is data such as maintenance manuals stored?

Question: 19-2
Which classification of information on an aircraft is handled with the highest level of security?

Question: 19-3
Name the 4 categories of information for the maintenance technician which is typically stored within and aircraft information system.

Question: 19-4
Name 4 types of information typically found in an Electronic Flight Bag.

Question: 19-5
Within an electronic maintenance manual, by what standard of organization can Maintenance instructions be located?

Question: 19-6
Where on board an aircraft will you find a record of reported defects?

ANSWERS

Answer: 19-1
ROM – Read-Only-Memory section.

Answer: 19-2
Avionics section.

Answer: 19-3
Manufacturer's maintenance manuals, parts catalog, wiring diagrams, log books.

Answer: 19-4
Navigational charts, aircraft operation manuals, minimum equipment lists, logbooks.

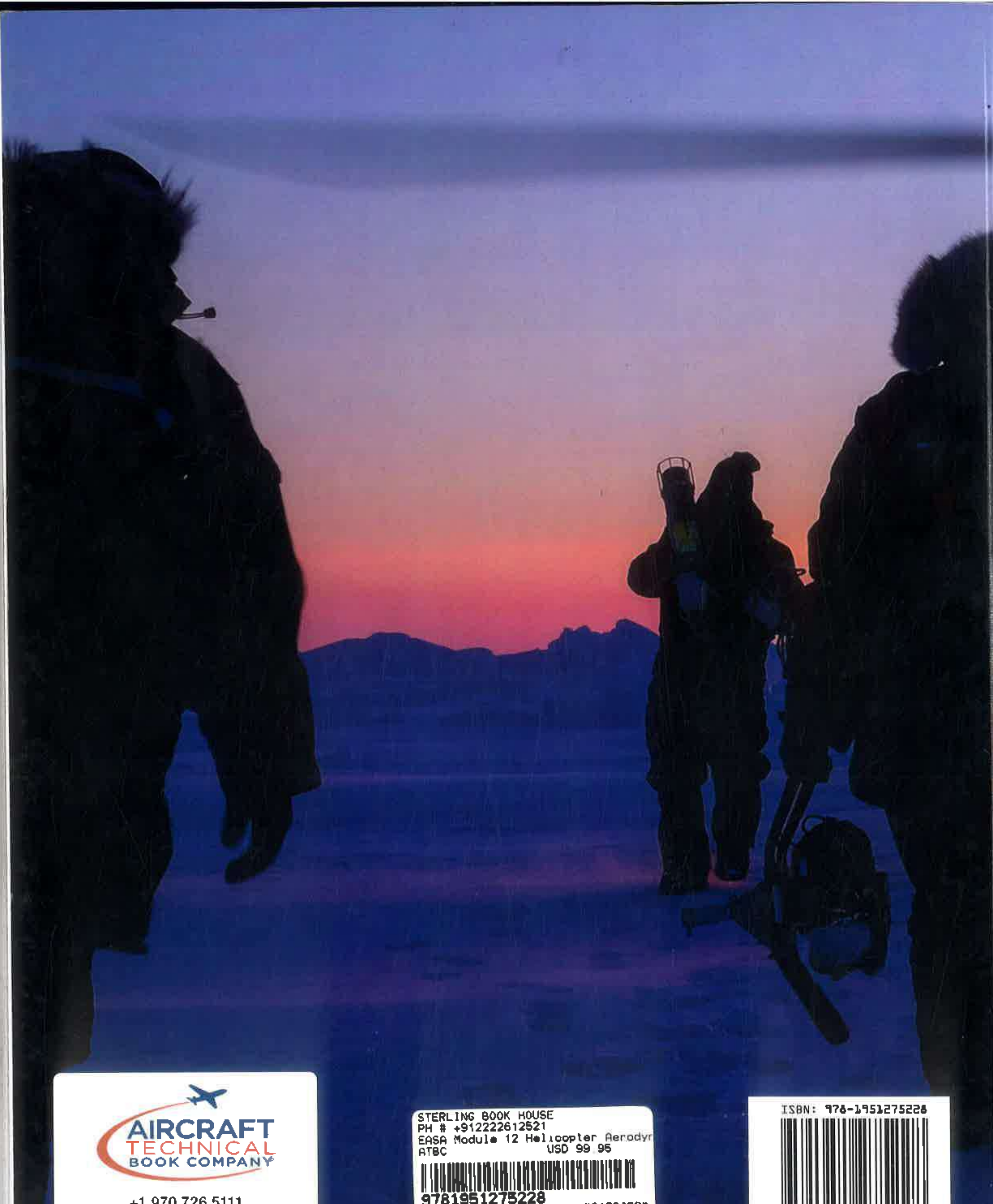
Answer: 19-5
By its ATA 100 classification.

Answer: 19-6
The electronic logbook.

AC	/	Alternating Current
AC	/	Air Conditioning
ACARS	/	Aircraft Communication Addressing and Reporting System
ACM	/	Air Cycle Machine
ADC	/	Air Data Computer
ADF	/	Automatic Direction Finder
AFC	/	Automatic Flight Control
AFDX	/	Avionics Full-Duplex Switched Ethernet
AFGS	/	Automatic Flight Guidance System
AHRS	/	Attitude Heading Reference System
AIMS	/	Aircraft Information Management System
AMU	/	Audio Management Unit
APU	/	Auxiliary Power Unit
ASCB	/	Avionics Standard Communication Bus
ATC	/	Air Traffic Control
ATCRBS	/	Air Traffic Control Radar Beacon System
ATT	/	Autopilot
AVCS	/	Active Vibration Control System
BITE	/	Built In Test Equipment
BPCU	/	Bus Power Control Unit
BTB	/	Bus Tie Breaker
CDI	/	Course Deviation Indicator
CG	/	Center of Gravity
CHT	/	Cylinder Head Temperature
CMC	/	Central Maintenance Computer
CP	/	Center of Pressure
CPIOM	/	Core Processor Input/Output Module
CS	/	Certification Specification
CSD	/	Constant Speed Drive
CT	/	Current Transformer
DC	/	Direct Current
DG	/	Directional Gyro
DME	/	Distance Measuring Equipment
DTI	/	Damage Tolerance Inspection
EADI	/	Electronic Attitude Director Indicator
EAI	/	Engine Anti-Ice
ECAM	/	Electronic Centralized Aircraft Monitoring
EFB	/	Electronic Flight Bag
EFIS	/	Electronic Flight Instrument System
EGT	/	Exhaust Gas Temperature
EICAS	/	Engine Indicating and Crew Alert System
ELS	/	Electronic Library System
ELT	/	Emergency Locator Transmitter
ETL	/	Effective Translational Lift
FCC	/	Flight Control Computer
FCS	/	Fatigue Critical Structure

FDP	/	Flight Data Processor
FMC	/	Flight Management Computer
FMS	/	Flight Management System
FPM	/	Feet Per Minute
GB	/	Generator Breaker
GCB	/	Generator Control Breaker
GCU	/	Generator Control Unit
GPS	/	Global Positioning System
HCP	/	HeliSAS Control Panel
HF	/	High Frequency
HMDG	/	Hydraulic Motor Driven Generator
HOT	/	Hold Over Time
HRD	/	High Rate of Discharge
HSI	/	Horizontal Situation Indicator
HUMS	/	Health and Usage Monitoring Systems
IDG	/	Integrated Drive Generator
IFR	/	Instrument Flight Rules
IGB	/	Intermediate Gear Box
IGE	/	In Ground Effect
ILS	/	Instrument Landing System
IMA	/	Integrated Module Avionics
IMU	/	Inertial Measurement Unit
INS	/	Inertial Navigation System
IRS	/	Inertial Reference System
IVSI	/	Instantaneous Vertical Speed Indicator
LAN	/	Local Area Network
LASER	/	Light Amplification by Stimulated Emission of Radiation
LCD	/	Liquid Crystal Display
LF	/	Low Frequency
LRU	/	Line Replaceable Units
MCDU	/	Multipurpose Control Display Unit
MDDU	/	Multipurpose Disk Drive Unit
MEL	/	Minimum Equipment List
MEMS	/	Micro Electro Mechanical Systems
MF	/	Medium Frequency
MGB	/	Main Gear Box
MLS	/	Microwave Landing System
MMR	/	Multi Mode Receiver
MOPSC	/	Maximum Operational Passenger Seating
MSL	/	Mean Sea Level
NAA	/	National Aviation Administration
NDB	/	Non-Directional Beacon
NOTAR	/	No Tail Rotor
OBS	/	Omni Bearing Selector
OGE	/	Out Of Ground Effect
PSI	/	Pounds per Square Inch

PTU	/	Power Transfer Unit
RA	/	Radio Altimeter
RDF	/	Radio Direction Finder
RIPS	/	Rotor Ice Protection System
RLG	/	Ring Laser Gyro
ROM	/	Read Only Memory
RPM	/	Revolutions Per Minute
RTD	/	Resistance Temperature Detector
SAS	/	Stability Augmentation System
SATCOM	/	Satellite Communications
SCR	/	Silicon Controlled Rectifier
SNS	/	Satellite Navigation System
SRM	/	Structural Repair Manual
SSB	/	Split System Breaker
TAF	/	Total Aerodynamic Force
TAT	/	Total Air Temperature
TCAS	/	Traffic Alert And Collision Avoidance Systems
TGB	/	Tail Gear Box
TIT	/	Turbine Inlet Temperature
TR	/	Transformer Rectifier
TSO	/	Technical Standard Order
UHF	/	Ultra High Frequency
VHF	/	Very High Frequency
VLR	/	Variable Load Resistor
VNE	/	Never Exceed Speed
VOR	/	VHF Omnidirectional Range
VOT	/	VOR Test Facility
VRLA	/	Valve Regulated Lead-Acid Batteries
VSI	/	Vertical Speed Indicator
VVI	/	Vertical Velocity Indicator
WOW	/	Weight On Wheels



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