Module 06

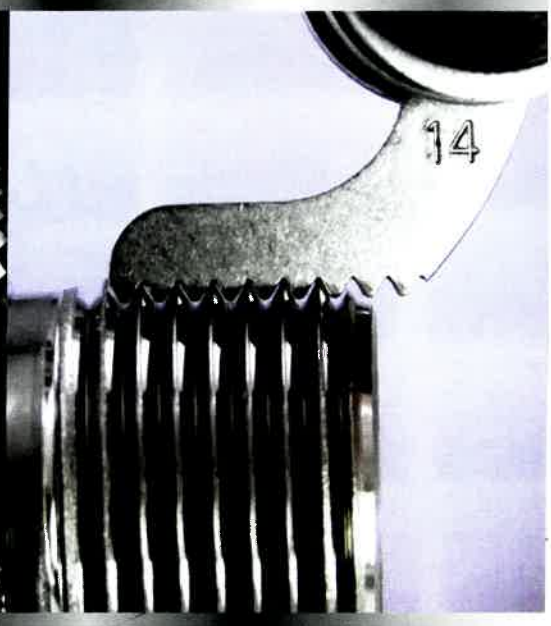
FOR B1 CERTIFICATION

MATERIALS AND HARDWARE

Aviation Maintenance Technician Certification Series







- Ferrous
- Non-Ferrous
- Composite and Non-Metallic
- Corrosion
- Fasteners
- Pipes and Unions
- Springs
- Bearings
- Transmissions
- Control Cables
- Electrical Cables and Connectors

COMPLIANT WITH EASA

European Aviation Safety Agency

EASA PART 66/147

Version 003.1



MODULE 06

FOR B1 CERTIFICATION

MATERIALS AND HARDWARE

Aviation Maintenance Technician Certification Series

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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	2013 01	Module Creation and Release
002	2014 07	Format Update/Minor Corrections
003	2018 07	Content alignment to Part-66 Appendix 1 Modified sub-module sequence in accordance to Appendix A. Added content in the following areas: 6.3 – Thixotropic agents; Wood preservation methods; Types of fabric defects. 6.4 – Stress corrosion; Exfoliated corrosion detection; Galvanic corrosion growth; Fretting corrosion. 6.5 – Screw threads; Dowels. 6.6 – Rigid fluid line sizes; Swaged fittings. 6.7 – Laws obeyed by springs. 6.8 – Bearing material and construction. 6.11 – Connector types, Identification codes, and ratings.
003.1	2023 04	Inclusion of Measurement Standards for clarification, page iv. Minor appearance and format updates.

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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MEASUREMENT STANDARDS

SI Units

Measurements in this book are presented with International System of Units (SI) standards in all cases except when otherwise specified by ICAO (for example, altitude expressed in feet or performance numbers as specified by a manufacturer). The chart below can be used should your studies call for conversions into imperial numbers.

Number Groups

This book uses the International Civil Aviation Organization (ICAO) standard of writing numbers. This method separates groups of 3 digits with a space, versus the European method by periods and the American method by commas. For example, the number one million is expressed as:

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

Prefixes

The prefixes in the table below form names of the decimal equivalents in 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
0.0 1 = 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

COMMON CONVERSIONS

IMPERIAL SYSTEM	TO	SI (METRIC)
Distance		
1 Inch	is equal to	2.54 Centimeters
1 Foot	is equal to	0.304 Meters
1 (Statute) Mile	is equal to	1.609 Kilometers
Weight		
1 Pound	is equal to	0.454 Kilograms
Volume		
1 Quart	is equal to	0.946 Liters
1 Gallon	is equal to	3.785 Liters
Temperature		
°0 Fahrenheit	is equal to	(-)17.778 Celsius (°C)
°0 Fahrenheit	is equal to	255.37 Kelvin (K)
Area		
1 Square Inch	is equal to	6.451 Square Centimeters
1 Square Foot	is equal to	0.093 Square Meters
1 Square Mile	is equal to	2.59 Square Kilometers
Velocity		
1 Foot Per Second	is equal to	0.304 Meters Per Second
1 Mile Per Hour	is equal to	1.609 Kilometers Per Hour
1 Knot	is equal to	1.852 Kilometers Per Hour

SI (METRIC)	TO	IMPERIAL SYSTEM
Distance		
1 Centimeter	is equal to	0.394 Inches
1 Meter	is equal to	3.28 Feet
1 Kilometer	is equal to	0.621 Miles
Weight		
1 Kilogram	is equal to	2.204 Pounds
Volume		
1 Liter	is equal to	1.057 Quarts
1 Liter	is equal to	0.264 Gallons
Temperature		
°0 Celsius (°C)	is equal to	33.8° Fahrenheit
°0 Kelvin (K)	is equal to	(-)437.87 Fahrenheit
Area		
1 Square Centimeter	is equal to	0.155 Square Inches
1 Square Meter	is equal to	10.764 Square Feet
1 Square Kilometer	is equal to	0.386 Square Miles
Velocity		
1 Meter Per Second	is equal to	3.281 Feet Per Second
1 Kilometer Per Hour	is equal to	0.621 Miles Per Hour
1 Kilometer Per Hour	is equal to	0.540 Knots

Pressure		
pounds per square inch (psi)	kiloPascals (kPa)	6.897
pounds per square inch (psi)	Pascals (Pa)	6.894

EASA LICENSE CATEGORY CHART

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

GENERAL KNOWLEDGE REQUIREMENTS

MODULE 06 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.

Sub-Module 01 - Aircraft Materials — Ferrous

(a) Characteristics, properties and identification of common alloy steels used in aircraft; **2**

(b) Testing of ferrous materials for hardness, tensile strength, fatigue strength and impact resistance. **1**

Sub-Module 02 - Aircraft Materials — Non-Ferrous

(a) Characteristics, properties and identification of common non-ferrous materials used in aircraft; Heat treatment and application of non-ferrous materials; **2**

(b) Testing of non-ferrous material for hardness, tensile strength, fatigue strength and impact resistance. **1**

Sub-Module 03 - Aircraft Materials — Composite and Non-Metallic

6.3.1 Composite and non-metallic other than wood and fabric

(a) Characteristics, properties and identification of common composite and non-metallic materials, other than wood, used in aircraft; Sealant and bonding agents; **2**

(b) The detection of defects/deterioration in composite and non-metallic material; Repair of composite and non-metallic material. **2**

6.3.2 Wooden Structure

Construction methods of wooden airframe structures; **2**
Characteristics, properties and types of wood and glue used in aeroplanes;
Preservation and maintenance of wooden structure;
Types of defects in wood material and wooden structures;
The detection of defects in wooden structure;
Repair of wooden structure.

6.3.3 Fabric Covering

Characteristics, properties and types of fabrics used in aeroplanes; **2**
Inspections methods for fabric;
Types of defects in fabric;
Repair of fabric covering.

Sub-Module 04 - Corrosion

(a) Chemical fundamentals; **1**
Formation by, galvanic action process, microbiological, stress;

(b) Types of corrosion and their identification; **3**
Causes of corrosion;
Material types, susceptibility to corrosion.

Sub-Module 05 - Fasteners

6.5.1 Screw Threads

Screw nomenclature; 2
Thread forms, dimensions and tolerances for standard threads used in aircraft;
Measuring screw threads.

6.5.2 Bolts, Studs and Screws

Bolt types: specification, identification and marking of aircraft bolts, international standards; 2
Nuts: self locking, anchor, standard types;
Machine screws: aircraft specifications;
Studs: types and uses, insertion and removal;
Self tapping screws, dowels.

6.5.3 Locking Devices

Tab and spring washers, locking plates, split pins, pal-nuts, wire locking, quick release fasteners, keys, circlips, cotter pins. 2

6.5.4 Aircraft Rivets

Types of solid and blind rivets: specifications and identification, heat treatment. 2

Sub-Module 06 - Pipes and Unions

(a) Identification of, and types of rigid and flexible pipes and their connectors used in aircraft; 2

(b) Standard unions for aircraft hydraulic, fuel, oil, pneumatic and air system pipes. 2

Sub-Module 07 - Springs

Types of springs, materials, characteristics and applications. 2

Sub-Module 08 - Bearings

Purpose of bearings, loads, material, construction; 2
Types of bearings and their application.

Sub-Module 09 - Transmissions

Gear types and their application; 2
Gear ratios, reduction and multiplication gear systems, driven and driving gears, idler gears, mesh patterns;
Belts and pulleys, chains and sprockets.

Sub-Module 10 - Control Cables

Types of cables; 2
End fittings, turnbuckles and compensation devices;
Pulleys and cable system components;
Bowden cables;
Aircraft flexible control systems.

Sub-Module 11 - Electrical Cables and Connectors

2

Cable types, construction and characteristics;
High tension and co-axial cables;
Crimping;
Connector types, pins, plugs, sockets, insulators, current and voltage rating, coupling, identification codes.

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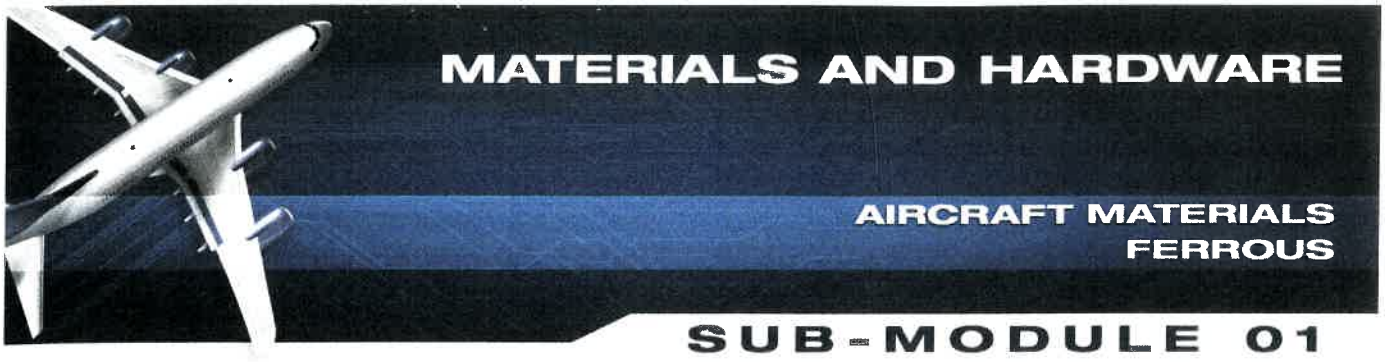
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PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY → B1

Sub-Module 01

AIRCRAFT MATERIALS — FERROUS

Knowledge Requirements

6.1 - Aircraft Materials — Ferrous

(a)	Characteristics, properties and identification of common alloy steels used in aircraft; Heat treatment and application of alloy steels.	2
(b)	Testing of ferrous materials for hardness, tensile strength, fatigue strength and impact resistance.	1

6.1(A) - AIRCRAFT METALS

Knowledge and understanding of the uses, strengths, limitations, and other characteristics of structural metals is vital to properly construct and maintain any equipment, especially airframes. In aircraft maintenance and repair, even a slight deviation from design specification, or the substitution of inferior materials, may result in the loss of both lives and equipment. The use of unsuitable materials can readily erase the finest craftsmanship. The selection of the correct material for a specific repair job demands familiarity with the most common physical properties of various metals.

PROPERTIES OF METALS

Of primary concern in aircraft maintenance are such general properties of metals and their alloys as hardness, malleability, ductility, elasticity, toughness, density, brittleness, fusibility, conductivity contraction and expansion, and so forth. These terms are explained to further the discussion of structural metals.

HARDNESS

Hardness refers to the ability of a material to resist abrasion, penetration, cutting action, or permanent distortion. Hardness may be increased by cold working the metal and, in the case of steel and certain aluminum alloys, by heat treatment. Structural parts are often formed from metals in their soft state and are then heat treated to harden them so that the finished shape will be retained. Hardness and strength are closely associated properties of metals.

STRENGTH

One of the most important properties of a material is strength. Strength is the ability of a material to resist deformation. Strength is also the ability of a material to resist stress without breaking. The type of load or stress on the material affects the strength it exhibits.

DENSITY

Density is the weight of a unit volume of a material. In aircraft work, the specified weight of a material per cubic inch is preferred since this figure can be used in determining the weight of a part before actual manufacture. Density is an important consideration when choosing a material to be used in the design of a part in order to maintain the proper weight and balance of the aircraft.

MALLEABILITY

A metal which can be hammered, rolled, or pressed into various shapes without cracking, breaking, or leaving some other detrimental effect, is said to be malleable. This property is necessary in sheet metal that is worked into curved shapes, such as cowlings, fairings, or wingtips. Copper is an example of a malleable metal.

DUCTILITY

Ductility is the property of a metal which permits it to be permanently drawn, bent, or twisted into various shapes without breaking. This property is essential for metals used in making wire and tubing. Ductile metals are greatly preferred for aircraft use because of their ease of forming and resistance to failure under shock loads. For this reason, aluminum alloys are used for cowl rings, fuselage and wing skin, and formed or extruded parts, such as ribs, spars, and bulkheads. Chrome molybdenum steel is also easily formed into desired shapes. Ductility is similar to malleability.

ELASTICITY

Elasticity is that property that enables a metal to return to its original size and shape when the force which causes the change of shape is removed. This property is extremely valuable because it would be highly undesirable to have a part permanently distorted after an applied load was removed.

Each metal has a point known as the elastic limit, beyond which it cannot be loaded without causing permanent distortion. In aircraft construction, members and parts are so designed that the maximum loads to which they are subjected will not stress them beyond their elastic limits. This desirable property is present in spring steel.

TOUGHNESS

A material which possesses toughness will withstand tearing or shearing and may be stretched or otherwise deformed without breaking. Toughness is a desirable property in aircraft metals.

BRITTLENESS

Brittleness is the property of a metal which allows little bending or deformation without shattering. A brittle metal is apt to break or crack without change of shape.

Because structural metals are often subjected to shock loads, brittleness is not a very desirable property. Cast iron, cast aluminum, and very hard steel are examples of brittle metals.

FUSIBILITY

Fusibility is the ability of a metal to become liquid by the application of heat. Metals are fused in welding. Steels fuse around 1 425°C (2 600°F) and aluminum alloys at approximately 600°C (1 100°F).

CONDUCTIVITY

Conductivity is the property which enables a metal to carry heat or electricity. The heat conductivity of a metal is especially important in welding because it governs the amount of heat that will be required for proper fusion. Conductivity of the metal, to a certain extent, determines the type of jig to be used to control expansion and contraction. In aircraft, electrical conductivity must also be considered in conjunction with bonding to eliminate radio interference.

THERMAL EXPANSION

Thermal expansion refers to contraction and expansion that are reactions produced in metals as the result of heating or cooling. Heat applied to a metal will cause it to expand or become larger. Cooling and heating affect the design of welding jigs, castings, and tolerances necessary for hot rolled material.

TYPES OF FERROUS METALS

Many different metals are required in the repair of aircraft. This is a result of the varying needs with respect to strength, weight, durability, and resistance to deterioration of specific structures or parts. In addition, the particular shape or form of the material plays an important role. In selecting materials for aircraft repair, these factors plus many others are considered in relation to the mechanical and physical properties. Among the common materials used are ferrous metals. The term "ferrous" applies to the group of metals having iron as their principal constituent.

IRON

If carbon is added to iron, in percentages ranging up to approximately 1 percent, the product is vastly superior to iron alone and is classified as carbon steel. Carbon steel forms the base of those alloy steels produced by combining carbon steel with other elements known to improve the properties of steel. A base metal (such as iron) to which small quantities of other metals have been added is called an alloy. The addition of other metals changes or improves the chemical or physical properties of the base metal for a particular use.

STEEL AND STEEL ALLOYS

To facilitate the discussion of steels, some familiarity with their nomenclature is desirable. A numerical index, sponsored by the Society of Automotive Engineers (SAE) and the American Iron and Steel Institute (AISI), is used to identify the chemical compositions of the structural steels. In this system, a four numeral series is used to designate the plain carbon and alloy steels; five numerals are used to designate certain types of alloy steels. The first two digits indicate the type of steel, the second digit also generally (but not always) gives the approximate amount of the major alloying element, and the last two (or three) digits are intended to indicate the approximate middle of the carbon range. However, a deviation from the rule of indicating the carbon range is sometimes necessary.

Small quantities of certain elements are present in alloy steels that are not specified as required. These elements are considered as incidental and may be present to the maximum amounts as follows: copper, 0.35 percent; nickel, 0.25 percent; chromium, 0.20 percent; molybdenum, 0.06 percent.

The list of standard steels is altered from time to time to accommodate steels of proven merit and to provide for changes in the metallurgical and engineering requirements of industry. (***Figure 1-1***)

Metal stock is manufactured in several forms and shapes, including sheets, bars, rods, tubing, extrusions, forgings, and castings. Sheet metal is made in a number of sizes and thicknesses. Specifications designate thicknesses in thousandths of an inch. Bars and rods are supplied in a variety of shapes, such as round, square, rectangular, hexagonal, and octagonal. Tubing can be obtained in round, oval, rectangular, or streamlined shapes. The size of tubing is generally specified by outside diameter and wall thickness.

The sheet metal is usually formed cold in such machines as presses, bending brakes, drawbenches, or rolls. Forgings are shaped or formed by pressing or hammering heated

Series Designation	Types
10xx	Non-sulfurized carbon steels
11xx	Resulfurized carbon steels (free machining)
12xx	Rephosphorized and resulfurized carbon steels (free machining)
13xx	Manganese 1.75%
*23xx	Nickel 3.50%
*25xx	Nickel 5.00%
31xx	Nickel 1.25%, chromium 0.65%
33xx	Nickel 3.50%, chromium 1.55%
40xx	Molybdenum 0.20 or 0.25%
41xx	Chromium 0.50% or 0.95%, molybdenum 0.12 or 0.20%
43xx	Nickel 1.80%, chromium 0.5 or 0.80%, molybdenum 0.25%
44xx	Molybdenum 0.40%
45xx	Molybdenum 0.52%
46xx	Nickel 1.80%, molybdenum 0.25%
47xx	Nickel 1.05% chromium 0.45%, molybdenum 0.20 or 0.35%
48xx	Nickel 3.50%, molybdenum 0.25%
50xx	Chromium 0.25, or 0.40 or 0.50%
50xxx	Carbon 1.00%, chromium 0.50%
51xx	Chromium 0.80, 0.90, 0.95 or 1.00%
51xxx	Carbon 1.00%, chromium 1.05%
52xxx	Carbon 1.00%, chromium 1.45%
61xx	Chromium 0.60, 0.80, 0.95%, vanadium 0.12%, 0.10% min., or 0.15% min.
81xx	Nickel 0.30%, chromium 0.40%, molybdenum 0.12%
86xx	Nickel 0.55%, chromium 0.50%, molybdenum 0.20%
87xx	Nickel 0.55%, chromium 0.05%, molybdenum 0.25%
88xx	Nickel 0.55%, chromium 0.05%, molybdenum 0.35%
92xx	Manganese 0.85%, silicon 2.00%, chromium 0 or 0.35%
93xx	Nickel 3.25%, chromium 1.20%, molybdenum 0.12%
94xx	Nickel 0.45%, chromium 0.40%, molybdenum 0.12%
98xx	Nickel 1.00%, chromium 0.80%, molybdenum 0.25%

*Not included in the current list of standard steels

Figure 1-1. SAE numerical index.

metal in dies. Castings are produced by pouring molten metal into molds. The casting is finished by machining.

Spark testing is a common means of identifying various ferrous metals. In this test the piece of iron or steel is held against a revolving grinding stone and the metal is identified by the sparks thrown off. Each ferrous metal has its own peculiar spark characteristics. The spark streams vary from a few tiny shafts to a shower of sparks several feet in length. (Few nonferrous metals give off sparks when touched to a grinding stone. Therefore, these metals cannot be successfully identified by the spark test.)

Identification by spark testing is often inexact unless performed by an experienced person, or the test pieces differ greatly in their carbon content and alloying constituents. Wrought iron produces long shafts that are straw colored as they leave the stone and white at the end. Cast iron sparks are red as they leave the stone and turn to a straw color. Low carbon steels give off long, straight shafts having a few white sprigs. As the carbon content of the steel increases, the number of sprigs along each shaft increases and the stream becomes whiter in color. Nickel steel causes the spark stream to contain small white blocks of light within the main burst.

TYPES OF ALLOYED STEEL

CARBON STEEL

Steel containing carbon in percentages ranging from 0.10 to 0.30 percent is classed as low carbon steel. The equivalent SAE numbers range from 1010 to 1030.

Steels of this grade are used for making such items as safety wire, certain nuts, cable bushings, or threaded rod ends. This steel in sheet form is used for secondary structural parts and clamps, and in tubular form for moderately stressed structural parts. Steel containing carbon in percentages ranging from 0.30 to 0.50 percent is classed as medium carbon steel. This steel is especially adaptable for machining or forging, and where surface hardness is desirable. Certain rod ends and light forgings are made from SAE 1035 steel.

Steel containing carbon in percentages ranging from 0.50 to 1.05 percent is classed as high carbon steel. The addition of other elements in varying quantities adds to the hardness of this steel. In the fully heat treated condition it is very hard, will withstand high shear and wear, and will have little deformation. It has limited use in aircraft. SAE 1095 in sheet form is used for making flat springs and in wire form for making coil springs.

NICKEL STEEL

The various nickel steels are produced by combining nickel with carbon steel. Steels containing from 3 to 3.75 percent nickel are commonly used. Nickel increases the hardness, tensile strength, and elastic limit of steel without appreciably decreasing the ductility. It also intensifies the hardening effect of heat treatment. SAE 2330 steel is used extensively for aircraft parts, such as bolts, terminals, keys, clevises, and pins.

CHROMIUM STEEL

Chromium steel is high in hardness, strength, and corrosion resistant properties, and is particularly adaptable for heat treated forgings which require greater toughness and strength than may be obtained in plain carbon steel. It can be used for such articles as the balls and rollers of anti-friction bearings.

STAINLESS STEEL

Chrome nickel or stainless steels are the corrosion resistant metals. The anti-corrosive degree of this steel is determined by the surface condition of the metal as well as by the composition, temperature, and concentration of the corrosive agent. The principal alloy of stainless steel is chromium. The corrosion resistant steel most often used in aircraft construction is known as 18-8 steel because of its content of 18 percent chromium and 8 percent nickel. One of the distinctive features of 18-8 steel is that its strength may be increased by cold working.

Stainless steel may be rolled, drawn, bent, or formed to any shape. Because these steels expand about 50 percent more than mild steel and conduct heat only about 40 percent as rapidly, they are more difficult to weld. Stainless steel can be used for almost any part of an aircraft. Some of its common applications are in the fabrication of exhaust collectors, stacks and manifolds, structural and machined parts, springs, castings, tie rods, control cables and firewalls.

CHROME-VANADIUM STEEL

The chrome-vanadium steels are made of approximately 18 percent vanadium and about 1 percent chromium. When heat treated, they have strength, toughness, and resistance to wear and fatigue. A special grade of this steel in sheet form can be cold formed into intricate shapes. It can be folded and flattened without signs of breaking or failure. SAE 6150 is used for making springs; chrome-vanadium with high carbon content, SAE 6195, is used for ball and roller bearings.

CHROMOLY STEEL

Molybdenum in small percentages is used in combination with chromium to form chrome-molybdenum steel, which has various uses in aircraft. Molybdenum is a strong alloying element. It raises the ultimate strength of steel without affecting ductility or workability. Molybdenum steels are tough and wear resistant, and

they harden throughout when heat treated. They are especially adaptable for welding and, for this reason, are used principally for welded structural parts and assemblies. This type steel has practically replaced carbon steel in the fabrication of fuselage tubing, engine mounts, landing gears, and other structural parts. For example, a heat-treated SAE X4130 tube is approximately four times as strong as an SAE 1025 tube of the same weight and size.

A series of chrome-molybdenum steel most used in aircraft construction is that series containing 0.25 to 0.55 percent carbon, 0.15 to 0.25 percent molybdenum, and 0.50 to 1.10 percent chromium. These steels, when suitably heat treated, are deep hardening, easily machined, readily welded by either gas or electric methods, and are especially adapted to high temperature service.

INCONEL

Inconel is a nickel-chromium-iron alloy closely resembling stainless steel (Corrosion Resistant Steel, CRES) in appearance. Aircraft exhaust systems use both alloys interchangeably. Because the two alloys look very much alike, a distinguishing test is often necessary. One method of identification is to use an electrochemical technique, as described in the following paragraph, to identify the Nickel (Ni) content of the alloy. Inconel has a nickel content greater than 50 percent, and the electrochemical test detects nickel. The tensile strength of Inconel is 100 000 psi annealed, and 125 000 psi when hard rolled. It is highly resistant to salt water and is able to withstand temperatures as high as 870°C (1 600°F). Inconel welds readily and has working qualities quite similar to those of corrosion resistant steels.

SUBSTITUTION OF AIRCRAFT METALS

In selecting substitute metals for the repair and maintenance of aircraft, it is very important to check the appropriate structural repair manual. Aircraft manufacturer's design structural members to meet a specific load requirement for a particular aircraft. The methods of repairing these members, apparently similar in construction, will thus vary with different aircraft.

Four requirements must be kept in mind when selecting substitute metals. The first and most important of these is maintaining the original strength of the structure. The other three are: (1) maintaining contour or aerodynamic smoothness, (2) maintaining original weight, if possible, or keeping added weight to a minimum, and (3) maintaining the original corrosion resistant properties of the metal.

METAL WORKING PROCESSES

There are three methods of metalworking: (1) hot working, (2) cold working, and (3) extruding. The method used will depend on the metal involved and the part required, although in some instances both hot and cold working methods may be used to make a single part.

HOT WORKING

Almost all steel is hot worked from the ingot into some form from which it is either hot or cold worked to the finished shape. When an ingot is stripped from its mold, its surface is solid, but the interior is still molten. The ingot is then placed in a soaking pit which retards loss of heat, and the molten interior gradually solidifies. After soaking, the temperature is equalized throughout the ingot, then it is reduced to intermediate size by rolling, making it more readily handled.

The rolled shape is called a bloom when its section dimensions are 6 inches by 6 inches or larger and approximately square. The section is called a billet when it is approximately square and less than 6 inches by 6 inches. Rectangular sections which have a width greater than twice their thickness are called slabs. The slab is the intermediate shape from which sheets are rolled.

Blooms, billets, or slabs are heated above the critical range and rolled into a variety of shapes of uniform cross section. Common rolled shapes are sheet, bar, channel, angle, and I-beam. As discussed later in this chapter, hot rolled material is frequently finished by cold rolling or drawing to obtain accurate finish dimensions and a bright, smooth surface.

Complicated sections which cannot be rolled, or sections of which only a small quantity is required, are usually forged. Forging of steel is a mechanical working at temperatures above the critical range to shape the metal as desired. Forging is done either by pressing or hammering the heated steel until the desired shape is obtained. Pressing is used when the parts to be forged are large and heavy; this process also replaces

hammering where high grade steel is required. Since a press is slow acting, its force is uniformly transmitted to the center of the section, thus affecting the interior grain structure as well as the exterior to give the best possible structure throughout.

Hammering can be used only on relatively small pieces. Since hammering transmits its force almost instantly, its effect is limited to a small depth. Thus, it is necessary to use a very heavy hammer or to subject the part to repeated blows to ensure complete working of the section. If the force applied is too weak to reach the center, the finished forged surface will be concave. If the center was properly worked, the surface will be convex or bulged. The advantage of hammering is that the operator has control over both the amount of pressure applied and the finishing temperature, and is able to produce small parts of the highest grade. This type of forging is usually referred to as smith forging. It is used extensively where only a small number of parts are needed. Considerable machining time and material are saved when a part is smith forged to approximately the finished shape.

Steel is often harder than necessary and too brittle for most practical uses when put under severe internal strain. To relieve such strain and reduce brittleness, it is tempered after being hardened. This consists of heating the steel in a furnace to a specified temperature and then cooling it in air, oil, water, or a special solution. Temper condition refers to the condition of metal or metal alloys with respect to hardness or toughness. Rolling, hammering, or bending these alloys, or heat treating and aging them, causes them to become tougher and harder. At times these alloys become too hard for forming and have to be reheat treated or annealed.

Metals are annealed to relieve internal stresses, soften the metal, make it more ductile, and refine the grain structure. Annealing consists of heating the metal to a prescribed temperature, holding it there for a specified length of time, and then cooling the metal back to room temperature. To produce maximum softness, the metal must be cooled very slowly. Some metals must be furnace cooled; others may be cooled in air. Normalizing applies to iron base metals only. Normalizing consists of heating the part to the proper temperature, holding it at that temperature until it is uniformly heated, and then cooling it in still air. Normalizing is used to relieve stresses in metals.

Strength, weight, and reliability are three factors which determine the requirements to be met by any material used in airframe construction and repair. Airframes must be strong and yet as light weight as possible. There are very definite limits to which increases in strength can be accompanied by increases in weight. An airframe so heavy that it could not support a few hundred pounds of additional weight would be of little use.

All metals, in addition to having a good strength/weight ratio, must be thoroughly reliable, thus minimizing the possibility of dangerous and unexpected failures. In addition to these general properties, the material selected for a definite application must possess specific qualities suitable for the purpose.

The material must possess the strength required by the dimensions, weight, and use. The five basic stresses which metals may be required to withstand are tension, compression, shear, bending, and torsion.

The tensile strength of a material is its resistance to a force which tends to pull it apart. Tensile strength is measured in pounds per square inch (psi) and is calculated by dividing the load in pounds required to pull the material apart by its cross-sectional area in square inches.

The compression strength of a material is its resistance to a crushing force which is the opposite of tensile strength. Compression strength is also measured in psi. When a piece of metal is cut, the material is subjected, as it comes in contact with the cutting edge, to a force known as shear. Shear is the tendency on the part of parallel members to slide in opposite directions. It is like placing a cord or thread between the blades of a pair of scissors (shears). The shear strength is the shear force in psi at which a material fails. It is the load divided by the shear area.

Bending can be described as the deflection or curving of a member due to forces acting upon it. The bending strength of material is the resistance it offers to deflecting forces. Torsion is a twisting force. Such action would occur in a member fixed at one end and twisted at the other. The torsional strength of material is its resistance to twisting.

The relationship between the strength of a material and its weight per cubic inch, expressed as a ratio, is known as the strength/weight ratio. This ratio forms the basis for comparing the desirability of various materials for use in airframe construction and repair.

Neither strength nor weight alone can be used as a means of true comparison. In some applications, such as the skin of monocoque structures, thickness is more important than strength, and, in this instance, the material with the lightest weight for a given thickness or gauge is best. Thickness or bulk is necessary to prevent bucking or damage caused by careless handling.

Corrosion is the eating away or pitting of the surface or the internal structure of metals. Because of the thin sections and the safety factors used in aircraft design and construction, it would be dangerous to select a material possessing poor corrosion resistant characteristics.

Another significant factor to consider in maintenance and repair is the ability of a material to be formed, bent, or machined to required shapes. The hardening of metals by cold working or forming is termed work hardening. If a piece of metal is formed (shaped or bent) while cold, it is said to be cold worked. Practically all the work an aviation mechanic does on metal is cold work. While this is convenient, it causes the metal to become harder and more brittle.

If the metal is cold worked too much, that is, if it is bent back and forth or hammered at the same place too often, it will crack or break. Usually, the more malleable and ductile a metal is, the more cold working it can stand. Any process which involves controlled heating and cooling of metals to develop certain desirable characteristics (such as hardness, softness, ductility, tensile strength, or refined grain structure) is called heat treatment or heat treating. With steels the term "heat treating" has a broad meaning and includes such processes as annealing, normalizing, hardening, and tempering.

Aircraft metals are subjected to both shock and fatigue (vibrational) stresses. Fatigue occurs in materials which are exposed to frequent reversals of loading or repeatedly applied loads. Repeated vibration or bending will ultimately cause a minute crack to occur at the weakest point. As vibration or bending continues, the crack lengthens until the part completely fails. This is termed shock and fatigue failure. Resistance to this condition is known as shock and fatigue resistance. It is essential that materials used for critical parts be resistant to these stresses.

HEAT TREATING

Heat treatment is a series of operations involving the heating and cooling of metals in the solid state. Its purpose is to change a mechanical property or combination of mechanical properties so that the metal will be more useful, serviceable, and safe for a definite purpose. By heat treating, a metal can be made harder, stronger, and more resistant to impact. Heat treating can also make a metal softer and more ductile. No one heat treating operation can produce all of these characteristics. In fact, some properties are often improved at the expense of others. In being hardened, for example, a metal may become brittle. The various heat-treating processes are similar in that they all involve the heating and cooling of metals. They differ, however, in the temperatures to which the metal is heated, the rate at which it is cooled, and, of course, in the final result.

The most common forms of heat treatment for ferrous metals are hardening, tempering, normalizing, annealing, and case hardening. Most nonferrous metals can be annealed and many of them can be hardened by heat treatment. However, there is only one nonferrous metal, titanium, that can be case hardened, and none can be tempered or normalized.

Internal Structure of Metals

The results obtained by heat treatment depend to a great extent on the structure of the metal and on the manner in which the structure changes when the metal is heated and cooled. A pure metal cannot be hardened by heat treatment because there is little change in its structure when heated. On the other hand, most alloys respond to heat treatment since their structures change with heating and cooling. An alloy may be in the form of a solid solution, a mechanical mixture, or a combination of a solid solution and a mechanical mixture. When an alloy is in the form of a solid solution, the elements and compounds which form the alloy are absorbed, one into the other, in much the same way that salt is dissolved in a glass of water, and the constituents cannot be identified even under a microscope.

When two or more elements or compounds are mixed but can be identified by microscopic examination, a mechanical mixture is formed. A mechanical mixture can be compared to the mixture of sand and gravel in concrete. The sand and gravel are both visible. Just as the sand and gravel are held together and kept in place by the matrix of cement, the other constituents of an alloy are embedded in the matrix formed by the base metal.

An alloy in the form of a mechanical mixture at ordinary temperatures may change to a solid solution when heated. When cooled back to normal temperature, the alloy may return to its original structure. On the other hand, it may remain a solid solution or form a combination of a solid solution and mechanical mixture. An alloy which consists of a combination of solid solution and mechanical mixture at normal temperatures may change to a solid solution when heated. When cooled, the alloy may remain a solid solution, return to its original structure, or form a complex solution.

Heat Treating Equipment

Successful heat treating requires close control over all factors affecting the heating and cooling of metals. Such control is possible only when the proper equipment is available and the equipment is selected to fit the particular job. Thus, the furnace must be of the proper size and type and must be so controlled that temperatures are kept within the limits prescribed for each operation. Even the atmosphere within the furnace affects the condition of the part being heat treated. Further, the quenching equipment and the quenching medium must be selected to fit the metal and the heat treating operation. Finally, there must be equipment for handling parts and materials, for cleaning metals, and for straightening parts.

FURNACES AND SALT BATHS

There are many different types and sizes of furnaces used in heat treatment. As a general rule, furnaces are designed to operate in certain specific temperature ranges and attempted use in other ranges frequently results in work of inferior quality. In addition, using a furnace beyond its rated maximum temperature shortens its life and may necessitate costly and time consuming repairs.

Fuel fired furnaces (gas or oil) require air for proper combustion and an air compressor or blower is therefore necessary. These furnaces are usually of the muffler type; that is, the combustion of the fuel takes place outside of and around the chamber in which the work is placed. If an open muffler is used, the furnace should be designed to prevent the direct impingement of flame on the work.

In furnaces heated by electricity, the heating elements are generally in the form of wire or ribbon. Good design requires incorporation of additional heating elements at locations where maximum heat loss may be expected. Such furnaces commonly operate at up to a maximum temperature of about 1 100°C. Furnaces operating at temperatures up to about 1 400°C usually employ resistor bars of sintered carbides.

Temperature Measurement and Control

Temperature in the heat-treating furnace is measured by a thermoelectric instrument known as a pyrometer. This instrument measures the electrical effect of a thermocouple and, hence, the temperature of the metal being treated. A complete pyrometer consists of three parts—a thermocouple, extension leads, and meter.

Furnaces intended primarily for tempering may be heated by gas or electricity and are frequently equipped with a fan for circulating the hot air.

Salt baths are available for operating at either tempering or hardening temperatures. Depending on the composition of the salt bath, heating can be conducted at temperatures as low as 165°C to as high as 1 350°C. Lead baths can be used in the temperature range of 345°C to 925°C. The rate of heating in lead or salt baths is much faster in furnaces.

Heat treating furnaces differ in size, shape, capacity, construction, operation, and control. They may be circular or rectangular and may rest on pedestals or directly on the floor. There are also pit type furnaces, which are below the surface of the floor. When metal is to be heated in a bath of molten salt or lead, the furnace must contain a pot or crucible for the molten bath.

The size and capacity of a heat treating furnace depends on the intended use. A furnace must be capable of heating rapidly and uniformly, regardless of the desired maximum temperature or the mass of the charge. An oven type furnace should have a working space (hearth) about twice as long and three times as wide as any part that will be heated in the furnace.

Accurate temperature measurement is essential to good heat treating. The usual method is by means of thermocouples: the most common base metal couples are copper-constantan (up to 370°C), iron-constantan (up to about 760°C), and chromel-alumel (up to 1 200°C). The most common noble metal couples (which can be used up to about 1 550°C) are platinum coupled with either the alloy 87 percent platinum (13 percent rhodium) or the alloy 90 percent platinum (10 percent rhodium). The temperatures quoted are for continuous operation.

The life of thermocouples is affected by the maximum temperature (which may frequently exceed those given above) and by the furnace atmosphere. Iron-constantan is more suited for use in reducing and chromel-alumel in oxidizing atmospheres. Thermocouples are usually encased in metallic or ceramic tubes closed at the hot end to protect them from the furnace gases. A necessary attachment is an instrument, such as a millivoltmeter or potentiometer, for measuring the electromotive force generated by the thermocouple. In the interest of accurate control, place the hot junction of the thermocouple as close to the work as possible. The use of an automatic controller is valuable in controlling the temperature at the desired value.

Pyrometers may have meters either of the indicating type or recording type. Indicating pyrometers give direct reading of the furnace temperature. The recording type produces a permanent record of the temperature range throughout the heating operation by means of an inked stylus attached to an arm which traces a line on a sheet of calibrated paper or temperature chart. Pyrometer installations on all modern furnaces provide automatic regulation of the temperature at any desired setting. Instruments of this type are called controlling potentiometer pyrometers. They include a current regulator and an operating mechanism, such as a relay.

HEATING

The object in heating is to transform pearlite (a mixture of alternate strips of ferrite and iron carbide in a single grain) to austenite as the steel is heated through the critical range. Since this transition takes time, a relatively slow rate of heating must be used. Ordinarily, the cold steel is inserted when the temperature in the furnace is from 150°C to 260°C below the hardening temperature. In this way, too rapid heating through the critical range is prevented.

If temperature measuring equipment is not available, it becomes necessary to estimate temperatures by some other means. An inexpensive, yet fairly accurate method involves the use of commercial crayons, pellets, or paints that melt at various temperatures within the range of 50°C to 870°C. The least accurate method of temperature estimation is by observation of the color of the hot hearth of the furnace or of the work. The heat colors observed are affected by many factors, such as the conditions of artificial or natural light, the character of the scale on the work, and so forth. Steel begins to appear dull red at about 550°C, and as the temperature increases, the color changes gradually through various shades of red to orange, to yellow, and finally to white. A rough approximation of the correspondence between color and temperature is indicated in ***Figure 1-2***.

It is also possible to secure some idea of the temperature of a piece of carbon or low alloy steel, in the low temperature range used for tempering, from the color of the thin oxide film that forms on the cleaned surface of the steel when heated in this range. The approximate temperature/color relationship is indicated on the lower portion of the scale in ***Figure 1-2***.

It is often necessary or desirable to protect steel or cast iron from surface oxidation (scaling) and loss of carbon from the surface layers (decarburization). Commercial furnaces, therefore, are generally equipped with some means of atmosphere control. This usually is in the form of a burner for burning controlled amounts of gas and air and directing the products of combustion into the furnace muffle. Water vapor, a product of this combustion, is detrimental and many furnaces are equipped with a means for eliminating it. For furnaces not equipped with atmosphere control, a variety of external atmosphere generators are available. The gas so generated is piped into the furnace and one generator may supply several furnaces. If no method of atmosphere control is available, some degree of protection may be secured by covering the work with cast iron borings or chips.

Since the work in salt or lead baths is surrounded by the liquid heating medium, the problem of preventing scaling or decarburization is simplified. Vacuum furnaces also are used for annealing steels, especially when a bright non-oxidized surface is a prime consideration.

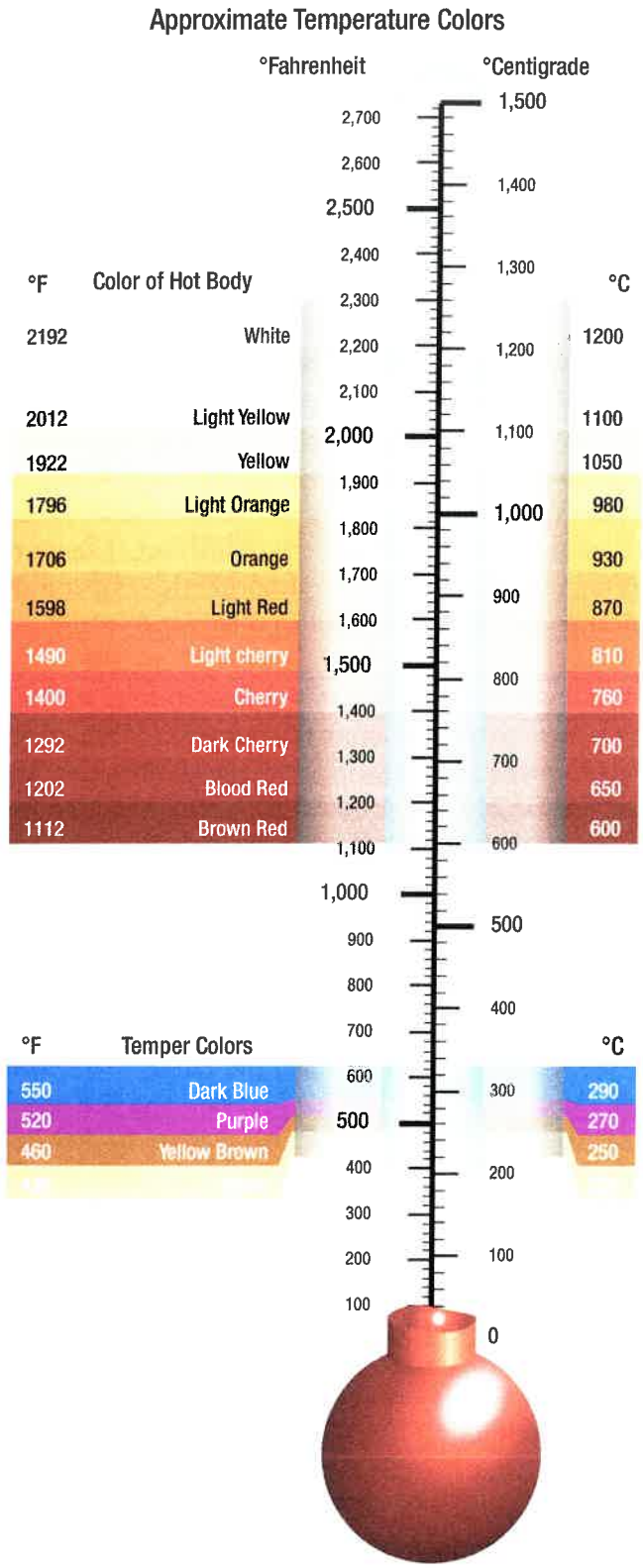


Figure 1-2. Temperature chart indicating conversion of Centigrade to Fahrenheit or vice versa, color temperature scale for hardening temperature range, and tempering temperature range.

SOAKING

The temperature of the furnace must be held constant during the soaking period, since it is during this period that rearrangement of the internal structure of the steel takes place. Soaking temperatures for various types of steel are specified in ranges varying as much as 40°C. (***Figure 1-3***) Small parts are soaked in the lower part of the specified range and heavy parts in the upper part of the specified range. The length of the soaking period depends upon the type of steel and the size of the part. Naturally, heavier parts require longer soaking to ensure equal heating throughout. As a general rule, a soaking period of 30 minutes to 1 hour is sufficient for the average heat-treating operation.

COOLING

The rate of cooling through the critical range determines the form that the steel will retain. Various rates of cooling are used to produce the desired results. Still air is a slow cooling medium, but is much faster than furnace cooling. Liquids are the fastest cooling media and are therefore used in hardening steels. There are three commonly used quenching liquids—brine, water, and oil. Brine is the strongest quenching medium, water is next, and oil is the least. Generally, an oil quench is used for alloy steels, and brine or water for carbon steels.

Quenching solutions act only through their ability to cool the steel. They have no beneficial chemical action on the quenched steel and in themselves impart no unusual properties. Most requirements for quenching media are met satisfactorily by water or aqueous solutions of inorganic salts, such as table salt or caustic soda, or by some type of oil. The rate of cooling is relatively rapid during quenching in brine, somewhat less rapid in water, and slow in oil.

Brine usually is made of a 5 to 10 percent solution of salt (sodium chloride) in water. In addition to its greater cooling speed, brine has the ability to "throw" the scale from steel during quenching. The cooling ability of both water and brine, particularly water, is considerably affected by their temperature. Both should be kept cold—well below 15°C. If the volume of steel being quenched tends to raise the temperature of the bath appreciably, add ice or use some means of refrigeration to cool the quenching bath.

There are many specially prepared quenching oils on the market; their cooling rates do not vary widely. A straight mineral oil with a Saybolt viscosity of about 100 at 38°C is generally used. Unlike brine and water, the oils have the greatest cooling velocity at a slightly elevated temperature—about 38 to 60°C—because of their decreased viscosity at these temperatures.

When steel is quenched, the liquid in immediate contact with the hot surface vaporizes; this vapor reduces the rate of heat abstraction markedly. Vigorous agitation of the steel or the use of a pressure spray quench is necessary to dislodge these vapor films and thus permit the desired rate of cooling. The tendency of steel to warp and crack during the quenching process is difficult to overcome because certain parts of the article cool more rapidly than others. The following recommendations will greatly reduce the warping tendency:

1. Never throw a part into the quenching bath. By permitting it to lie on the bottom of the bath, it is apt to cool faster on the top side than on the bottom side, thus causing it to warp or crack.
2. Agitate the part slightly to destroy the coating of vapor that could prevent it from cooling evenly and rapidly. This allows the bath to dissipate its heat to the atmosphere.
3. Immerse irregular shaped parts so that the heavy end enters the bath first.

QUENCHING EQUIPMENT

The quenching tank should be of the proper size to handle the material being quenched. Use circulating pumps and coolers to maintain approximately constant temperatures when doing a large amount of quenching. To avoid building up a high concentration of salt in the quenching tank, make provisions for adding fresh water to the quench tank used for molten salt baths.

Tank location in reference to the heat treating furnace is very important. Situate the tank to permit rapid transfer of the part from the furnace to the quenching medium. A delay of more than a few seconds will, in many instances, prove detrimental to the effectiveness of the heat treatment. When heat treating material of thin section, employ guard sheets to retard the loss of heat during transfer to the quench tank. Provide a rinse tank to remove all salt from the material after quenching if the salt is not adequately removed in the quenching tank.

HEAT TREATMENT OF FERROUS METALS

The first important consideration in the heat treatment of a steel part is to know its chemical composition. This, in turn, determines its upper critical point. When the upper critical point is known, the next consideration is the rate of heating and cooling to be used. Carrying out these operations involves the use of uniform heating furnaces, proper temperature controls, and suitable quenching mediums.

BEHAVIOR OF STEEL DURING HEATING AND COOLING

Changing the internal structure of a ferrous metal is accomplished by heating to a temperature above its upper critical point, holding it at that temperature for a time sufficient to permit certain internal changes to occur, and then cooling to atmospheric temperature under predetermined, controlled conditions.

At ordinary temperatures, the carbon in steel exists in the form of particles of iron carbide scattered throughout an iron matrix known as "ferrite." The number, size, and distribution of these particles determine the hardness of the steel. At elevated temperatures, the carbon is dissolved in the iron matrix in the form of a solid solution called "austenite," and the carbide particles appear only after the steel has been cooled. If the cooling is slow, the carbide particles are relatively coarse and few. In this condition, the steel is soft. If the cooling is rapid, as by quenching in oil or water, the carbon precipitates as a cloud of very fine carbide particles, and the steel is hard. The fact that the carbide particles can be dissolved in austenite is the basis of the heat treatment of steel. The temperatures at which this transformation takes place are called the critical points and vary with the composition of the steel. The percentage of carbon in the steel has the greatest influence on the critical points of heat treatment.

HARDENING

Pure iron, wrought iron, and extremely low carbon steels cannot be appreciably hardened by heat treatment, since they contain no hardening element. Cast iron can be hardened, but its heat treatment is limited. When cast iron is cooled rapidly, it forms white iron, which is hard and brittle. When cooled slowly, it forms gray iron, which is soft but brittle under impact.

Steel Number	Temperatures			Quenching Medium (n)	Tempering (drawing) Temperature for Tensile Strength (psi)				
	Normalizing Air Cool (°C)	Annealing (°C)	Hardening (°C)		100 000 (°C)	125 000 (°C)	150 000 (°C)	180 000 (°C)	200 000 (°C)
1020	900–955	870–925	855–910	Water	—	—	—	—	—
1022 (x1020)	900–955	870–925	855–910	Water	—	—	—	—	—
1025	870–925	870–900	855–910	Water	(a)	—	—	—	—
1035	855–900	855–885	830–870	Water	470	—	—	—	—
1045	845–870	845–870	800–845	Oil or Water	620	—	—	(n)	—
1095	800–845	790–815	775–815	Oil	(b)	—	595	455	400
2330	800–830	775–800	790–815	Oil or Water	595	510	423	—	—
3135	870–900	815–845	800–830	Oil	675	565	480	400	345
3140	870–900	815–845	800–830	Oil	720	580	495	413	370
4037	870	830–855	830–855	Oil or Water	665	595	525	—	—
4130 (x4130)	870–925	830–855	830–885	Oil (c)	(d)	565	480	370	300
4140	870–900	830–855	830–855	Oil	730	595	550	440	355
4150	845–870	800–830	845	Oil	—	690	635	565	510
4340 (x4340)	845–885	830–855	800–845	Oil	—	650	565	510	455
4640	910–925	830–855	815–845	Oil	—	650	565	400	330
6135	870–925	845–870	855–885	Oil	705	580	495	425	400
6150	870–900	830–855	845–885	Oil	(d)(e)	650	540	480	425
6195	870–900	830–855	815–845	Oil	(f)	—	—	—	—
NE8620	—	—	830–855	Oil	—	540	—	—	—
NE8630	900	830–855	830–855	Oil	—	605	525	415	355
NE8735	900	830–855	830–855	Oil	—	635	550	470	415
NE8740	885	815–845	815–845	Oil	—	650	580	495	455
30905	—	(g)(h)	(l)	—	—	—	—	—	—
51210	830–855	830–855	970–995(j)	Oil	650	595	(k)	400	—
51335	—	830–855	970–1 010	Oil	—	—	—	—	—
52100	885–925	760–790	830–845	Oil	(f)	—	—	—	—
Corrosion resisting (16–2)(1)	—	—	—	—	(m)	—	—	—	—
Silicon chromium (for springs)	—	—	925–940	Oil	—	—	—	—	—

NOTES:

(a) Draw at 620°C for tensile strength of 70 000 psi.

(b) For spring temper draw at 425-480°C. Rockwell hardness C-40-45.

(c) Bars or forgings may be quenched in water from 815–870°C.

(d) Air cooling from the normalizing temperature produces a tensile strength of approximately 90 000 psi.

(e) For spring temper draw at 455–510°C. Rockwell hardness C-40-45.

(f) Draw at 175–230°C to remove quenching strains. Rockwell hardness C-60-65.

(g) Anneal at 870–925°C to remove residual stresses due to welding or cold work. May be applied only to steel containing titanium or columbium.

(h) Anneal at 1 040–1 150°C to produce maximum softness and corrosion resistance. Cool in air or quench in water.

(l) Harden by cold work only.

(j) Lower side of range for sheet 0.06 inch and under. Middle of range for sheet and wire 0.125 inch. Upper side of range for forgings.

(k) Not recommended for intermediate tensile strengths because of low impact.

(l) AN-QQ-S-770—It is recommended that prior to tempering corrosion–resisting (16 Cr–2 Ni) steel be quenched in oil from a temperature of 1 025–1 040°C after a soaking period of 30 minutes at this temperature. To obtain a tensile strength at 115 000 psi the tempering temperature should be approximately 275°C. A holding time at these temperatures of about 2 hours is recommended. Tempering temperatures between 370°C and 595°C is not approved.

(m) Draw at approximately 425°C and cool in air for Rockwell hardness of C-50.

(n) Water used for quenching shall be within the temperature range of 25–60°C.

Figure 1-3. Heat treatment procedures for steels.

In plain carbon steel, the maximum hardness depends almost entirely on the carbon content of the steel. As the carbon content increases, the ability of the steel to be hardened increases. However, this increase in the ability to harden with an increase in carbon content continues only to a certain point. In practice, that point is 0.85 percent carbon content. When the carbon content is increased beyond 0.85 percent, there is no increase in wear resistance.

For most steels, the hardening treatment consists of heating the steel to a temperature just above the upper critical point, soaking or holding for the required length of time, and then cooling it rapidly by plunging the hot steel into oil, water, or brine. Although most steels must be cooled rapidly for hardening, a few may be cooled in still air. Hardening increases the hardness and strength of the steel but makes it less ductile.

When hardening carbon steel, it must be cooled to below 540°C in less than 1 second. Should the time required for the temperature to drop to 540°C exceed 1 second, the austenite begins to transform into fine pearlite. This pearlite varies in hardness, but is much harder than the pearlite formed by annealing and much softer than the martensite desired. After the 540°C temperature is reached, the rapid cooling must continue if the final structure is to be all martensite.

When alloys are added to steel, the time limit for the temperature drop to 540°C increases above the 1 second limit for carbon steels. Therefore, a slower quenching medium will produce hardness in alloy steels. Because of the high internal stresses in the "as quenched" condition, steel must be tempered just before it becomes cold. The part should be removed from the quenching bath at a temperature of approximately 95°C, since the temperature range from 95°C down to room temperature is the cracking range. Hardening temperatures and quenching mediums for the various types of steel are listed in ***Figure 1-3***.

Hardening Precautions

A variety of different shapes and sizes of tongs for handling hot steels is necessary. It should be remembered that cooling of the area contacted by the tongs is retarded and that such areas may not harden, particularly if the steel being treated is very shallow hardening. Small parts may be wired together or quenched in baskets made of wire mesh. Special quenching jigs and fixtures are frequently used to hold steels during quenching in a manner to restrain distortion.

When selective hardening is desired, portions of the steel may be protected by covering with alundum cement or some other insulating material. Selective hardening may be accomplished also by the use of water or oil jets designed to direct quenching medium on the areas to be hardened. This also is accomplished by the induction and flame hardening procedures previously described, particularly on large production jobs.

Shallow hardening steels, such as plain carbon and certain varieties of alloy steels, have such a high critical cooling rate that they must be quenched in brine or water to effect hardening. In general, intricately shaped sections should not be made of shallow hardening steels because of the tendency of these steels to warp and crack during hardening. Such items should be made of deeper hardening steels capable of being hardened by quenching in oil or air.

TEMPERING

Tempering reduces the brittleness imparted by hardening and produces definite physical properties within the steel. Tempering always follows, never precedes, the hardening operation. In addition to reducing brittleness, tempering softens the steel. Tempering is always conducted at temperatures below the low critical point of the steel. In this respect, tempering differs from annealing, normalizing, or hardening, all of which require temperatures above the upper critical point. When hardened steel is reheated, tempering begins at 100°C and continues as the temperature increases toward the low critical point. By selecting a definite tempering temperature, the resulting hardness and strength can be predetermined. Approximate temperatures for various tensile strengths are listed in ***Figure 1-3***. The minimum time at the tempering temperature should be 1 hour. If the part is over 1 inch in thickness, increase the time by 1 hour for each additional inch of thickness. Tempered steels used in aircraft work have from 125 000 to 200 000 psi ultimate tensile strength.

Generally, the rate of cooling from the tempering temperature has no effect on the resulting structure; therefore, the steel is usually cooled in still air after being removed from the furnace.

ANNEALING

Annealing of steel produces a fine grained, soft, ductile metal without internal stresses or strains. In the annealed state, steel has its lowest strength. In general, annealing is the opposite of hardening.

Annealing of steel is accomplished by heating the metal to just above the upper critical point, soaking at that temperature, and cooling very slowly in the furnace. (***Figure 1-3*** for recommended temperatures.) Soaking time is approximately 1 hour per inch of thickness of the material. To produce maximum softness in steel, the metal must be cooled very slowly. Slow cooling is obtained by shutting off the heat and allowing the furnace and metal to cool together to 480°C or lower, then removing the metal from the furnace and cooling in still air. Another method is to bury the heated steel in ashes, sand, or other substance that does not conduct heat readily.

NORMALIZING

The normalizing of steel removes the internal stresses set up by heat treating, welding, casting, forming, or machining. Stress, if not controlled, will lead to failure.

Because of the better physical properties, aircraft steels are often used in the normalized state, but seldom, if ever, in the annealed state. One of the most important uses of normalizing in aircraft work is in welded parts. Welding causes strains to be set up in the adjacent material. In addition, the weld itself is a cast structure as opposed to the wrought structure of the rest of the material. These two types of structures have different grain sizes, and to refine the grain as well as to relieve the internal stresses, all welded parts should be normalized after fabrication.

Normalizing is accomplished by heating the steel above the upper critical point and cooling in still air. The more rapid quenching obtained by air cooling, as compared to furnace cooling, results in a harder and stronger material than that obtained by annealing. Recommended normalizing temperatures for the various types of aircraft steels are listed in ***Figure 1-3***.

CASE HARDENING

Case hardening produces a hard wear resistant surface or case over a strong, tough core. Case hardening is ideal for parts which require a wear resistant surface and, at the same time, must be tough enough internally to withstand the applied loads. The steels best suited to case hardening are the low carbon and low alloy steels. If high carbon steel is case hardened, the hardness penetrates the core and causes brittleness. In case hardening, the surface of the metal is changed chemically by introducing a high carbide or nitride content. The core is unaffected chemically.

When heat treated, the surface responds to hardening while the core toughens. The common forms of case hardening are carburizing, cyaniding, and nitriding. Since cyaniding is not used in aircraft work, only carburizing and nitriding are discussed in this section.

Carburizing

Carburizing is a case hardening process in which carbon is added to the surface of low carbon steel. Thus, a carburized steel has a high carbon surface and a low carbon interior. When the carburized steel is heat treated, the case is hardened while the core remains soft and tough.

A common method of carburizing is called "pack carburizing." When carburizing is to be done by this method, the steel parts are packed in a container with charcoal or some other material rich in carbon. The container is then sealed with fire clay, placed in a furnace, heated to approximately 925°C, and soaked at that temperature for several hours. As the temperature increases, carbon monoxide gas forms inside the container and, being unable to escape, combines with the gamma iron in the surface of the steel. The depth to which the carbon penetrates depends on the length of the soaking period. For example, when carbon steel is soaked for 8 hours, the carbon penetrates to a depth of about 0.062 inch.

Another method of carburizing is called "gas carburizing," a material rich in carbon is introduced into the furnace atmosphere. The carburizing atmosphere is produced by the use of various gases or by the burning of oil, wood, or other materials. When the steel parts are heated in this atmosphere, carbon monoxide combines with the gamma iron to produce practically the same results as those described under the pack carburizing process.

A third method of carburizing is that of "liquid carburizing." In this method, the steel is placed in a molten salt bath that contains the chemicals required to produce a case comparable with one resulting from pack or gas carburizing.

Alloy steels with low carbon content as well as low carbon steels may be carburized by any of the three processes. However, some alloys, such as nickel, tend to retard the absorption of carbon. As a result, the time required to produce a given thickness of case varies with the composition of the metal.

Nitriding

Nitriding is unlike other casehardening processes in that, before nitriding, the part is heat treated to produce definite physical properties. Thus, parts are hardened and tempered before being nitrided. Most steels can be nitrided, but special alloys are required for best results. These special alloys contain aluminum as one of the alloying elements and are called "nitralloys."

In nitriding, the part is placed in a special nitriding furnace and heated to a temperature of approximately 540°C. With the part at this temperature, ammonia gas is circulated within the specially constructed furnace chamber. The high temperature cracks the ammonia gas into nitrogen and hydrogen. The ammonia which does not break down is caught in a water trap below the regions of the other two gases. The nitrogen reacts with the iron to form nitride. The iron nitride is dispersed in minute particles at the surface and works inward.

The depth of penetration depends on the length of the treatment. In nitriding, soaking periods as long as 72 hours are frequently required to produce the desired thickness of case. Nitriding can be accomplished with a minimum of distortion, because of the low temperature at which parts are casehardened and because no quenching is required after exposure to the ammonia gas.

FORGING

Forging is the process of forming a product by hammering or pressing. When the material is forged below the recrystallization temperature, it is called cold forged. When worked above the recrystallization temperature, it is referred to as hot forged. Drop forging is a hammering process that uses a hot ingot that is placed between a pair of formed dies in a machine called a drop hammer and a weight of several tons is dropped on the upper die. This results in the hot metal being forced to take the form of the dies. Because the process is very rapid, the grain structure of the metal is altered, resulting in a significant increase in the strength of the finished part.

CASTING

Casting is formed by melting the metal and pouring it into a mold of the desired shape. Since plastic deformation of the metal does not occur, no alteration of the grain shape or orientation is possible. The grain size of the metal can be controlled by the cooling rate, the alloys of the metal, and the thermal treatment. Castings are normally lower in strength and are more brittle than a wrought product of the same material. For intricate shapes or items with internal passages, such as turbine blades, casting may be the most economical process. Except for engine parts, most metal components found on an aircraft are wrought instead of cast.

All metal products start in the form of casting. Wrought metals are converted from cast ingots by plastic deformation. For high strength aluminum alloys, an 80 to 90 percent reduction (dimensional change in thickness) of the material is required to obtain the high mechanical properties of a fully wrought structure.

Both iron and aluminum alloys are cast for aircraft uses. Cast iron contains 6 to 8 percent carbon and silicon. Cast iron is a hard unmalleable pig iron made by casting or pouring into a mold. Cast aluminum alloy has been heated to its molten state and poured into a mold to give it the desired shape.

EXTRUDING

The extrusion process involves the forcing of metal through an opening in a die, thus causing the metal to take the shape of the die opening. The shape of the die will be the cross section of an angle, channel, tube, or some other shape. Some metals such as lead, tin, and aluminum may be extruded cold; however, most metals are heated before extrusion. The main advantage of the extrusion process is its flexibility. For example, because of its workability, aluminum can be economically extruded to more intricate shapes and larger sizes than is practical with other metals.

Extruded shapes are produced in very simple as well as extremely complex sections. In this process a cylinder of aluminum, for instance, is heated to 400–455°C and is then forced through the opening of a die by a hydraulic ram. The opening is the shape desired for the cross section of the finished extrusion. Many structural parts, such as channels, angles, T-sections, and Z-sections, are formed by the extrusion process.

Aluminum is the most extruded metal used in aircraft. Aluminum is extruded at a temperature of 371–482°C and requires pressure of up to 80 000 psi (552 MPa). After extrusion, the product frequently will be subjected to both thermal and mechanical processes to obtain the desired properties. Extrusion processes are limited to the more ductile materials.

COLD WORKING/HARDENING

Cold working applies to mechanical working performed at temperatures below the critical range. It results in a strain hardening of the metal. In fact, the metal often becomes so hard that it is difficult to continue the forming process without softening the metal by annealing.

Since the errors attending shrinkage are eliminated in cold working, a much more compact and better metal is obtained. The strength and hardness, as well as the elastic limit, are increased; but the ductility decreases. Since this makes the metal more brittle, it must be heated from time to time during certain operations to remove the undesirable effects of the working.

While there are several cold working processes, the two with which the aviation mechanic will be principally concerned are cold rolling and cold drawing. These processes give the metals desirable qualities which cannot be obtained by hot working.

Cold rolling usually refers to the working of metal at room temperature. In this operation, the materials that have been rolled to approximate sizes are pickled to remove the scale, after which they are passed through chilled finishing rolls. This gives a smooth surface and also brings the pieces to accurate dimensions. The principal forms of cold rolled stocks are sheets, bars, and rods.

Cold drawing is used in making seamless tubing, wire, streamlined tie rods, and other forms of stock. Wire is made from hot rolled rods of various diameters. These rods are pickled in acid to remove scale, dipped in lime water, and then dried in a steam room where they remain until ready for drawing. The lime coating adhering to the metal serves as a lubricant during the drawing operation.

The size of the rod used for drawing depends upon the diameter wanted in the finished wire. To reduce the rod to the desired size, it is drawn cold through a die. One end of the rod is filed or hammered to a point and slipped through the die opening. Here it is gripped by the jaws of the drawing block and pulled through the die. This series of operations is done by a mechanism known as a drawbench. To reduce the rod gradually to the desired size, it is necessary to draw the wire through successively smaller dies. Because each of these drawings reduces the ductility of the wire, it must be annealed from time to time before further drawings can be accomplished. Although cold working reduces the ductility, it increases the tensile strength of the wire.

In making seamless steel aircraft tubing, the tubing is cold drawn through a ring shaped die with a mandrel or metal bar inside the tubing to support it while the drawing operations are being performed. This forces the metal to flow between the die and the mandrel and affords a means of controlling the wall thickness and the inside and outside diameters.

6.1(B) - HARDNESS TESTING

Hardness testing is a method of determining the results of heat treatment as well as the state of a metal prior to heat treatment. Since hardness values can be tied in with tensile strength values and, in part, with wear resistance, hardness tests are a valuable check of heat-treat control and of material properties.

Practically all hardness testing equipment now uses the resistance to penetration as a measure of hardness. Included among the better known hardness testers are the Brinell and Rockwell, both of which are described and illustrated in this section. Also included is a popular portable-type hardness tester currently in use.

BRINELL TESTER

The Brinell hardness tester (***Figure 1-4***) uses a hardened spherical ball, which is forced into the surface of the metal. This ball is 10 millimeters (0.393 7 inch) in diameter. A pressure of 3 000 kilograms is used for ferrous metals and 500 kilograms for nonferrous metals. The pressure must be maintained at least 10 seconds for ferrous metals and at least 30 seconds for nonferrous metals. The load is applied by hydraulic pressure. The hydraulic pressure is built up by a hand pump or an electric motor, depending on the model of tester. A pressure gauge indicates the amount of pressure. There is a release mechanism for relieving the pressure after the test has been made, and a calibrated microscope is provided for measuring the diameter of the impression in millimeters. The machine has various shaped anvils for supporting the specimen and an elevating screw for bringing the specimen in contact with the ball penetrator. These are attachments for special tests.

To determine the Brinell hardness number for a metal, measure the diameter of the impression, using the calibrated microscope furnished with the tester. Then convert the measurement into the Brinell hardness number on the conversion table furnished with the tester.

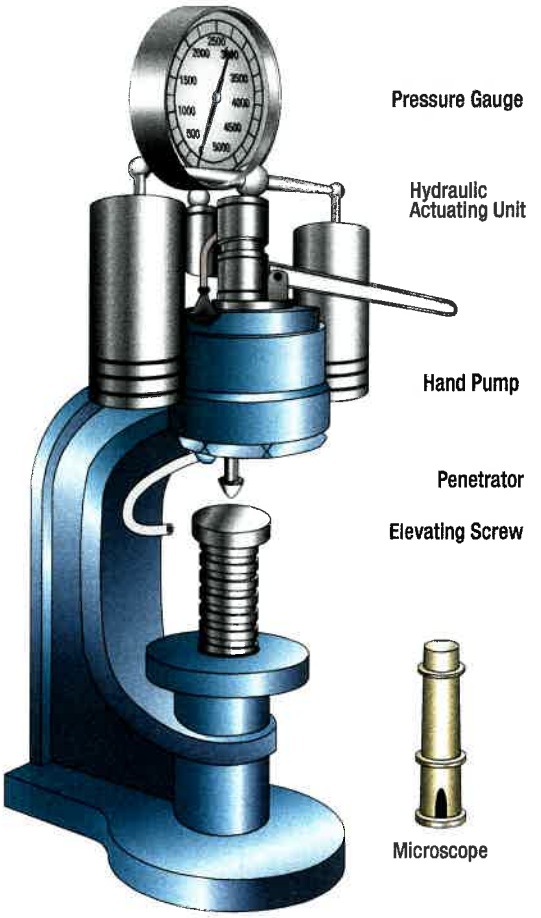


Figure 1-4. Brinell hardness tester.

ROCKWELL TESTER

The Rockwell hardness tester (***Figure 1-5***) measures the resistance to penetration, as does the Brinell tester. Instead of measuring the diameter of the impression, the Rockwell tester measures the depth, and the hardness is indicated directly on a dial attached to the machine.

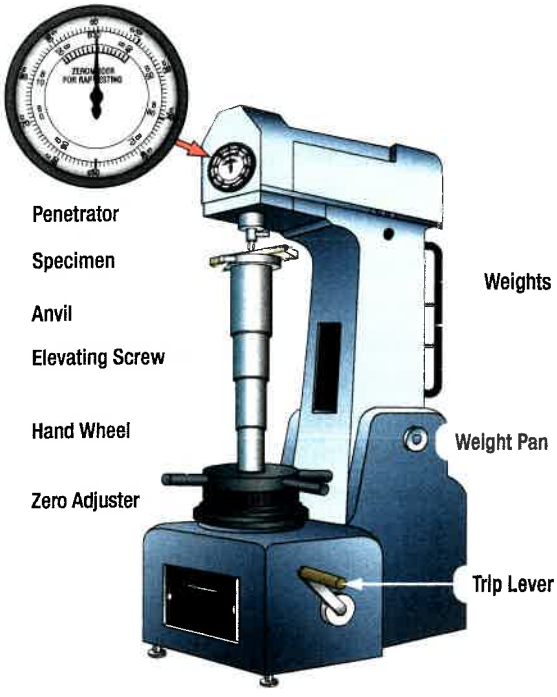


Figure 1-5. Rockwell hardness tester.

The dial numbers in the outer circle are black, and the inner numbers are red. Rockwell hardness numbers are based on the difference between the depth of penetration at major and minor loads. The greater this difference, the lower the hardness number and the softer the material.

Two types of penetrators are used with the Rockwell tester: a diamond cone and a hardened steel ball. The load which forces the penetrator into the metal is called the major load and is measured in kilograms. The results of each penetrator and load combination are reported on separate scales, designated by letters.

The penetrator, the major load, and the scale vary with the kind of metal being tested. For hardened steels, the diamond penetrator is used; the major load is 150 kilograms; and the hardness is read on the "C" scale. When this reading is recorded, the letter "C" must precede the number indicated by the pointer.

The C-scale setup is used for testing metals ranging in hardness from C-20 to the hardest steel (usually about C-70). If the metal is softer than C-20, the B-scale setup is used. With this setup, the 1/16-inch ball is used as a penetrator; the major load is 100 kilograms; and the hardness is read on the B-scale. In addition to the "C" and "B" scales, there are other setups for special testing. The scales, penetrators, major loads, and dial numbers to be read are listed in ***Figure 1-6***.

The Rockwell tester is equipped with a weight pan, and two weights are supplied with the machine. One weight is marked in red. The other weight is marked in black. With no weight in the weight pan, the machine applies a major load of 60 kilograms. If the scale setup calls for a 100 kilogram load, the red weight is placed in the pan. For a 150 kilogram load, the black weight is added to the red weight. The black weight is always used with the red weight; it is never used alone.

Scale Symbol	Penetrator	Major Load (kg)	Dial Color/Number
A	Diamond	60	Black
B	1/16-inch ball	100	Red
C	Diamond	150	Black
D	Diamond	100	Black
E	1/8-inch ball	100	Red
F	1/16-inch ball	60	Red
G	1/16-inch ball	150	Red
H	1/8-inch ball	60	Red
K	1/8-inch ball	150	Red

Figure 1-6. Standard Rockwell hardness scales.

Practically all testing is done with either the B-scale setup or the C-scale setup. For these scales, the colors may be used as a guide in selecting the weight (or weights) and in reading the dial. For the B-scale test, use the red weight and read the red numbers. For a C-scale test, add the black weight to the red weight and read the black numbers.

In setting up the Rockwell machine, use the diamond penetrator for testing materials known to be hard. If the hardness is unknown, try the diamond, since the steel ball may be deformed if used for testing hard materials. If the metal tests below C-22, then change to the steel ball. Use the steel ball for all soft materials, those testing less than B-100. Should an overlap occur at the top of the B-scale and the bottom of the C-scale, use the C-scale setup.

Before the major load is applied, securely lock the test specimen in place to prevent slipping and to seat the anvil and penetrator properly. To do this, apply a load of 10 kilograms before the lever is tripped. This preliminary load is called the minor load. The minor load is 10 kilograms regardless of the scale setup.

The metal to be tested in the Rockwell tester must be ground smooth on two opposite sides and be free of scratches and foreign matter. The surface should be perpendicular to the axis of penetration, and the two opposite ground surfaces should be parallel. If the specimen is tapered, the amount of error will depend on the taper. A curved surface will also cause a slight error in the hardness test. The amount of error depends on the curvature; i.e., the smaller the radius of curvature, the greater the error. To eliminate such error, a small flat should be ground on the curved surface if possible.

Clad aluminum alloy sheets cannot be tested directly with any accuracy with a Rockwell hardness tester. If the hardness value of the base metal is desired, the pure aluminum coating must be removed from the area to be checked prior to testing.

OTHER TESTING

TENSILE STRENGTH TESTING

Materials strength testing is largely done by manufacturer's and design engineers long before the field technician ever encounters the aircraft for maintenance. It is vitally important that all materials, not just metals, have the strength required for the application in which they are used. The tensile strength of a material is what is commonly thought of as the strength of a material. It is a measurement of tension, which is the stress that resists a force that tends to pull a material apart. To perform a test of tensile strength, the material is secured in a specially designed electromagnetically or hydraulically powered machine that exerts force on the material to pull it apart. While the force is applied and increased, dimensional measurements are made and recorded. Most materials will elongate before the force becomes so great that there is critical failure. Many materials also have an elastic range where a relatively small amount of force elongates the material, and, when released, the material returns to it's original dimensions.

A tensile strength test could be performed by clamping a sample of the material securely and then hanging weight from it until it fails. From this, one can understand that tensile strength is a weight-related measurement. But the area of the sample is an important factor that contributes to when a sample will fail. Thus, to be able to compare tensile strengths of various materials, weight per unit area is used. In the Imperial system, this is denominated in of pounds per square inch (psi). In SI, measurement is typically in Pascals (Pa), hectoPascals (hPa) or kiloPascals (kPa) depending on the relative strength of the material being tested.

Many materials are tested along a single axis since they are isotropic. This means tensile strength is uniform in all directions. Aviation metals are in this category. Aircraft fabrics and composites are anisotropic, meaning that tensile strength varies depending upon which direction the force is applied. Standards exist for size, and shape of the sample material to be tested and the attachment methods into the testing machine are also carefully regulated. Alignment is important as well, especially in anisotropic. Test results can be graphed and numerous characteristics of the material can be examined or derived. ***Figure 1-7*** illustrated a tensile testing machine.

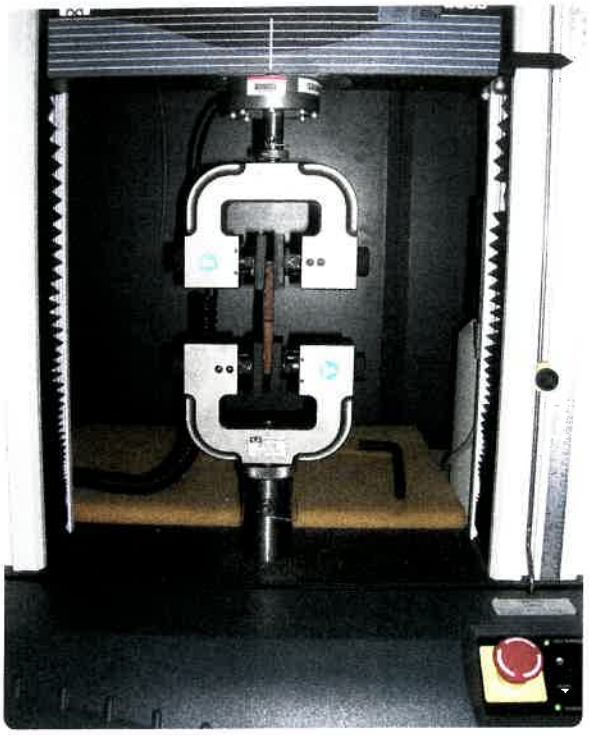


Figure 1-7. A tensile testing machine.

FATIGUE STRENGTH TESTING

Fatigue is a weakness in materials, especially metals, caused by repeated variations of stress. Fatigue is of great concern in aviation due to the constantly changing loads experience by nearly all parts of the structure during operation. Older aircraft are particularly suspect of weakness due to fatigue. The critical issue with fatigue is that numerous, repeated applications of over and over a relatively small load can result in failure. A single small load of the size under considerations size would have virtually no effect on the material. But application and release of the small load gradually weakens it. An example of fatigue failure is bending a piece of metal back and force until is weakened and breaks.

Aircraft and materials manufacturer's cannot rely solely on tradition strength measurements such as tensile strength or shear strength. Simple and elaborate repeated load and unload machines are constructed to observe fatigue in various structural aircraft components. There are also commercially available machines designed to test the fatigue strength of a variety of components. Stresses beyond those projected during service life are produced. The effects are measured so the component can be designed to perform without failure when installed on the aircraft.

Metal fatigue can cause micro cracks and other fatigue damage precursors that are sought during periodic inspection. Visual, eddy current, ultra sound, fluorescent penetrant, magnetic particle, and even radiography inspection techniques are all employed in various situation to detect metal fatigue. The progression from a micro crack to cracks that join to cause a large section of metal to fail can be rapid. It is critical, especially on aircraft with a high number of operating cycles, to follow all manufacturer's specifications for investigating fatigue. New studies and method of detection are being developed such as electrochemical and light scattering inspection processes. Some country authorities have placed limits on the number of cycles an aircraft may accrue before it must be frequently inspected or retired from service.

IMPACT TESTING

Impact tests are used to indicate the toughness of a material and most importantly its ability to resist mechanical shock, to ensure that temper brittleness has not been introduced during heat treatments.

Toughness is, broadly, a measure of the amount of energy required to cause an item - a test piece or a bridge or a pressure vessel - to fracture and fail. The more energy that is required then the tougher the material.

There are two types of machine used for testing aircraft materials, both of which use a pendulum weight to fracture the specimen. The energy absorbed by the specimen is measured from the angle through which the pendulum swings after causing the fracture. The IZOD test is required by most of the British material specifications, but where the test piece must be tested at high or low temperatures the CHARPY test is used. The test is carried out within the 6 seconds of removal of the test piece from the heating or cooling bath. Machines are available which carry out both the Izod and Charpy tests. (***Figure 1-8***)

ELECTROCHEMICAL TESTING OF STAINLESS VS INCONEL

Prepare a wiring assembly as shown in ***Figure 1-9***, and prepare the two reagents (ammonium fluoride and dimethylglyoxime solutions) placing them in separate dedicated dropper solution bottles. Before testing, you must thoroughly clean the metal in order for the electrolytic deposit to take place. You may use nonmetallic hand scrubbing pads or 320 to 600 grit "crocus cloth" to remove deposits and corrosion products (thermal oxide).

Connect the alligator clip of the wiring assembly to the bare metal being tested. Place one drop of a 0.05 percent reagent grade ammonium fluoride solution in deionized water on the center of a 1 inch × 1 inch sheet of filter paper. Lay the moistened filter paper over the bare metal alloy being tested. Firmly press the end of the aluminum rod over the center of the moist paper. Maintain connection for 10 seconds while rocking the aluminum rod on the filter paper. Ensure that the Light Emitting Diode (LED) remains lit (indicating good electrical contact and current flow) during this period.



Figure 1-8. An IZOD impact test machine.

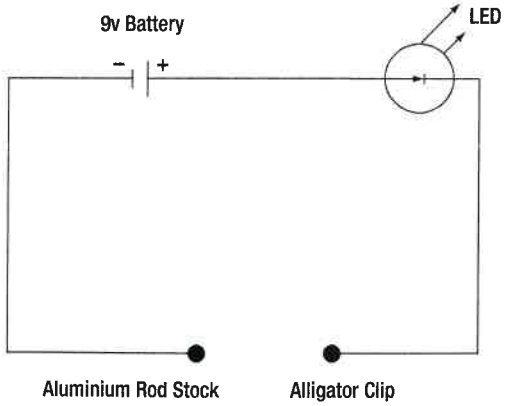


Figure 1-9. Wiring assembly schematic.

Disconnect the wiring assembly and set it aside. Remove the filter paper and examine it to determine that a light spot appears where the connection was made.

Deposit one drop of 1.0 percent solution of reagent grade dimethylglyoxime in ethyl alcohol on the filter paper (same side that was in contact with the test metal). A bright, distinctly pink spot will appear within seconds on the filter paper if the metal being tested is Inconel. A brown spot will appear if the test metal is stainless steel. Some stainless steel alloys may leave a very light pink color. However, the shade and depth of color will be far less than would appear for Inconel. For flat surfaces, the test spot will be circular while for curved surfaces, such as the outside of a tube or pipe, the test spot may appear as a streak. (Refer to ***Figure 1-10*** for sample test results.) This procedure should not be used in the heat affected zone of weldments or on nickel coated surfaces.

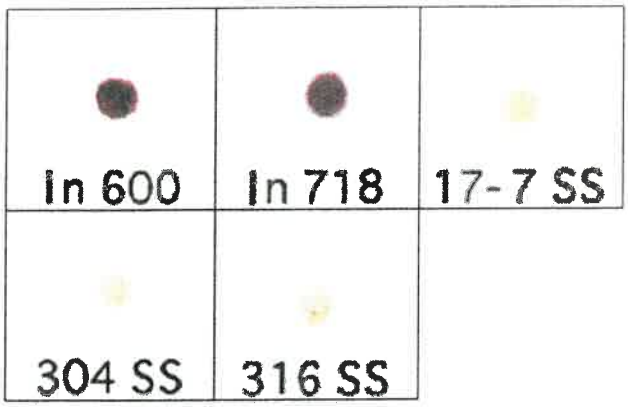


Figure 1-10. Electrochemical test results, Inconel and stainless steel alloys.

Question: 1-1
A metal which can be hammered, rolled, or pressed into various shapes without cracking, breaking, or leaving some other detrimental effect, is said to be _____.

Question: 1-2
_____ testing is a common means of identifying various ferrous metals in the shop.

Question: 1-3
_____ is done by pressing or hammering steel at temperatures above the critical range to shape the metal as desired.

Question: 1-4
Steel is _____ through its critical range to transform pearlite to austenite.

Question: 1-5
When quenching heated steel, _____, _____, or _____ is used.

Question: 1-6
Name two types of case hardening.

Question: 1-7
Many structural parts, such as channels, angles, T-sections, and Z-sections, are formed by the _____ process.

Question: 1-8
Metal _____ can cause micro cracks and other precursor to failure that inspection methods try to detect, especially on older aircraft.

ANSWERS

Answer: 1-1
Malleable.

Answer: 1-2
Spark.

Answer: 1-3
Forging.

Answer: 1-4
Heated.

Answer: 1-5
Water.
Brine
Oil.

Answer: 1-6
Carburizing.
Nitriding.
Cyaniding (not used in aircraft work).

Answer: 1-7
Extrusion.

Answer: 1-8
Fatigue.



MATERIALS AND HARDWARE

AIRCRAFT MATERIALS NON-FERROUS

SUB-MODULE 02

PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY → B1

Sub-Module 02

AIRCRAFT MATERIALS — NON-FERROUS

Knowledge Requirements

6.2 - Aircraft Materials — Non-Ferrous

(a)	Characteristics, properties and identification of common non-ferrous materials used in aircraft; Heat treatment and application of non-ferrous materials;	2
(b)	Testing of non-ferrous material for hardness, tensile strength, fatigue strength and impact resistance.	1

6.2(A) - CHARACTERISTICS OF NON-FERROUS METALS

The term "non-ferrous" refers to all metals which have elements other than iron as their base or principal constituent. This group includes such metals as aluminum, titanium, copper, and magnesium, as well as such alloyed metals as Monel and babbit.

ALUMINUM/ALUMINUM ALLOYS

Commercially pure aluminum is a white lustrous metal which stands second in the scale of malleability, sixth in ductility, and ranks high in its resistance to corrosion. Aluminum combined with various percentages of other metals forms alloys which are used in aircraft construction.

Aluminum alloys in which the principal alloying ingredients are manganese, chromium, or magnesium and silicon show little attack in corrosive environments. Alloys in which substantial percentages of copper are used are more susceptible to corrosive action. The total percentage of alloying elements is seldom more than 6 or 7 percent in the wrought alloys. Aluminum is one of the most widely used metals in modern aircraft construction. It is vital to the aviation industry because of its high strength to weight ratio and its comparative ease of fabrication. The outstanding characteristic of aluminum is its light weight. Aluminum melts at the comparatively low temperature of 675°C. It is nonmagnetic and is an excellent conductor.

Commercially pure aluminum has a tensile strength of about 13 000 psi, but its strength may be approximately doubled by rolling or other cold working processes. By alloying with other metals, or by using heat-treating processes, the tensile strength may be raised to as high as 65 000 psi or to within the strength range of structural steel.

Aluminum alloys, although strong, are easily worked because they are malleable and ductile. They may be rolled into sheets as thin as 0.001 7 inch or drawn into wire 0.004 inch in diameter. Most aluminum alloy sheet stock used in aircraft construction range from 0.016 to 0.096 inch in thickness; however, some of the larger aircraft use sheet stock which may be as thick as 0.356 inch.

The various types of aluminum may be divided into two general classes: (1) casting alloys (those suitable for casting in sand, permanent mold, or die castings) and (2) wrought alloys (those which may be shaped by rolling, drawing, or forging). Of these two, the wrought alloys are the most widely used in aircraft construction, being used for stringers, bulkheads, skin, rivets, and extruded sections.

Aluminum casting alloys are divided into two basic groups. In one, the physical properties of the alloys are determined by the alloying elements and cannot be changed after the metal is cast. In the other, the alloying elements make it possible to heat treat the casting to produce the desired physical properties.

The casting alloys are identified by a letter preceding the alloy number. When a letter precedes a number, it indicates a slight variation in the composition of the original alloy. This variation in composition is simply to impart some desirable quality. In casting alloy 214, for example, the addition of zinc to improve its pouring qualities is indicated by the letter A in front of the number, thus creating the designation A214.

When castings have been heat treated, the heat treatment and the composition of the casting is indicated by the letter T, followed by an alloying number. An example of this is the sand casting alloy 355, which has several different compositions and tempers and is designated by 355-T6, 355-T51, or C355-T51.

Aluminum alloy castings are produced by one of three basic methods: (1) sand mold, (2) permanent mold, or (3) die cast. In casting aluminum, it must be remembered that in most cases different types of alloys must be used for different types of castings. Sand castings and die castings require different types of alloys than those used in permanent molds.

Sand and permanent mold castings are parts produced by pouring molten metal into a previously prepared mold, allowing the metal to solidify or freeze, and then removing the part. If the mold is made of sand, the part is a sand casting; if it is a metallic mold (usually cast iron) the part is a permanent mold casting. Sand and permanent castings are produced by pouring liquid metal into the mold, the metal flowing under the force of gravity alone.

The two principal types of sand casting alloys are 112 and 212. Little difference exists between the two metals from a mechanical properties standpoint, since both are adaptable to a wide range of products.

The permanent mold process is a later development of the sand casting process, the major difference being in the material from which the molds are made. The advantage of this process is that there are fewer openings (called porosity) than in sand castings. The sand and the binder, which is mixed with the sand to hold it together, give off a certain amount of gas which causes porosity in a sand casting.

Permanent mold castings are used to obtain higher mechanical properties, better surfaces, or more accurate dimensions. There are two specific types of permanent mold castings: (1) permanent metal mold with metal cores, and (2) semipermanent types containing sand cores. Because finer grain structure is produced in alloys subjected to the rapid cooling of metal molds, they are far superior to the sand type castings. Alloys 122, A132, and 142 are commonly used in permanent mold castings, the principal uses of which are in internal combustion engines.

Die castings used in aircraft are usually aluminum or magnesium alloy. If weight is of primary importance, magnesium alloy is used because it is lighter than aluminum alloy. However, aluminum alloy is frequently used because it is stronger than most magnesium alloys.

A die casting is produced by forcing molten metal under pressure into a metallic die and allowing it to solidify; then the die is opened and the part removed. The basic difference between permanent mold casting and die casting is that in the permanent mold process the metal flows into the die under gravity. In the die casting operation, the metal is forced under great pressure. Die castings are used where relatively large production of a given part is involved. Remember, any shape which can be forged can be cast.

WROUGHT ALUMINUM

Wrought aluminum and wrought aluminum alloys are divided into two general classes: non-heat-treatable alloys and heat-treatable alloys.

Non-heat-treatable alloys are those in which the mechanical properties are determined by the amount of cold work introduced after the final annealing operation. The mechanical properties obtained by cold working are destroyed by any subsequent heating and cannot be restored except by additional cold working, which is not always possible. The "full hard" temper is produced by the maximum amount of cold work that is commercially possible. Metal in the "as fabricated" condition is produced from the ingot without any subsequent controlled amount of cold working or thermal treatment. There is, consequently, a variable amount of strain hardening, depending upon the thickness of the section.

For heat-treatable aluminum alloys, the mechanical properties are obtained by heat treating to a suitable temperature, holding at that temperature long enough to allow the alloying constituent to enter into solid solution, and then quenching to hold the constituent in solution. The metal is left in a supersaturated, unstable state and is then age hardened either by natural aging at room temperature or by artificial aging at some elevated temperature. Wrought aluminum and wrought aluminum alloys are designated by a four digit index system. The system is broken into three distinct groups: 1xxx group, 2xxx through 8xxx group, and 9xxx group (which is currently unused).

The first digit of a designation identifies the alloy type. The second digit indicates specific alloy modifications. Should the second number be zero, it would indicate no special control over individual impurities. Digits 1 through 9, however, when assigned consecutively as needed for the second number in this group, indicate the number of controls over individual impurities in the metal.

The last two digits of the 1xxx group are used to indicate the hundredths of 1 percent above the original 99 percent designated by the first digit. Thus, if the last two digits were 30, the alloy would contain 99 percent plus 0.30 percent of pure aluminum, or a total of 99.30 percent pure aluminum. Examples of alloys in this group are:

- 1100—99.00 percent pure aluminum with one control over individual impurities.
- 1130—99.30 percent pure aluminum with one control over individual impurities.
- 1275—99.75 percent pure aluminum with two controls over individual impurities.

Percentage of Alloying Elements
(aluminum and normal impurities constitute remainder)

Alloy	Copper	Silicon	Manganese	Magnesium	Zinc	Nickel	Chromium	Lead	Bismuth
1100	—	—	—	—	—	—	—	—	—
3003	—	—	1.2	—	—	—	—	—	—
2011	5.5	—	—	—	—	—	—	0.5	0.5
2014	4.4	0.8	0.8	0.4	—	—	—	—	—
2017	4.0	—	0.5	0.5	—	—	—	—	—
2117	2.5	—	—	0.3	—	—	—	—	—
2018	4.0	—	—	0.5	—	2.0	—	—	—
2024	4.5	—	0.6	1.5	—	—	—	—	—
2025	4.5	0.8	0.8	—	—	—	—	—	—
4032	0.9	12.5	—	1.0	—	0.9	—	—	—
6151	—	1.0	—	0.6	—	—	0.25	—	—
5052	—	—	—	2.5	—	—	0.25	—	—
6053	—	0.7	—	1.3	—	—	0.25	—	—
6061	0.25	0.6	—	1.0	—	—	0.25	—	—
7075	1.6	—	—	2.5	5.6	—	0.3	—	—

Figure 2-1. Nominal composition of wrought aluminum alloys.

In the 2xxx through 8xxx groups, the first digit indicates the major alloying element used in the formation of the alloy as follows:

- 2xxx—copper
- 3xxx—manganese
- 4xxx—silicon
- 5xxx—magnesium
- 6xxx—magnesium and silicon
- 7xxx—zinc
- 8xxx—other elements

In the 2xxx through 8xxx alloy groups, the second digit in the alloy designation indicates alloy modifications. If the second digit is zero, it indicates the original alloy, while digits 1 through 9 indicate alloy modifications. The last two of the four digits in the designation identify the different alloys in the group. (***Figure 2-1***)

EFFECT OF ALLOYING ELEMENT

- 1000 series: Made of 99 percent aluminum or higher, excellent for its corrosion resistance, high thermal and electrical conductivity, low mechanical properties, and excellent workability. Iron and silicon are major impurities.
- 2000 series: Copper is the principal alloying element. Solution heat treatment, optimum properties equal to mild steel, poor corrosion resistance unclad. It is usually clad with 6000 or high purity alloy. Its best known alloy is 2024.
- 3000 series—Manganese is the principal alloying element of this group which is generally non-heat treatable. The percentage of manganese which will be alloy effective is 1.5 percent. The most popular is 3003, which is of moderate strength and has good working characteristics.
- 4000 series—Silicon is the principal alloying element of this group, and lowers melting temperature. Its primary use is in welding and brazing. When used in welding heat-treatable alloys, this group will respond to a limited amount of heat treatment.
- 5000 series—Magnesium is the principal alloying element. It has good welding and corrosion resistant characteristics. High temperatures (over 150°F) or excessive cold working will increase susceptibility to corrosion.
- 6000 series: Silicon and magnesium form magnesium silicide which makes alloys heat treatable. It is of medium strength, good forming qualities, and has corrosion resistant characteristics.
- 7000 series: Zinc is the principal alloying element. The most popular alloy of the series is 6061. When coupled with magnesium, it results in heat-treatable alloys of very high strength. It usually has copper and chromium added. The principal alloy of this group is 7075.

HARDNESS IDENTIFICATION

Where used, the temper designation follows the alloy designation and is separated from it by a dash: i.e., 7075-T6, 2024-T4, and so forth. The temper designation consists of a letter indicating the basic temper which may be more specifically defined by the addition of one or more digits. These designations are as follows:

- F - as fabricated
- O - annealed, recrystallized (wrought products only)
- H - strain hardened
- H1 (plus one or more digits) — strain hardened only
- H2 (plus one or more digits) — strain hardened and partially annealed
- H3 (plus one or more digits) — strain hardened and stabilized

The digit following the designations H1, H2, and H3 indicates the degree of strain hardening, number 8 representing the ultimate tensile strength equal to that achieved by a cold reduction of approximately 75 percent following a full anneal, 0 representing the annealed state.

MAGNESIUM/MAGNESIUM ALLOYS

Magnesium, the world's lightest structural metal, is a silvery white material weighing only two-thirds as much as aluminum. Magnesium does not possess sufficient strength in its pure state for structural uses, but when alloyed with zinc, aluminum, and manganese it produces an alloy having the highest strength to weight ratio of any of the commonly used metals. Magnesium is probably more widely distributed in nature than any other metal. It can be obtained from such ores as dolomite and magnesite, and from sea water, underground brines, and waste solutions of potash.

With about 10 million pounds of magnesium in 1 cubic mile of sea water, there is no danger of a dwindling supply. Some of today's aircraft require in excess of one-half ton of this metal for use in hundreds of vital spots. Some wing panels are fabricated entirely from magnesium alloys, weigh 18 percent less than standard aluminum panels, and have flown hundreds of satisfactory hours. Among the aircraft parts that have been made from magnesium with a substantial savings in weight are nosewheel doors, flap cover skin, aileron cover skin, oil tanks, floorings, fuselage parts, wingtips, engine nacelles, instrument panels, radio masts, hydraulic fluid tanks, oxygen bottle cases, ducts, and seats.

Magnesium alloys possess good casting characteristics. Their properties compare favorably with those of cast aluminum. In forging, hydraulic presses are ordinarily used, although, under certain conditions, forging can be accomplished in mechanical presses or by using drop hammers. Magnesium alloys are subject to such treatments as annealing, quenching, solution heat treatment, aging, and stabilizing. Sheet and plate magnesium are annealed at the rolling mill. The solution heat treatment is used to put as much of the alloying ingredients as possible into solid solution, which results in high tensile strength and maximum ductility. Aging is applied to castings following heat treatment where maximum hardness and yield strength are desired.

Magnesium embodies fire hazards of an unpredictable nature. When in large sections, its high thermal conductivity makes it difficult to ignite and prevents it from burning. It will not burn until the melting point of 651°C is reached. However, magnesium dust and fine chips are ignited easily. Precautions must be taken to avoid this if possible. Should a fire occur, it can be extinguished with an extinguishing powder, such as soapstone or graphite. Water or any standard liquid or foam fire extinguisher cause magnesium to burn more rapidly and can cause explosions.

Magnesium alloys produced in the United States consist of magnesium alloyed with varying proportions of aluminum, manganese, and zinc. These alloys are designated by a letter of the alphabet, with the number 1 indicating high purity and maximum corrosion resistance. Many of the magnesium alloys manufactured in the United States are produced by the Dow Chemical Company and have been given the trade name of Dowmetal™ alloys. To distinguish between these alloys, each is assigned a letter. Thus, there are Dowmetal J, Dowmetal M, etc.

Another manufacturer of magnesium alloys is the American Magnesium Corporation, a subsidiary of the Aluminum Company of America. This company uses an identification system similar to that used for aluminum alloys, with the exception that magnesium alloy numbers are preceded with the letters AM. Thus, AM240C is a cast alloy, and AM240C4 is the same alloy in the heat-treated state. AM3S0 is an annealed wrought alloy, and AM3SRT is the same alloy rolled after heat treatment.

TITANIUM/TITANIUM ALLOYS

Titanium was discovered by an English priest named Gregot. A crude separation of titanium ore was accomplished in 1825. In 1906 a sufficient amount of pure titanium was isolated in metallic form to permit a study. Following this study, in 1932, an extraction process was developed which became the first commercial method for producing titanium. The United States Bureau of Mines began making titanium sponge in 1946, and 4 years later the melting process began.

The use of titanium is widespread. It is used in many commercial enterprises and is in constant demand for such items as pumps, screens, and other tools and fixtures where corrosion attack is prevalent. In aircraft construction and repair, titanium is used for fuselage skins, engine shrouds, firewalls, longerons, frames, fittings, air ducts, and fasteners. Titanium is used for making compressor disks, spacer rings, compressor blades and vanes, through bolts, turbine housings and liners, and miscellaneous hardware for turbine engines.

Titanium, in appearance, is similar to stainless steel. One quick method used to identify titanium is the spark test. Titanium gives off a brilliant white trace ending in a brilliant white burst. Also, identification can be accomplished by moistening the titanium and using it to draw a line on a piece of glass. This will leave a dark line similar in appearance to a pencil mark.

Titanium falls between aluminum and stainless steel in terms of elasticity, density, and elevated temperature strength. It has a melting point of from 1 500°C to 1 735°C, low thermal conductivity, and a low coefficient of expansion. It is light, strong, and resistant to stress corrosion cracking.

Titanium is approximately 60 percent heavier than aluminum and about 50 percent lighter than stainless steel. Because of the high melting point of titanium, high temperature properties are disappointing. The ultimate yield strength of titanium drops rapidly above 425°C. The absorption of oxygen and nitrogen from the air at temperatures above 540°C makes the metal so brittle on long exposure that it soon becomes worthless. However, titanium does have some merit for short time exposure up to 1 650°C where strength is not important. Aircraft firewalls demand this requirement.

Titanium is nonmagnetic and has an electrical resistance comparable to that of stainless steel. Some of the base alloys of titanium are quite hard. Heat treating and alloying do not develop the hardness of titanium to the high levels of some of the heat-treated alloys of steel. It was only recently that a heat-treatable titanium alloy was developed. Prior to the development of this alloy, heating and rolling was the only method of forming that could be accomplished. However, it is possible to form the new alloy in the soft condition and heat treat it for hardness.

Iron, molybdenum, and chromium are used to stabilize titanium and produce alloys that will quench harden and age harden. The addition of these metals also adds ductility. The fatigue resistance of titanium is greater than that of aluminum or steel.

Titanium becomes softer as the degree of purity is increased. It is not practical to distinguish between the various grades of commercially pure or unalloyed titanium by chemical analysis; therefore, the grades are determined by mechanical properties.

TITANIUM DESIGNATIONS

The A-B-C classification of titanium alloys was established to provide a convenient and simple means of describing all titanium alloys. Titanium and titanium alloys possess three basic types of crystals: A (alpha), B (beta), and C (combined alpha and beta). Their characteristics are:

- A (alpha) — all around performance; good weldability; tough and strong both cold and hot, and resistant to oxidation.
- B (beta) — bendability; excellent bend ductility; strong both cold and hot, but vulnerable to contamination.
- C (combined alpha and beta for compromise performances) — strong when cold and warm, but weak when hot; good bendability; moderate contamination resistance; excellent forgeability.

Titanium is manufactured for commercial use in two basic compositions: commercially pure titanium and alloyed titanium. A-55 is an example of a commercially pure titanium. It has a yield strength of 55 000 to 80 000 psi and is a general purpose grade for moderate to severe forming. It is sometimes used for nonstructural aircraft parts and for all types of corrosion resistant applications, such as tubing.

Type A-70 titanium is closely related to type A-55 but has a yield strength of 70 000 to 95 000 psi. It is used where higher strength is required, and it is specified for many moderately stressed aircraft parts. For many corrosion applications, it is used interchangeably with type A-55. Both type A-55 and type A-70 are weldable.

One of the widely used titanium base alloys is designated as C-110M. It is used for primary structural members and aircraft skin, has 110 000 psi minimum yield strength, and contains 8 percent manganese.

Type A-110AT is a titanium alloy which contains 5 percent aluminum and 2.5 percent tin. It also has a high minimum yield strength at elevated temperatures with the excellent welding characteristics inherent in alpha-type titanium alloys.

CORROSION CHARACTERISTICS OF TITANIUM

The corrosion resistance of titanium deserves special mention. The resistance of the metal to corrosion is caused by the formation of a protective surface film of stable oxide or chemi-absorbed oxygen. Film is often produced by the presence of oxygen and oxidizing agents. Corrosion of titanium is uniform. There is little evidence of pitting or other serious forms of localized attack. Normally, it's not subject to stress corrosion, corrosion fatigue, intergranular corrosion, or galvanic corrosion. Its corrosion resistance is equal or superior to 18-8 stainless steel.

Laboratory tests with acid and saline solutions show titanium polarizes readily. The net effect, in general, is to decrease current flow in galvanic and corrosion cells. Corrosion currents on the surface of titanium and metallic couples are naturally restricted. This partly accounts for good resistance to many chemicals; also, the material may be used with some dissimilar metals with no harmful galvanic effect on either.

COPPER/COPPER ALLOYS

Copper is one of the most widely distributed metals. It is the only reddish colored metal and is second only to silver in electrical conductivity. Its use as a structural material is limited because of its great weight. However, some of its outstanding characteristics, such as its high electrical and heat conductivity, in many cases overbalance the weight factor.

Because it is very malleable and ductile, copper is ideal for making wire. It is corroded by salt water but is not affected by fresh water. The ultimate tensile strength of copper varies greatly. For cast copper, the tensile strength is about 25 000 psi, and when cold rolled or cold drawn its tensile strength increases to a range of 40 000 to 67 000 psi. In aircraft, copper is used primarily in the electrical system for bus bars, bonding, and as lockwire.

Beryllium copper is one of the most successful of all the copper base alloys. It is a recently developed alloy containing about 97 percent copper, 2 percent beryllium, and sufficient nickel to increase the percentage of elongation. The most valuable feature of this metal is that the physical properties can be greatly stepped up by heat treatment, the tensile strength rising from 70 000 psi in the annealed state to 200 000 psi in the heat-treated state. The resistance of beryllium copper to fatigue and wear makes it suitable for diaphragms, precision bearings and bushings, ball cages, and spring washers.

Brass is a copper alloy containing zinc and small amounts of aluminum, iron, lead, manganese, magnesium, nickel, phosphorous, and tin. Brass with a zinc content of 30 to 35 percent is very ductile, but that containing 45 percent has relatively high strength.

Muntz metal is a brass composed of 60 percent copper and 40 percent zinc. It has excellent corrosion resistant qualities in salt water. Its strength can be increased by heat treatment. As cast, this metal has an ultimate tensile strength of 50 000 psi, and it can be elongated 18 percent. It is used in making bolts and nuts, as well as parts that come in contact with salt water. Red brass, sometimes termed "bronze" because of its tin content, is used in fuel and oil line fittings. This metal has good casting and finishing properties and machines freely.

Bronzes are copper alloys containing tin. The true bronzes have up to 25 percent tin, but those with less than 11 percent are most useful, especially for such items as tube fittings in aircraft.

Among the copper alloys are the copper aluminum alloys, of which the aluminum bronzes rank very high in aircraft usage. They would find greater usefulness in structures if it were not for their strength to weight ratio as compared with alloy steels. Wrought aluminum bronzes are almost as strong and ductile as medium carbon steel,

and they possess a high degree of resistance to corrosion by air, salt water, and chemicals. They are readily forged, hot or cold rolled, and many react to heat treatment.

These copper base alloys contain up to 16 percent of aluminum (usually 5 to 11 percent), to which other metals, such as iron, nickel, or manganese, may be added. Aluminum bronzes have good tearing qualities, great strength, hardness, and resistance to both shock and fatigue. Because of these properties, they are used for diaphragms, gears, and pumps. Aluminum bronzes are available in rods, bars, plates, sheets, strips, and forgings.

Cast aluminum bronzes, using about 89 percent copper, 9 percent aluminum, and 2 percent of other elements, have high strength combined with ductility, and are resistant to corrosion, shock, and fatigue. Because of these properties, cast aluminum bronze is used in bearings and pump parts. These alloys are useful in areas exposed to salt water and corrosive gases.

Manganese bronze is an exceptionally high strength, tough, corrosion resistant copper zinc alloy containing aluminum, manganese, iron and, occasionally, nickel or tin. This metal can be formed, extruded, drawn, or rolled to any desired shape. In rod form, it is generally used for machined parts, for aircraft landing gears and brackets.

Silicon bronze is a more recent development composed of about 95 percent copper, 3 percent silicon, and 2 percent manganese, zinc, iron, tin, and aluminum. Although not a bronze in the true sense because of its small tin content, silicon bronze has high strength and great corrosion resistance.

NICKEL/NICKEL ALLOYS

There are basically two nickel alloys used in aircraft. They are Monel and Inconel. Monel contains about 68 percent nickel and 29 percent copper, plus small amounts of iron and manganese. Nickel alloys can be welded or easily machined. Some of the nickel Monel, especially the nickel Monels containing small amounts of aluminum, are heat-treatable to similar tensile strengths of steel. Nickel Monel is used in gears and parts that require high strength and toughness, such as exhaust systems that require high strength and corrosion resistance at elevated temperatures.

MONEL

Monel, the leading high nickel alloy, combines the properties of high strength and excellent corrosion resistance. This metal consists of 68 percent nickel, 29 percent copper, 0.2 percent iron, 1 percent manganese, and 1.8 percent of other elements. It cannot be hardened by heat treatment. Monel, adaptable to casting and hot or cold working, can be successfully welded. It has working properties similar to those of steel. When forged and annealed, it has a tensile strength of 80 000 psi. This can be increased by cold working to 125 000 psi, sufficient for classification among the tough alloys.

Monel has been successfully used for gears and chains to operate retractable landing gears, and for structural parts subject to corrosion. In aircraft, Monel is used for parts demanding both strength and high resistance to corrosion, such as exhaust manifolds and carburetor needle valves and sleeves.

K-MONEL

K-Monel is a nonferrous alloy containing mainly nickel, copper, and aluminum. It is produced by adding a small amount of aluminum to the Monel formula. It is corrosion resistant and capable of being hardened by heat treatment. K-Monel has been successfully used for gears, and structural members in aircraft which are subjected to corrosive attacks. This alloy is nonmagnetic at all temperatures. K-Monel sheet has been successfully welded by both oxyacetylene and electric arc welding.

INCONEL

Inconel alloys of nickel produce a high strength, high temperature alloy containing approximately 80 percent nickel, 14 percent chromium, and small amounts of iron and other elements. The nickel Inconel alloys are frequently used in turbine engines because of their ability to maintain their strength and corrosion resistance under extremely high temperature conditions.

Inconel and stainless steel are similar in appearance and are frequently found in the same areas of the engine. Sometimes it is important to identify the difference between the metal samples. A common test is to apply one drop of cupric chloride and hydrochloric acid solution to the unknown metal and allow it to remain for 2 minutes. At the end of the soak period, a shiny spot indicates the material is nickel Inconel, and a copper colored spot indicates stainless steel.

SUBSTITUTION OF AIRCRAFT METALS

In selecting substitute metals for the repair and maintenance of aircraft, it is very important to check the appropriate structural repair manual. Aircraft manufacturer's design structural members to meet a specific load requirement for a particular aircraft. The methods of repairing these members, apparently similar in construction, will thus vary with different aircraft.

Four requirements must be kept in mind when selecting substitute metals. The first and most important of these is maintaining the original strength of the structure. The other three are: (1) maintaining contour or aerodynamic smoothness, (2) maintaining original weight, or keeping added weight to a minimum, and (3) maintaining the original corrosion resistant properties of the metal.

HEAT TREATMENT OF NON-FERROUS ALLOYS

ALUMINUM ALLOYS

In the wrought form, commercially pure aluminum is known as 1100. It has a high degree of resistance to corrosion and is easily formed into intricate shapes. It is relatively low in strength and does not have the properties required for structural aircraft parts. High strengths are generally obtained by the process of alloying. The resulting alloys are less easily formed and, with some exceptions, have lower resistance to corrosion than 1100 aluminum.

Alloying is not the only method of increasing the strength of aluminum. Like other materials, aluminum becomes stronger and harder as it is rolled, formed, or otherwise cold worked. Since the hardness depends on the amount of cold working done, 1100 and some wrought aluminum alloys are available in several strain hardened tempers. The soft or annealed condition is designated O. If the material is strain hardened, it is said to be in the H condition.

The most widely used alloys in aircraft construction are hardened by heat treatment rather than by cold work. These alloys are designated by a somewhat different set of symbols: T4 and W indicate solution heat treated and quenched but not aged, and T6 indicates an alloy in the heat treated hardened condition.

- W - Solution heat treated, unstable temper
- T - Treated to produce stable tempers other than F, O, or H
- T2 - Annealed (cast products only)
- T3 - Solution heat treated and then cold worked
- T4 - Solution heat treated
- T5 - Artificially aged only
- T6 - Solution heat treated and then artificially aged
- T7 - Solution heat treated and then stabilized
- T8 - Solution heat treated, cold worked, and then artificially aged
- T9 - Solution heat treated, artificially aged, and then cold worked
- T10 - Artificially aged and then cold worked

Additional digits may be added to T1 through T10 to indicate a variation in treatment which significantly alters the characteristics of the product.

Aluminum alloy sheets are marked with the specification number on approximately every square foot of material. If for any reason this identification is not on the material, it is possible to separate the heat treatable alloys from the non-heat-treatable alloys by immersing a sample of the material in a 10 percent solution of caustic soda (sodium hydroxide). The heat-treatable alloys will turn black due to the copper content, whereas the others will remain bright. In the case of clad material, the surface will remain bright, but there will be a dark area in the middle when viewed from the edge.

ALCLAD ALUMINUM

The terms "Alclad and Pureclad" are used to designate sheets that consist of an aluminum alloy core coated with a layer of pure aluminum to a depth of approximately 5½ percent on each side. The pure aluminum coating affords a dual protection for the core, preventing contact with any corrosive agents, and protecting the core electrolytically by preventing any attack caused by scratching or from other abrasions.

There are two types of heat treatments applicable to aluminum alloys. One is called solution heat treatment, and the other is known as precipitation heat treatment. Some alloys, such as 2017 and 2024, develop their full properties as a result of solution heat treatment followed by about 4 days of aging at room temperature. Other alloys, such as 2014 and 7075, require both heat treatments.

The alloys that require precipitation heat treatment (artificial aging) to develop their full strength also age to a limited extent at room temperature; the rate and amount of strengthening depends upon the alloy. Some reach their maximum natural or room temperature aging strength in a few days, and are designated as -T4 or -T3 temper. Others continue to age appreciably over a long period of time.

Because of this natural aging, the -W designation is specified only when the period of aging is indicated, for example, 7075-W (½ hour). Thus, there is considerable difference in the mechanical and physical properties of freshly quenched (-W) material and material that is in the -T3 or -T4 temper.

The hardening of an aluminum alloy by heat treatment consists of four distinct steps:

1. Heating to a predetermined temperature.
2. Soaking at temperature for a specified length of time.
3. Rapidly quenching to a relatively low temperature.
4. Aging or precipitation hardening either spontaneously at room temperature, or as a result of a low temperature thermal treatment.

The first three steps above are known as solution heat treatment, although it has become common practice to use the shorter term, "heat treatment". Room temperature hardening is known as natural aging, while hardening done at moderate temperatures is called artificial aging, or precipitation heat treatment.

SOLUTION HEAT TREATMENT

Temperature

The temperatures used for solution heat treating vary with different alloys and range from 440°C to 525°C. As a rule, they must be controlled within a very narrow range (±5.5°C) to obtain specified properties.

If the temperature is too low, maximum strength will not be obtained. When excessive temperatures are used, there is danger of melting the low melting constituents of some alloys with consequent lowering of the physical properties of the alloy. Even if melting does not occur, the use of higher than recommended temperatures promotes discoloration and increases quenching strains.

Time at Temperature

The time at temperature, referred to as soaking time, is measured from the time the coldest metal reaches the minimum limit of the desired temperature range. The soaking time varies, depending upon the alloy and thickness, from 10 minutes for thin sheets to approximately 12 hours for heavy forgings. For the heavy sections, the nominal soaking time is approximately 1 hour for each inch of cross-sectional thickness. (***Figure 2-2***)

Choose the minimum soaking time necessary to develop the required physical properties. The effect of an abbreviated soaking time is obvious. An excessive soaking period aggravates high temperature oxidation. With clad material, prolonged heating results in excessive diffusion of copper and other soluble constituents into the protective cladding and may defeat the purpose of cladding.

QUENCHING

After the soluble constituents are in solid solution, the material is quenched to prevent or retard immediate reprecipitation. Three distinct quenching methods are employed. The one to be used in any particular instance depends upon the part, the alloy, and the properties desired.

Cold Water Quenching

Parts produced from sheet, extrusions, tubing, small forgings, and similar type material are generally quenched in a cold water bath. The temperature of the water before quenching should not exceed 30°C. Using a sufficient quantity of water keeps the temperature rise under 11°C. Such a drastic quench ensures maximum resistance to corrosion. This is particularly important when working with such alloys as 2017, 2024, and 7075. This is the reason a drastic quench is preferred, even though a slower quench may produce the required mechanical properties.

Thickness (inch)	Time (minutes)
Up to .032	30
.032 to ⅛	30
⅛ to ¼	40
Over ¼	60

Note: Soaking time starts when the metal (or the molten bath) reaches a temperature within the range specified above.

Figure 2-2. Typical soaking times for heat treatment.

Hot Water Quenching

Large forgings and heavy sections can be quenched in hot or boiling water. This type of quench minimizes distortion and alleviates cracking which may be produced by the unequal temperatures obtained during the quench. The use of a hot water quench is permitted with these parts because the temperature of the quench water does not critically affect the resistance to corrosion of the forging alloys. In addition, the resistance to corrosion of heavy sections is not as critical a factor as for thin sections.

Spray Quenching

High velocity water sprays are useful for parts formed from clad sheet and for large sections of almost all alloys. This type of quench also minimizes distortion and alleviates quench cracking. However, many specifications forbid the use of spray quenching for bare 2017 and 2024 sheet materials because of the effect on their resistance to corrosion.

Lag Between Soaking and Quenching

The time interval between the removal of the material from the furnace and quenching is critical for some alloys and should be held to a minimum. When solution heat treating 2017 or 2024 sheet material, the elapsed time must not exceed 10 seconds. The allowable time for heavy sections may be slightly greater.

Allowing the metal to cool slightly before quenching promotes reprecipitation from the solid solution. The precipitation occurs along grain boundaries and in certain slip planes causing poorer formability. In the case of 2017, 2024, and 7075 alloys, their resistance to intergranular corrosion is adversely affected.

REHEAT TREATMENT

The treatment of material which has been previously heat treated is considered a reheat treatment. The unclad heat-treatable alloys can be solution heat treated repeatedly without harmful effects.

The number of solution heat treatments allowed for clad sheet is limited due to increased diffusion of core and cladding with each reheating. Existing specifications allow one to three reheat treatments of clad sheet depending upon cladding thickness.

STRAIGHTENING AFTER SOLUTION HEAT TREATMENT

Some warping occurs during solution heat treatment, producing kinks, buckles, waves, and twists. These imperfections are generally removed by straightening and flattening operations. Where the straightening operations produce an appreciable increase in the tensile and yield strengths and a slight decrease in the percent of elongation, the material is designated -T3 temper. When the above values are not materially affected, the material is designated -T4 temper.

PRECIPITATION HEAT TREATING

As previously stated, the aluminum alloys are in a comparatively soft state immediately after quenching from a solution heat-treating temperature. To obtain their maximum strengths, they must be either naturally aged or precipitation hardened.

During this hardening and strengthening operation, precipitation of the soluble constituents from the supersaturated solid solution takes place. As precipitation progresses, the strength of the material increases, often by a series of peaks, until a maximum is reached. Further aging (over aging) causes the strength to steadily decline until a somewhat stable condition is obtained. The submicroscopic particles that are precipitated provide the keys or locks within the grain structure and between the grains to resist internal slippage and distortion when a load of any type is applied. In this manner, the strength and hardness of the alloy are increased.

Precipitation hardening produces a great increase in the strength and hardness of the material with corresponding decreases in the ductile properties. The process used to obtain the desired increase in strength is therefore known as aging, or precipitation hardening.

The strengthening of the heat treatable alloys by aging is not due merely to the presence of a precipitate. The strength is due to both the uniform distribution of a finely dispersed submicroscopic precipitate and its effects upon the crystal structure of the alloy. The aging practices used depend upon many properties other than strength. As a rule, the artificially aged alloys are slightly over aged to increase their resistance to corrosion. This is especially true with the artificially aged high copper content alloys that are susceptible to intergranular corrosion when inadequately aged.

Alloy	SOLUTION HEAT TREATMENT			PRECIPITATION HEAT TREATMENT		
	Temperature (°F)	Quench	Temperature Designation	Temperature (°F)	Time of Aging	Temperature Designation
2017	930–950	Cold Water	T4			T
2117	930–950	Cold Water	T4			T
2024	910–930	Cold Water	T4			T
6053	960–980	Water	T4	445–455	1–2 hours	T5
					or	
				345–355	8 hours	T6
6061	960–980	Water	T4	315–325	18 hours	T6
					or	
				345–355	8 hours	T6
7075	870	Water		250	24 hours	T6

Figure 2-3. Conditions for heat treatment of aluminum alloys.

The heat treatable aluminum alloys are subdivided into two classes: those that obtain their full strength at room temperature and those that require artificial aging. The alloys that obtain their full strength after 4 or 5 days at room temperature are known as natural aging alloys. Precipitation from the super saturated solid solution starts soon after quenching, with 90 percent of the maximum strength generally being obtained in 24 hours. Alloys 2017 and 2024 are natural aging alloys.

The alloys that require precipitation thermal treatment to develop their full strength are artificially aged alloys. However, these alloys also age a limited amount at room temperature, the rate and extent of the strengthening depending upon the alloys.

Many of the artificially aged alloys reach their maximum natural or room temperature aging strengths after a few days. These can be stocked for fabrication in the -T4 or -T3 temper. High zinc content alloys such as 7075 continue to age appreciably over a long period of time, their mechanical property changes being sufficient to reduce their formability.

The advantage of -W temper formability can be utilized, however, in the same manner as with natural aging alloys; that is, by fabricating shortly after solution heat treatment, or retaining formability by the use of refrigeration. Refrigeration retards the rate of natural aging. At 0°C, the beginning of the aging process is delayed for several hours, while dry ice (–45°C to –73°C) retards aging for an extended period of time.

Precipitation Practices

The temperatures used for precipitation hardening depend upon the alloy and the properties desired, ranging from 125°C to 190°C. They should be controlled within a very narrow range (±3°C) to obtain best results. (***Figure 2-3***)

The time at temperature is dependent upon the temperature used, the properties desired, and the alloy. It ranges from 8 to 96 hours. Increasing the aging temperature decreases the soaking period necessary for proper aging. However, a closer control of both time and temperature is necessary when using the higher temperatures. After receiving the thermal precipitation treatment, the material should be air cooled to room temperature. Water quenching, while not necessary, produces no ill effects. Furnace cooling has a tendency to produce over aging.

ANNEALING OF ALUMINUM ALLOYS

The annealing procedure for aluminum alloys consists of heating the alloys to an elevated temperature, holding or soaking them at this temperature for a length of time depending upon the mass of the metal, and then cooling in still air. Annealing leaves the metal in the best condition for cold working. However, when prolonged forming operations are involved, the metal will take on a condition known as "mechanical hardness" and will resist further working.

It may be necessary to anneal a part several times during the forming process to avoid cracking. Aluminum alloys should not be used in the annealed state for parts or fittings. Clad parts should be heated as quickly and carefully as possible, since long exposure to heat tends to cause some of the constituents of the core to diffuse into the cladding. This reduces the corrosion resistance of the cladding.

HEAT TREATMENT OF ALUMINUM RIVETS

Aluminum alloy rivets are furnished in the following compositions: Alloys 1100, 5056, 2117, 2017, and 2024.

Alloy 1100 rivets are used in the "as fabricated" condition for riveting aluminum alloy sheets where a low strength rivet is suitable. Alloy 5056 rivets are used in the "as fabricated" condition for riveting magnesium alloy sheets.

Alloy 2117 rivets have moderately high strength and are suitable for riveting aluminum alloy sheets. These rivets receive only one heat treatment, which is performed by the manufacturer, and are anodized after being heat treated. They require no further heat treatment before they are used. Alloy 2117 rivets retain their characteristics indefinitely after heat treatment and can be driven anytime. Rivets made of this alloy are the most widely used in aircraft construction.

Alloy 2017 and 2024 rivets are high strength rivets suitable for use with aluminum alloy structures. They are purchased from the manufacturer in the heat-treated condition. Since the aging characteristics of these alloys at room temperatures are such that the rivets are unfit for driving, they must be reheat treated just before they are to be used. Alloy 2017 rivets become too hard for driving in approximately 1 hour after quenching.

Alloy 2024 rivets become hardened in 10 minutes after quenching. Both of these alloys may be reheat treated as often as required; however, they must be anodized before the first reheat treatment to prevent intergranular oxidation of the material. If these rivets are stored in a refrigerator at a temperature lower than 0°C immediately after quenching, they will remain soft enough to be usable for several days.

Rivets requiring heat treatment are heated either in tubular containers in a salt bath, or in small screen wire baskets in an air furnace. The heat treatment of alloy 2017 rivets consists of subjecting the rivets to a temperature between 500°C to 510°C for approximately 30 minutes, and immediately quenching in cold water. These rivets reach maximum strength in about 9 days after being driven.

Alloy 2024 rivets should be heated to a temperature of 490°C to 500°C and immediately quenched in cold water. These rivets develop a greater shear strength than 2017 rivets and are used in locations where extra strength is required. Alloy 2024 rivets develop their maximum shear strength in 1 day after being driven.

The 2017 rivet should be driven within approximately 1 hour and the 2024 rivet within 10 to 20 minutes after heat treating or removal from refrigeration. If not used within these times, the rivets should be reheat treated before being refrigerated.

MAGNESIUM ALLOYS

Magnesium alloy castings respond readily to heat treatment, and about 95 percent of the magnesium used in aircraft construction is in the cast form. The heat treatment of magnesium alloy castings is similar to the heat treatment of aluminum alloys in that there are two types of heat treatment: (1) solution heat treatment and (2) precipitation (aging) heat treatment. Magnesium, however, develops a negligible change in its properties when allowed to age naturally at room temperatures.

SOLUTION HEAT TREATMENT

Magnesium alloy castings are solution heat treated to improve tensile strength, ductility, and shock resistance. This heat-treatment condition is indicated by using the symbol -T4 following the alloy designation. Solution heat treatment plus artificial aging is designated -T6. Artificial aging is necessary to develop the full properties of the metal.

Solution heat-treatment temperatures for magnesium alloy castings range from 390°C to 415°C, the exact range depending upon the type of alloy. The temperature range for each type of alloy is listed in Specification MIL-H-6857. The upper limit of each range listed in the specification is the maximum temperature to which the alloy may be heated without danger of melting the metal.

The soaking time ranges from 10 to 18 hours, the exact time depending upon the type of alloy as well as the thickness of the part. Soaking periods longer than 18 hours may be necessary for castings over 2 inches in thickness. NEVER heat magnesium alloys in a salt bath as this may result in an explosion.

A serious potential fire hazard exists in the heat treatment of magnesium alloys. If through oversight or malfunctioning of equipment, the maximum temperatures are exceeded, the casting may ignite and burn freely. For this reason, the furnace used should be equipped with a safety cutoff that will turn off the power to the heating elements and blowers if the regular control equipment malfunctions or fails. Some magnesium alloys require a protective atmosphere of sulfur dioxide gas during solution heat treatment. This aids in preventing the start of a fire even if the temperature limits are slightly exceeded. Air quenching is used after solution heat treatment of magnesium alloys since there appears to be no advantage in liquid cooling.

PRECIPITATION HEAT TREATMENT

After solution treatment, magnesium alloys may be given an aging treatment to increase hardness and yield strength. Generally, the aging treatments are used merely to relieve stress and stabilize the alloys in order to prevent dimensional changes later, especially during or after machining. Both yield strength and hardness are improved somewhat by this treatment at the expense of a slight amount of ductility. The corrosion resistance is also improved, making it closer to the "as cast" alloy. Precipitation heat treatment temperatures are considerably lower than solution heat-treatment temperatures and range from 165°C to 260°C. Soaking time ranges from 4 to 18 hours.

TITANIUM

Titanium is heat treated for the following purposes:

- Relief of stresses set up during cold forming or machining.
- Annealing after hot working or cold working, or to provide maximum ductility for subsequent cold working.
- Thermal hardening to improve strength.

STRESS RELIEVING

Stress relieving is generally used to remove stress concentrations resulting from forming of titanium sheet. It is performed at temperatures ranging from 345°C to 540°C. The time at temperature varies from a few minutes for a very thin sheet to an hour or more for heavier sections. A typical stress relieving treatment is 480°C for 30 minutes, followed by an air cool.

The discoloration or scale which forms on the surface of the metal during stress relieving is easily removed by pickling in acid solutions. The recommended solution contains 10 to 20 percent nitric acid and 1 to 3 percent hydrofluoric acid. The solution should be at room temperature or slightly above.

ANNEALING OF TITANIUM

The annealing of titanium and titanium alloys provides toughness, ductility at room temperature, dimensional and structural stability at elevated temperatures, and improved machinability.

The full anneal is usually called for as preparation for further working. It is performed at 650–900°C. The time at temperature varies from 16 minutes to several hours, depending on the thickness of the material and the amount of cold work to be performed. The usual treatment for the commonly used alloys is 705°C for 1 hour, followed by an air cool. A full anneal generally results in sufficient scale formation to require the use of caustic descaling, such as sodium hydride salt bath.

THERMAL HARDENING

Unalloyed titanium cannot be heat treated, but the alloys commonly used in aircraft construction can be strengthened by thermal treatment, usually at some sacrifice in ductility. For best results, a water quench from 790°C, followed by reheating to 480°C for 8 hours is recommended.

CASE HARDENING

The chemical activity of titanium and its rapid absorption of oxygen, nitrogen, and carbon at relatively low temperatures make case hardening advantageous for special applications. Nitriding, carburizing, or carbonitriding can be used to produce a wear-resistant case of 0.000 1 to 0.000 2 inch in depth.

CASTING

Casting is formed by melting the metal and pouring it into a mold of the desired shape. Since plastic deformation of the metal does not occur, no alteration of the grain shape or orientation is possible. The grain size of the metal can be controlled by the cooling rate, the alloys of the metal, and the thermal treatment. Castings are normally lower in strength and are more brittle than a wrought product of the same material. For intricate shapes or items with internal passages, such as turbine

blades, casting may be the most economical process. Except for engine parts, most metal components found on an aircraft are wrought instead of cast.

All metal products start in the form of casting. Wrought metals are converted from cast ingots by plastic deformation. For high strength aluminum alloys, an 80 to 90 percent reduction (dimensional change in thickness) of the material is required to obtain the high mechanical properties of a fully wrought structure. Both iron and aluminum alloys are cast for aircraft uses. Cast iron contains 6 to 8 percent carbon and silicon.

Cast iron is a hard unmalleable pig iron made by casting or pouring into a mold. Cast aluminum alloy has been heated to its molten state and poured into a mold to give it the desired shape.

EXTRUDING

The extrusion process involves the forcing of metal through an opening in a die, thus causing the metal to take the shape of the die opening. The shape of the die will be the cross section of an angle, channel, tube, or some other shape. Some metals such as lead, tin, and aluminum may be extruded cold; however, most metals are heated before extrusion. The main advantage of the extrusion process is its flexibility. For example, because of its workability, aluminum can be economically extruded to more intricate shapes and larger sizes than is practical with other metals.

Extruded shapes are produced in very simple as well as extremely complex sections. In this process a cylinder of aluminum, for instance, is heated to 400–455°C and is then forced through the opening of a die by a hydraulic ram. The opening is the shape desired for the cross section of the finished extrusion.

Many structural parts, such as channels, angles, T-sections, and Z-sections, are formed by the extrusion process. Aluminum is the most extruded metal used in aircraft. Aluminum is extruded at a temperature of 371–482°C and requires pressure of up to 80 000 psi (552 MPa).

After extrusion, the product frequently will be subjected to both thermal and mechanical processes to obtain the desired properties. Extrusion processes are limited to the more ductile materials.

COLD WORKING/HARDENING

Cold working applies to mechanical working performed at temperatures below the critical range. It results in a strain hardening of the metal. In fact, the metal often becomes so hard that it is difficult to continue the forming process without softening the metal by annealing.

Since the errors attending shrinkage are eliminated in cold working, a much more compact and better metal is obtained. The strength and hardness, as well as the elastic limit, are increased; but the ductility decreases. Since this makes the metal more brittle, it must be heated from time to time during certain operations to remove the undesirable effects of the working.

While there are several cold working processes, the two with which the aviation mechanic will be principally concerned are cold rolling and cold drawing. These processes give the metals desirable qualities which cannot be obtained by hot working.

Cold rolling usually refers to the working of metal at room temperature. In this operation, the materials that have been rolled to approximate sizes are pickled to remove the scale, after which they are passed through chilled finishing rolls. This gives a smooth surface and also brings the pieces to accurate dimensions. The principal forms of cold rolled stocks are sheets, bars, and rods.

Cold drawing is used in making seamless tubing, wire, streamlined tie rods, and other forms of stock. Wire is made from hot rolled rods of various diameters. These rods are pickled in acid to remove scale, dipped in lime water, and then dried in a steam room where they remain until ready for drawing. The lime coating adhering to the metal serves as a lubricant during the drawing operation.

The size of the rod used for drawing depends upon the diameter wanted in the finished wire. To reduce the rod to the desired size, it is drawn cold through a die. One end of the rod is filed or hammered to a point and slipped through the die opening. Here it is gripped by the jaws of the drawing block and pulled through the die. This series of operations is done by a mechanism known as a drawbench.

To reduce the rod gradually to the desired size, it is necessary to draw the wire through successively smaller dies. Because each of these drawings reduces the ductility of the wire, it must be annealed from time to time before further drawings can be accomplished. Although cold working reduces the ductility, it increases the tensile strength of the wire.

In making seamless steel aircraft tubing, the tubing is cold drawn through a ring shaped die with a mandrel or metal bar inside the tubing to support it while the drawing operations are being performed. This forces the metal to flow between the die and the mandrel and affords a means of controlling the wall thickness and the inside and outside diameters.

6.2(B) - TESTING OF NON-FERROUS METALS

HARDNESS TESTING

Hardness testing is a method of determining the results of heat treatment as well as the state of a metal prior to heat treatment. Since hardness values can be tied in with tensile strength values and, in part, with wear resistance, hardness tests are a valuable check of heat treat control and of material properties. Practically all hardness testing equipment now uses the resistance to penetration as a measure of hardness. Included among the better known hardness testers are the Brinell and Rockwell, both of which are described and illustrated in this section. Also included is a popular portable-type hardness tester currently in use.

BRINELL TESTER

The Brinell hardness tester (***Figure 1-4***) uses a hardened spherical ball, which is forced into the surface of the metal. This ball is 10 millimeters (0.393 7 inch) in diameter. A pressure of 3 000 kilograms is used for ferrous metals and 500 kilograms for nonferrous metals. The pressure must be maintained at least 10 seconds for ferrous metals and at least 30 seconds for nonferrous metals. The load is applied by hydraulic pressure. The hydraulic pressure is built up by a hand pump or an electric motor, depending on the model of tester. A pressure gauge indicates the amount of pressure. There is a release mechanism for relieving the pressure after the test has been made, and a calibrated microscope is provided for measuring the diameter of the impression in millimeters. The machine has various shaped anvils for supporting the specimen and an elevating screw for bringing the specimen in contact with the ball penetrator. These are attachments for special tests.

To determine the Brinell hardness number for a metal, measure the diameter of the impression, using the calibrated microscope furnished with the tester. Then convert the measurement into the Brinell hardness number on the conversion table furnished with the tester.

ROCKWELL TESTER

The Rockwell hardness tester (***Figure 1-5***) measures the resistance to penetration, as does the Brinell tester. Instead of measuring the diameter of the impression, the Rockwell tester measures the depth, and the hardness is indicated directly on a dial attached to the machine. The dial numbers in the outer circle are black, and the inner numbers are red. Rockwell hardness numbers are based on the difference between the depth of penetration at major and minor loads. The greater this difference, the lower the hardness number and the softer the material.

Two types of penetrators are used with the Rockwell tester: a diamond cone and a hardened steel ball. The load which forces the penetrator into the metal is called the major load and is measured in kilograms. The results of each penetrator and load combination are reported on separate scales, designated by letters.

The penetrator, the major load, and the scale vary with the kind of metal being tested. For hardened steels, the diamond penetrator is used; the major load is 150 kilograms; and the hardness is read on the "C" scale. When this reading is recorded, the letter "C" must precede the number indicated by the pointer. The C-scale setup is used for testing metals ranging in hardness from C-20 to the hardest steel (usually about C-70). If the metal is softer than C-20, the B-scale setup is used. With this setup, the 1/16-inch ball is used as a penetrator; the major load is 100 kilograms; and the hardness is read on the B-scale. In addition to the "C" and "B" scales, there are other setups for special testing.

The scales, penetrators, major loads, and dial numbers to be read are listed in ***Figure 1-6***.

The Rockwell tester is equipped with a weight pan, and two weights are supplied with the machine. One weight is marked in red. The other weight is marked in black. With no weight in the weight pan, the machine applies a major load of 60 kilograms. If the scale setup calls for a 100 kilogram load, the red weight is placed in the pan. For a 150 kilogram load, the black weight is added to the red weight. The black weight is always used with the red weight; it is never used alone. Practically all testing is done with either the B-scale setup or the C-scale setup. For these scales, the colors may be used as a guide in selecting the weight (or weights) and in reading the dial. For the B-scale test, use the red weight and read the red numbers. For a C-scale test, add the black weight to the red weight and read the black numbers.

In setting up the Rockwell machine, use the diamond penetrator for testing materials known to be hard. If the hardness is unknown, try the diamond, since the steel ball may be deformed if used for testing hard materials. If the metal tests below C-22, then change to the steel ball.

Use the steel ball for all soft materials, those testing less than B-100. Should an overlap occur at the top of the B-scale and the bottom of the C-scale, use the C-scale setup. Before the major load is applied, securely lock the test specimen in place to prevent slipping and to seat the anvil and penetrator properly. To do this, apply a load of 10 kilograms before the lever is tripped. This preliminary load is called the minor load. The minor load is 10 kilograms regardless of the scale setup.

The metal to be tested in the Rockwell tester must be ground smooth on two opposite sides and be free of scratches and foreign matter. The surface should be perpendicular to the axis of penetration, and the two opposite ground surfaces should be parallel. If the specimen is tapered, the amount of error will depend on the taper. A curved surface will also cause a slight error in the hardness test. The amount of error depends on the curvature; i.e., the smaller the radius of curvature, the greater the error. To eliminate such error, a small flat should be ground on the curved surface if possible.

Clad aluminum alloy sheets cannot be tested directly with any accuracy with a Rockwell hardness tester. If the hardness value of the base metal is desired, the pure aluminum coating must be removed from the area to be checked prior to testing.

BARCOL TESTER

The Barcol tester (***Figure 2-4***) is a portable unit designed for testing aluminum alloys, copper, brass, or other relatively soft materials. It should not be used on aircraft steels. Approximate range of the tester is 25 to 100 Brinell. The unit can be used in any position and in any space that will allow for the operator's hand. It is of great value in the hardness testing of assembled or installed parts, especially to check for proper heat treatment. The hardness is indicated on a dial conveniently divided into 100 graduations.

The design of the Barcol tester is such that operating experience is not necessary. It is only necessary to exert a light pressure against the instrument to drive the spring loaded indenter into the material to be tested. The hardness reading is instantly indicated on the dial.

Several typical readings for aluminum alloys are listed in ***Figure 2-5***. Note that the harder the material is, the higher the Barcol number will be. To prevent damage to the point, avoid sliding or scraping when it is in contact with the material being tested. If the point should become damaged, it must be replaced with a new one. Do not attempt to grind the point.



Figure 2-4. Barcol portable hardness tester.

Alloy and Temper	Barcol Number
1100-0	35
3003-0	42
3003-H14	56
2024-0	60
5052-0	62
5052-H34	75
6061-T	78
2024-T	85

Figure 2-5. Typical Barcol readings for aluminum alloy.

Each tester is supplied with a test disk for checking the condition of the point. To check the point, press the instrument down on the test disk. When the downward pressure brings the end of the lower plunger guide against the surface of the disk, the indicator reading should be within the range shown on the test disk.

OTHER TESTING OF NON-FERROUS METALS

Tensile testing, fatigue testing, and impact testing of non-ferrous metals is similar to that of testing in ferrous metals. Refer to *Sub-Module 01* for more information.

Question: 2-1
Aluminum is one of the most widely used metals in modern aircraft construction because of its high _____ ratio and its comparative ease of fabrication.

Question: 2-2
In 2000 series aluminum alloy the principle alloying element is _____ which adds strength to the pure aluminum.

Question: 2-3
What is the principle safety concern when working with magnesium?

Question: 2-4
The use of copper and copper alloys in aircraft is often in _____ system components.

Question: 2-5
"Alclad" and "Pureclad" are often sheets of alloy aluminum clad with _____ to prevent corrosion.

Question: 2-6
When annealing aluminum to prepare it for cold working, it is cooled in _____.

Question: 2-7
What is used to determine the results of heat treatment as well as the state of a metal before heat treatment?

Question: 2-8
When cold working causes strain hardening making it difficult to form metal, it can be softened by _____.

ANSWERS

Answer: 2-1
strength to weight.

Answer: 2-2
copper.

Answer: 2-3
Fire.

Answer: 2-4
electrical.

Answer: 2-5
pure aluminum.

Answer: 2-6
still air.

Answer: 2-7
Hardness testing.

Answer: 2-8
annealing.



MATERIALS AND HARDWARE

AIRCRAFT MATERIALS COMPOSITE AND NON-METALLIC

SUB-MODULE 03

PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY → B1

Sub-Module 03

AIRCRAFT MATERIALS — COMPOSITE AND NON-METALLIC

Knowledge Requirements

6.3 Aircraft Materials — Composite and Non-Metallic

6.3.1 Composite and Non-Metallic other than Wood and Fabric

(a)	Characteristics, properties and identification of common composite and non-metallic materials, other than wood, used in aircraft; Sealant and bonding agents;	2
(b)	The detection of defects/deterioration in composite and non-metallic material; Repair of composite and non-metallic material.	2

6.3.2 Wooden Structure

	Construction methods of wooden airframe structures; Characteristics, properties and types of wood and glue used in aeroplanes; Preservation and maintenance of wooden structure; Types of defects in wood material and wooden structures; The detection of defects in wooden structure; Repair of wooden structure.	2

6.3.3 Fabric Covering

	Characteristics, properties and types of fabrics used in aeroplanes; Inspections methods for fabric; Types of defects in fabric; Repair of fabric covering.	2

COMPOSITE MATERIALS

In the 1940s, the aircraft industry began to develop synthetic fibers to enhance aircraft design. Since that time, composite materials have been used more and more. When composites are mentioned, most people think of only fiberglass, or maybe graphite or aramids (Kevlar). Composites began in aviation, but now are being embraced by many other industries, including auto racing, sporting goods, and boating, as well as defense industry uses.

A "composite" material is defined as a mixture of different materials or things. This definition is so general that it could refer to metal alloys made from several different metals to enhance the strength, ductility, conductivity or whatever characteristics are desired. Likewise, the composition of composite materials is a combination of reinforcement, such as a fiber, whisker, or particle, surrounded and held in place by a resin, forming a structure. Separately, the reinforcement and the resin are very different from their combined state. Even in their combined state, they can still be individually identified and mechanically separated. One composite, concrete, is composed of cement (resin) and gravel or reinforcement rods for the reinforcement to create the concrete.

ADVANTAGES/DISADVANTAGES OF COMPOSITES

The advantages for using composite materials are:

- High strength to weight ratio.
- Fiber-to-fiber transfer of stress allowed by chemical bonding.
- Modulus (stiffness to density ratio) 3.5 to 5 times that of steel or aluminum.
- Longer life than metals.
- Higher corrosion resistance.
- Tensile strength 4 to 6 times that of steel/aluminum.
- Greater design flexibility.
- Bonded construction eliminates joints and fasteners.
- Easily repairable.

The disadvantages of composites include:

- Inspection methods difficult to conduct, especially delamination detection (Advancements in technology will eventually correct this problem).
- Lack of long term design database, relatively new technology methods.
- Cost.
- Very expensive processing equipment.
- Lack of standardized system of methodology.
- Great variety of materials, processes, and techniques.
- General lack of repair knowledge and expertise.
- Products often toxic and hazardous.
- Lack of standardized methodology for construction and repairs.

The increased strength and the ability to design for the performance needs of the product makes composites much superior to the traditional materials used in today's aircraft. As more and more composites are used, the costs, design, inspection ease, and information about strength to weight advantages will help composites become the material of choice for aircraft construction.

COMPOSITE SAFETY

Composite products can be very harmful to the skin, eyes, and lungs. In the long or short term, people can become sensitized to the materials with serious irritation and health issues. Personal protection is often uncomfortable, hot, and difficult to wear; however, a little discomfort while working with the composite materials can prevent serious health issues or even death.

Respirator particle protection is very important to protecting the lungs from permanent damage from tiny glass bubbles and fiber pieces. At a minimum, a dust mask approved for fiberglass is a necessity. The best protection is a respirator with dust filters. The proper fit of a respirator or dust mask is very important because if the air around the seal is breathed, the mask cannot protect the wearer's lungs. When working with resins, it is important to use vapor protection. Charcoal filters in a respirator will remove the vapors for a period of time. If you can smell the resin vapors after placing the mask back on after a break, replace the filters immediately. Sometimes, charcoal filters last less than 4 hours. Store the respirator in a sealed bag when not in use. If working with toxic materials for an extended period of time, a supplied air mask and hood are recommended.

Avoid skin contact with the fibers and other particles by wearing long pants and long sleeves along with gloves or barrier creams. The eyes must be protected using

leak-proof goggles (no vent holes) when working with resins or solvents because chemical damage to the eyes is usually irreversible.

FIBER REINFORCED MATERIALS

The purpose of reinforcement in reinforced materials is to provide most of the strength. The three main forms of fiber reinforcements are particles, whiskers, and fibers.

A particle is a square piece of material. Glass bubbles (Q-cell) are hollow glass spheres, and since their dimensions are equal on all axes, they are called a particle. A whisker is a piece of material that is longer than it is wide. Whiskers are usually single crystals. They are very strong and used to reinforce ceramics and metals.

Fibers are single filaments that are much longer than they are wide. Fibers can be made of almost any material, and are not crystalline like whiskers. Fibers are the base for most composites. Fibers are smaller than the finest human hair and are normally woven into cloth-like materials.

LAMINATED STRUCTURES

Composites can be made with or without an inner core of material. Laminated structure with a core center is called a sandwich structure. Laminate construction is strong and stiff, but heavy. The sandwich laminate is equal in strength, and its weight is much less; less weight is very important to aerospace products. The core of a laminate can be made from nearly anything. The decision is normally based on use, strength, and fabricating methods to be used.

Various types of cores for laminated structures include rigid foam, wood, metal, or the aerospace preference of honeycomb made from paper, Nomex, carbon, fiberglass or metal. ***Figure 3-1*** shows a typical sandwich structure. It is very important to follow proper techniques to construct or repair laminated structures to ensure the strength is not compromised. A sandwich assembly is made by taking a high density laminate or solid face and backplate and sandwiching a core in the middle. The selection of materials for the face and backplate are decided by the design engineer, depending on the intended application of the part. It is important to follow manufacturer's maintenance manual specific instructions regarding testing and repair procedures as they apply to a particular aircraft.

ADVANCED LAMINATED COMPOSITE MATERIALS

Composite materials are becoming more important for aerospace structures. Aircraft parts made from composite materials, such as fairings, spoilers, and flight controls, were developed during the 1960s for their weight savings over aluminum parts. New generation large aircraft are designed with all composite fuselage and wing structures, and the repair of these advanced composite materials requires an in-depth knowledge of composite structures, materials, and tooling. The primary advantages of composite materials are their high strength, relatively low weight, and corrosion resistance.

Some applications of composites on aircraft include:

- Fairings.
- Flight control surfaces.
- Landing gear doors.
- Leading and trailing edge panels on the wing and stabilizer.

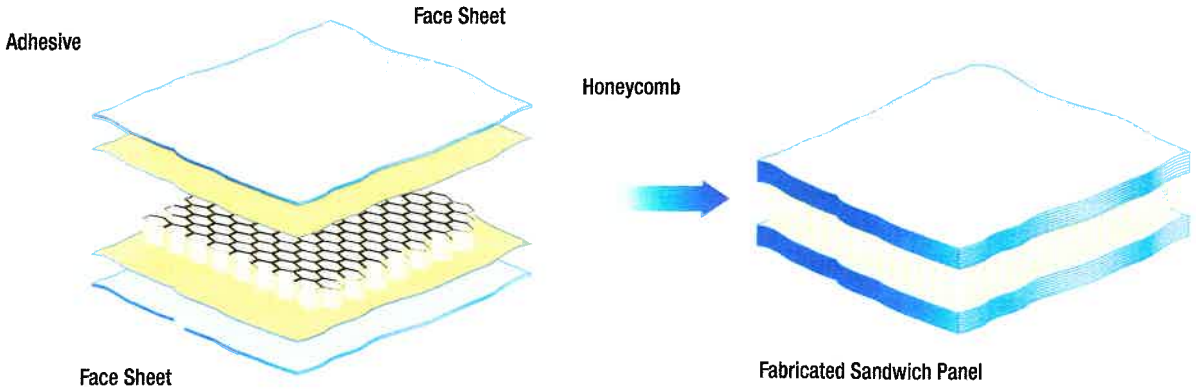


Figure 3-1. Sandwich structure.

- Interior components.
- Floor beams and floor boards.
- Vertical and horizontal stabilizer primary structure on large aircraft.
- Primary wing and fuselage structure on new generation large aircraft.
- Turbine engine fan blades.
- Propellers.

MAJOR COMPONENTS OF A LAMINATE

An isotropic material has uniform properties in all directions. The measured properties of an isotropic material are independent of the axis of testing. Metals such as aluminum and titanium are examples of isotropic materials.

A fiber is the primary load carrying element of the composite material. The composite material is only strong and stiff in the direction of the fibers. Unidirectional composites have predominant mechanical properties in one direction and are said to be anisotropic, having mechanical and/or physical properties that vary with direction relative to natural reference axes inherent in the material. Components made from fiber reinforced composites can be designed so that the fiber orientation produces optimum mechanical properties, but they can only approach the true isotropic nature of metals, such as aluminum and titanium.

A matrix supports the fibers and bonds them together in the composite material. The matrix transfers any applied loads to the fibers, keeps the fibers in their position and chosen orientation, gives the composite environmental resistance, and determines the maximum service temperature of a composite.

STRENGTH CHARACTERISTICS

Structural properties, such as stiffness, dimensional stability, and strength of a composite laminate, depend on the stacking sequence of the plies. The stacking sequence describes the distribution of ply orientations through the laminate thickness. As the number of plies with chosen orientations increases, more stacking sequences are possible. For example, a symmetric eight-ply laminate with four different ply orientations has 24 different stacking sequences.

FIBER

Fiber Orientation

The strength and stiffness of a composite buildup depends on the orientation sequence of the plies. The practical range of strength and stiffness of carbon fiber extends from values as low as those provided by fiberglass to as high as those provided by titanium. This range of values is determined by the orientation of the plies to the applied load. Proper selection of ply orientation in advanced composite materials is necessary to provide a structurally efficient design. The part might require 0° plies to react to axial loads, ±45° plies to react to shear loads, and 90° plies to react to side loads. Because the strength design requirements are a function of the applied load direction, ply orientation and ply sequence have to be correct. It is critical during a repair to replace each damaged ply with a ply of the same material and ply orientation.

The fibers in a unidirectional material run in one direction and the strength and stiffness is only in the direction of the fiber. Preimpregnated (prepreg) tape is an example of a unidirectional ply orientation.

The fibers in a bidirectional material run in two directions, typically 90° apart. A plain weave fabric is an example of a bidirectional ply orientation. These ply orientations have strength in both directions but not necessarily the same strength. (***Figure 3-2***)

The plies of a quasi-isotropic layup are stacked in a 0°, −45°, 45°, and 90° sequence or in a 0°, −60°, and 60° sequence. (***Figure 3-3***) These types of ply orientation simulate the properties of an isotropic material. Many aerospace composite structures are made of quasi-isotropic materials.

Warp Clock

Warp indicates the longitudinal fibers of a fabric. The warp is the high strength direction due to the straightness of the fibers. A warp clock is used to describe direction of fibers on a diagram, spec sheet, or manufacturer's sheets. If the warp clock is not available on the fabric, the orientation is defaulted to zero as the fabric comes off the roll. Therefore, 90° to zero is the width of the fabric across. (***Figure 3-4***)

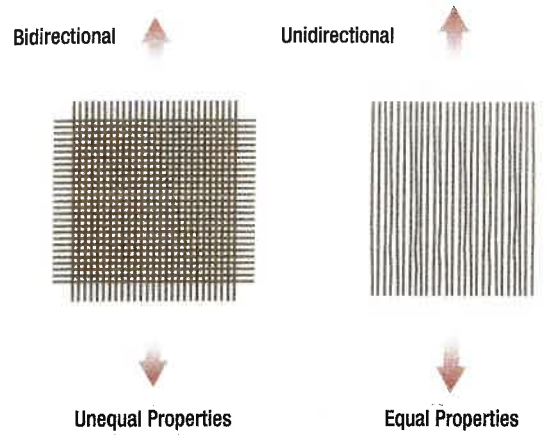


Figure 3-2. Bidirectional and unidirectional material properties.

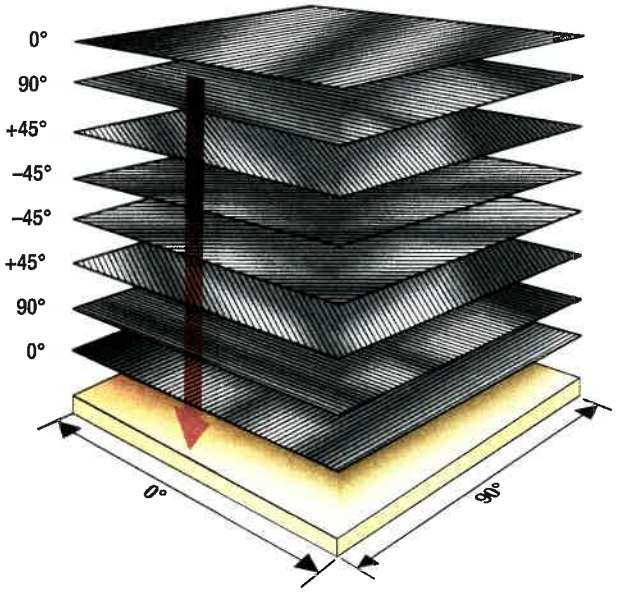


Figure 3-3. Quasi-isotropic material layup.

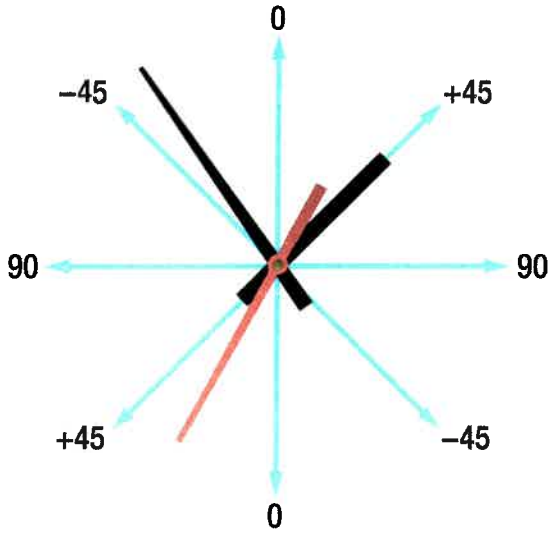


Figure 3-4. A warp clock.

Fiber Forms

All product forms generally begin with spooled unidirectional raw fibers packaged as continuous strands. An individual fiber is called a filament. The word strand is also used to identify an individual glass fiber. Bundles of filaments are identified as tows, yarns, or rovings. Fiberglass yarns are twisted, while Kevlar® yarns are not. Tows and rovings do not have any twist. Most fibers are available as dry fiber that needs to be impregnated (impreg) with a resin before use or prepreg materials where the resin is already applied to the fiber.

Roving

A roving is a single grouping of filament or fiber ends, such as 20-end or 60-end glass rovings. All filaments are in the same direction and they are not twisted. Carbon rovings are usually identified as 3K, 6K, or 12K rovings, K meaning 1 000 filaments. Most applications for roving products utilize mandrels for filament winding and then resin cure to final configuration.

Unidirectional (Tape)

Unidirectional prepreg tapes have been the standard within the aerospace industry for many years, and the fiber is typically impregnated with thermosetting resins. The most common method of manufacture is to draw collimated raw (dry) strands into the impregnation machine where hot melted resins are combined with the strands using heat and pressure.

Tape products have high strength in the fiber direction and virtually no strength across the fibers. The fibers are held in place by the resin. Tapes have a higher strength than woven fabrics. (***Figure 3-5***)

Bidirectional (Fabric)

Most fabric constructions offer more flexibility for layup of complex shapes than straight unidirectional tapes offer. Fabrics offer the option for resin impregnation either by solution or the hot melt process. Generally, fabrics used for structural applications use like fibers or strands of the same weight or yield in both the warp (longitudinal) and fill (transverse) directions. For aerospace structures, tightly woven fabrics are usually the choice to save weight, minimizing resin void size, and maintaining fiber orientation during the fabrication process.

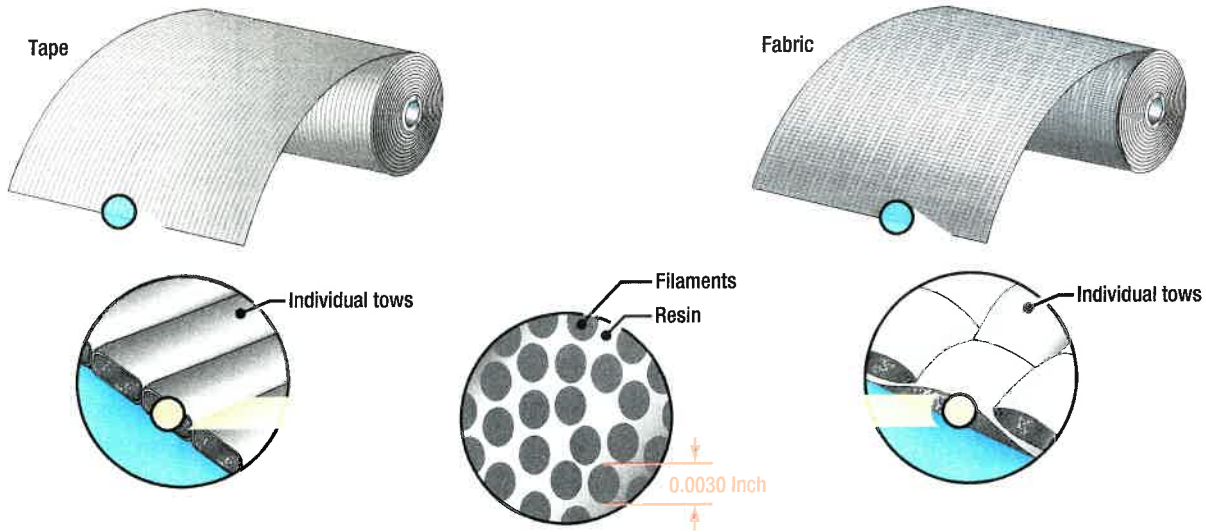


Figure 3-5. Tape and fabric products.

Woven structural fabrics are usually constructed with reinforcement tows, strands, or yarns interlocking upon themselves with over/under placement during the weaving process. The more common fabric styles are plain or satin weaves. The plain weave construction results from each fiber alternating over and then under each intersecting strand (tow, bundle, or yarn). With the common satin weaves, such as 5 harness or 8 harness, the fiber bundles traverse both in warp and fill directions changing over/under position less frequently.

These satin weaves have less crimp and are easier to distort than a plain weave. With plain weave fabrics and most 5 or 8 harness woven fabrics, the fiber strand count is equal in both warp and fill directions.

Example: 3K plain weave often has an additional designation, such as 12 × 12, meaning there are twelve tows per inch in each direction. This count designation can be varied to increase or decrease fabric weight or to accommodate different fibers of varying weight. (***Figure 3-6***)

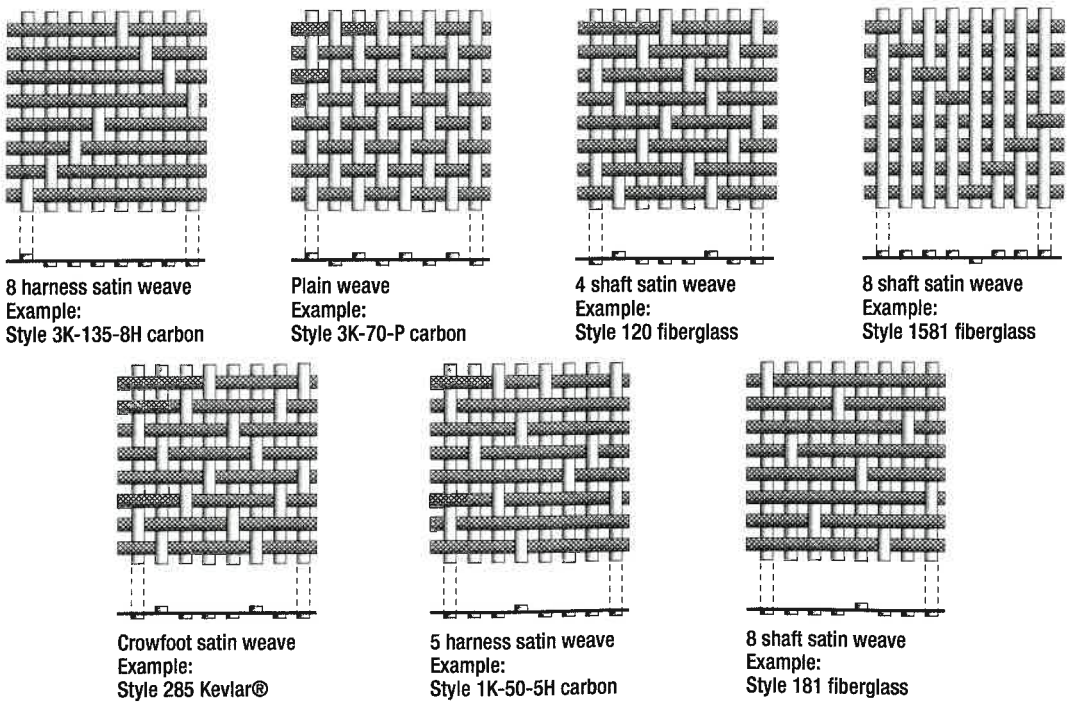


Figure 3-6. Typical fabric weave styles.

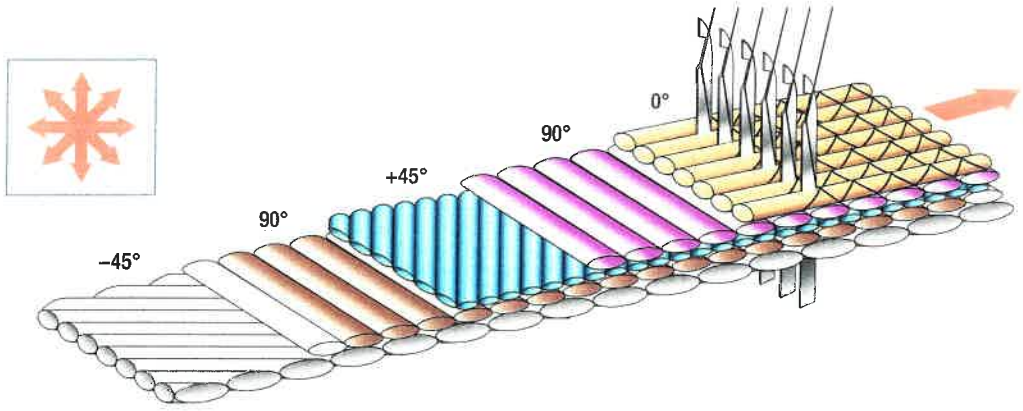


Figure 3-7. Nonwoven material (stitched).

Non-woven (Knitted or Stitched)

Knitted or stitched fabrics can offer many of the mechanical advantages of unidirectional tapes. Fiber placement can be straight or unidirectional without the over/under turns of woven fabrics. The fibers are held in place by stitching with fine yarns or threads after preselected orientations of one or more layers of dry plies. These types of fabrics offer a wide range of multi-ply orientations. Although there may be some added weight penalties or loss of some ultimate reinforcement fiber properties, some gain of interlaminar shear and toughness properties may be realized. Some common stitching yarns are polyester, aramid, or thermoplastics. (***Figure 3-7***)

Types of Fiber

Fiberglass

Fiberglass is often used for secondary structure on aircraft, such as fairings, radomes, and wing tips. Fiberglass is also used for helicopter rotor blades. There are several types of fiberglass used in the aviation industry. Electrical glass, or E-glass, is identified as such for electrical applications. It has high resistance to current flow. E-glass is made from borosilicate glass. S-glass and S2-glass identify structural fiberglass that have a higher strength than E-glass. S-glass is produced from magnesia-alumina-silicate. Advantages of fiberglass are lower cost than other composite materials, chemical or galvanic corrosion resistance, and electrical properties (fiberglass does not conduct electricity). Fiberglass has a white color and is available as a dry fiber fabric or prepreg material.

Kevlar®

Kevlar® is DuPont's name for aramid fibers. Aramid fibers are light weight, strong, and tough. Two types of Aramid fiber are used in the aviation industry. Kevlar® 49 has a high stiffness and Kevlar® 29 has a low stiffness. An advantage of aramid fibers is their high resistance to impact damage, so they are often used in areas prone to impact damage. The main disadvantage of aramid fibers is their general weakness in compression and hygroscopy. Service reports have indicated that some parts made from Kevlar® absorb up to 8 percent of their weight in water. Therefore, parts made from aramid fibers need to be protected from the environment. Another disadvantage is that Kevlar® is difficult to drill and cut. The fibers fuzz easily and special scissors are needed to cut the material. Kevlar® is often used for military ballistic and body armor applications. It has a natural yellow color and is available as dry fabric and prepreg material. Bundles of aramid fibers are not sized by the number of fibers like carbon or fiberglass but by the weight.

Carbon/Graphite

One of the first distinctions to be made among fibers is the difference between carbon and graphite fibers, although the terms are frequently used interchangeably. Carbon and graphite fibers are based on graphene (hexagonal) layer networks present in carbon. If the graphene layers, or planes, are stacked with three dimensional order, the material is defined as graphite. Usually extended time and temperature processing is required to form this order, making graphite fibers more expensive. Bonding between planes is weak. Disorder frequently occurs such that only two dimensional ordering within the layers is present. This material is defined as carbon.

Carbon fibers are very stiff and strong, 3 to 10 times stiffer than glass fibers. Carbon fiber is used for structural aircraft applications, such as floor beams, stabilizers, flight controls, and primary fuselage and wing structure. Advantages include its high strength

and corrosion resistance. Disadvantages include lower conductivity than aluminum; therefore, a lightning protection mesh or coating is necessary for aircraft parts that are prone to lightning strikes. Another disadvantage of carbon fiber is its high cost. Carbon fiber is gray or black in color and is available as dry fabric and prepreg material. Carbon fibers have a high potential for causing galvanic corrosion when used with metallic fasteners and structures. (***Figure 3-8***)

Boron

Boron fibers are very stiff and have a high tensile and compressive strength. The fibers have a relatively large diameter and do not flex well; therefore, they are available only as a prepreg tape product. An epoxy matrix is often used with the boron fiber. Boron fibers are used to repair cracked aluminum aircraft skins, because the thermal expansion of boron is close to aluminum and there is no galvanic corrosion potential. The boron fiber is difficult to use if the parent material surface has a contoured shape. The boron fibers are very expensive and can be hazardous for personnel. Boron fibers are used primarily in military aviation applications.

Ceramic Fibers

Ceramic fibers are used for high-temperature applications, such as turbine blades in a gas turbine engine. The ceramic fibers can be used to temperatures up to 1 204°C.

Lightning Protection Fibers

An aluminum airplane is quite conductive and is able to dissipate the high currents resulting from a lightning strike. Carbon fibers are 1 000 times more resistive than aluminum to current flow, and epoxy resin is 1 000 000 times more resistive (i.e., perpendicular to the skin). The surface of an external composite component often consists of a ply or layer of conductive material for lightning strike protection because composite materials are less conductive than aluminum. Many different types of conductive materials are used ranging from nickel coated graphite cloth to metal meshes to aluminized fiberglass to conductive paints. The materials are available for wet layup and as prepreg.

In addition to a normal structural repair, the technician must also recreate the electrical conductivity designed into the part. These types of repair generally require a conductivity test to be performed with an ohmmeter to verify minimum electrical resistance across the structure. When repairing these types of structures, it is extremely important to use only the approved materials from authorized vendors, including such items as potting compounds, sealants, adhesives, and so forth. (***Figures 3-9 and 3-10***)

Dry Fiber Material

Dry fiber materials, such as carbon, glass, and Kevlar®, are used for many repair procedures. The dry fabric is impregnated with a resin just before the repair work starts. This process is called wet layup. The main advantage of using the wet layup process is that the fiber and resin can be stored for a long time at room temperature. The composite can be cured at room temperature or an elevated temperature cure can speed up the curing process and increase the strength. The disadvantage is that the process is messy and reinforcement properties are less than prepreg material properties. (***Figure 3-11***)

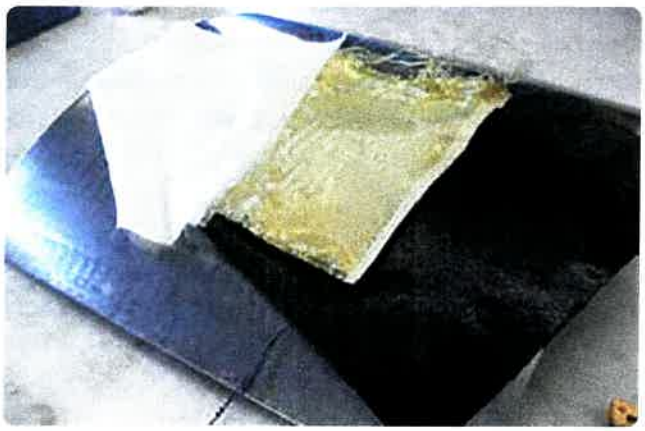


Figure 3-8. Fiberglass (left), Kevlar® (middle), and carbon fiber material (right).



Figure 3-9. Copper mesh lightning protection material.



Figure 3-10. Aluminum mesh lightning protection material.



Figure 3-11. Dry fabric materials (top to bottom: aluminum lightning protection mess, Kevlar®, fiberglass and carbon fiber).

THERMOSETTING RESINS

Resin is a generic term used to designate the polymer. The resin, its chemical composition, and physical properties fundamentally affect the processing, fabrication, and ultimate properties of a composite material. Thermosetting resins are the most diverse and widely used of all man-made materials. They are easily poured or formed into any shape, are compatible with most other materials, and cure readily (by heat or catalyst) into an insoluble solid. Thermosetting resins are also excellent adhesives and bonding agents.

Polyester Resins

Polyester resins are relatively inexpensive, fast processing resins used generally for low cost applications. Low smoke producing polyester resins are used for interior parts of the aircraft. Fiber-reinforced polyesters can be processed by many methods. Common processing methods include matched metal molding, wet layup, press (vacuum bag) molding, injection molding, filament winding, pultrusion, and autoclaving.

Vinyl Ester Resin

The appearance, handling properties, and curing characteristics of vinyl ester resins are the same as those of conventional polyester resins. However, the corrosion resistance and mechanical properties of vinyl ester composites are much improved over standard polyester resin composites.

Phenolic Resin

Phenol-formaldehyde resins were first produced commercially in the early 1900s for use in the commercial market. Ureaformaldehyde and melamine-formaldehyde appeared in the 1920–1930s as a less expensive alternative for lower temperature use. Phenolic resins are used for interior components because of their low smoke and flammability characteristics.

Epoxy

Epoxies are polymerizable thermosetting resins and are available in a variety of viscosities from liquid to solid. There are many different types of epoxy, and the technician should use the maintenance manual to select the correct type for a specific repair. Epoxies are used widely in resins for prepreg materials and structural adhesives. The advantages of epoxies are high strength and modulus, low levels of volatiles, excellent adhesion, low shrinkage, good chemical resistance, and ease of processing. Their major disadvantages are brittleness and the reduction of properties in the presence of moisture.

The processing or curing of epoxies is slower than polyester resins. Processing techniques include autoclave molding, filament winding, press molding, vacuum bag molding, resin transfer molding, and pultrusion. Curing temperatures vary from room temperature to approximately 180°C. The most common cure temperatures range between 120–180°C. (***Figure 3-12***)

Polyimides

Polyimide resins excel in high-temperature environments where their thermal resistance, oxidative stability, low coefficient of thermal expansion, and solvent resistance benefit the design. Their primary uses are circuit boards and hot engine and airframe structures. A polyimide may be either a thermoset resin or a thermoplastic. Polyimides require high cure temperatures, usually in excess of 290°C. Consequently, normal epoxy composite bagging materials are not usable, and steel tooling becomes a necessity. Polyimide bagging and



Figure 3-12. Two part wet layup epoxy resin system with pump dispenser.

release films, such as Kapton® are used. It is extremely important that Upilex® replace the lower cost nylon bagging and Polytetrafluoroethylene (PTFE) release films common to epoxy composite processing. Fiberglass fabrics must be used for bleeder and breather materials instead of polyester mat materials due to the low melting point of polyester.

Polybenzimidazoles (PBI)

Polybenzimidazole (PBI) resin is extremely high temperature resistant and is used for high temperature materials. These resins are available as adhesive and fiber.

Bismaleimides (BMI)

Bismaleimide (BMI) resins have a higher temperature capability and higher toughness than epoxy resins, and they provide excellent performance at ambient and elevated temperatures. The processing of bismaleimide resins is similar to that for epoxy resins. BMIs are used for aero engines and high temperature components. BMIs are suitable for standard autoclave processing, injection molding, resin transfer molding, and Sheet Molded Compound (SMC) among others.

Thermoplastic Resins

Thermoplastic materials can be softened repeatedly by an increase of temperature and hardened by a decrease in temperature. Processing speed is the primary advantage of thermoplastic materials. Chemical curing of the material does not take place during processing, and the material can be shaped by molding or extrusion when it is soft.

Semicrystalline Thermoplastics

Semicrystalline thermoplastics possess properties of inherent flame resistance, superior toughness, good mechanical properties at elevated temperatures and after impact, and low moisture absorption. They are used in secondary and primary aircraft structures. Combined with reinforcing fibers, they are available in injection molding compounds, compression moldable random sheets, unidirectional tapes, prepregs fabricated from tow (towpreg), and woven prepregs. Fibers impregnated in semicrystalline thermoplastics include carbon, nickel-coated carbon, aramid, glass, quartz, and others.

Amorphous Thermoplastics

Amorphous thermoplastics are available in several physical forms, including films, filaments, and powders. Combined with reinforcing fibers, they are also available in injection molding compounds, compressive moldable random sheets, unidirectional tapes, woven prepregs, etc. The fibers used are primarily carbon, aramid, and glass. The specific advantages of amorphous thermoplastics depend upon the polymer. Typically, the resins are noted for their processing ease and speed, high temperature capability, good mechanical properties, excellent toughness and impact strength, and chemical stability. The stability results in unlimited shelf life, eliminating the cold storage requirements of thermoset prepregs.

Polyether Ether Ketone (PEEK)

Polyether Ether Ketone, better known as PEEK, is a high temperature thermoplastic. This aromatic ketone material offers outstanding thermal and combustion characteristics and resistance to a wide range of solvents and proprietary fluids. PEEK can also be reinforced with glass and carbon.

Curing Stages of Resins

Thermosetting resins use a chemical reaction to cure. There are three curing stages, which are A, B, and C.

- A stage: The components of the resin (base material and hardener) have been mixed but the chemical reaction has not started. The resin is in the A stage during a wet layup procedure.
- B stage: The components of the resin have been mixed and the chemical reaction has started. The material has thickened and is tacky. The resins of prepreg materials are in the B stage. To prevent further curing the resin is placed in a freezer at -18°C. In the frozen state, the resin of the prepreg material stays in the B stage. The curing starts when the material is removed from the freezer and warmed again.

- C stage: The resin is fully cured. Some resins cure at room temperature and others need an elevated temperature cure cycle to fully cure.

Thixotropic Agents

Thixotropic agents are gel like at rest but become fluid when agitated. These materials have high static shear strength and low dynamic shear strength at the same time to lose viscosity under stress.

In some processes such as hand lay-ups, Thixoropic agents may be added to the resin in .5-6% ratios in order to control the viscosity of the resin. When at rest, resins containing thixotropic agents remain at elevated viscosities. This reduces the tendency of the resin to flow or drain from surfaces. When the resin is subjected to shear, the viscosity is reduced and the resin can easily be brushed onto a mold. Silica and certain clays are common thixotropic agents.

PRE-IMPREGNATED PRODUCTS (PREPREGS)

Prepreg material consists of a combination of a matrix and fiber reinforcement. It is available in unidirectional form (one direction of reinforcement) and fabric form (several directions of reinforcement). All five of the major families of matrix resins can be used to impregnate various fiber forms. The resin is then no longer in a low-viscosity stage, but has been advanced to a B stage level of cure for better handling characteristics. The following products are available in prepreg form: unidirectional tapes, woven fabrics, continuous strand rovings, and chopped mat. Prepreg materials must be stored in a freezer at a temperature below -18°C to retard the curing process. Prepreg materials are cured with an elevated temperature. Many prepreg materials used in aerospace are impregnated with an epoxy resin and they are cured at either 120°C or 175°C. Prepreg materials are cured with an autoclave, oven, or heat blanket. They are typically purchased and stored on a roll in a sealed plastic bag to avoid moisture contamination. (***Figure 3-13***)

ADHESIVES

Film Adhesives

Structural adhesives for aerospace applications are generally supplied as thin films supported on a release paper and stored under refrigerated conditions -18°C. Film adhesives are available using high temperature aromatic amine or catalytic curing agents with a wide range of flexibilizing and toughening agents. Rubber-toughened epoxy film adhesives are widely used in the aircraft industry. The upper temperature limit of 121–177°C is usually dictated by the degree of toughening required and by the overall choice of resins and curing agents. In general, toughening of a resin results in a lower usable service temperature. Film materials are frequently supported by fibers that serve to improve handling of the films prior to cure, control adhesive flow during bonding, and assist in bond line thickness control. Fibers can be incorporated as short fiber mats with random orientation or as woven cloth.

Commonly encountered fibers are polyesters, polyamides (nylon), and glass. Adhesives containing woven cloth may have slightly degraded environmental properties because of wicking of water by the fiber. Random mat scrim cloth is not as efficient for controlling film thickness as woven cloth because the unrestricted fibers move during bonding. Spunbonded nonwoven scrims do not move and are, therefore, widely used.
(***Figures 3-14 and 3-15***)

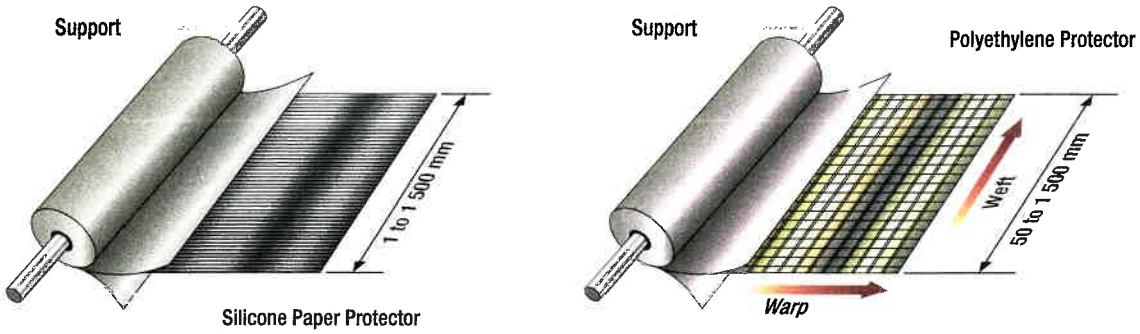


Figure 3-13. Tape and fabric prepreg materials.

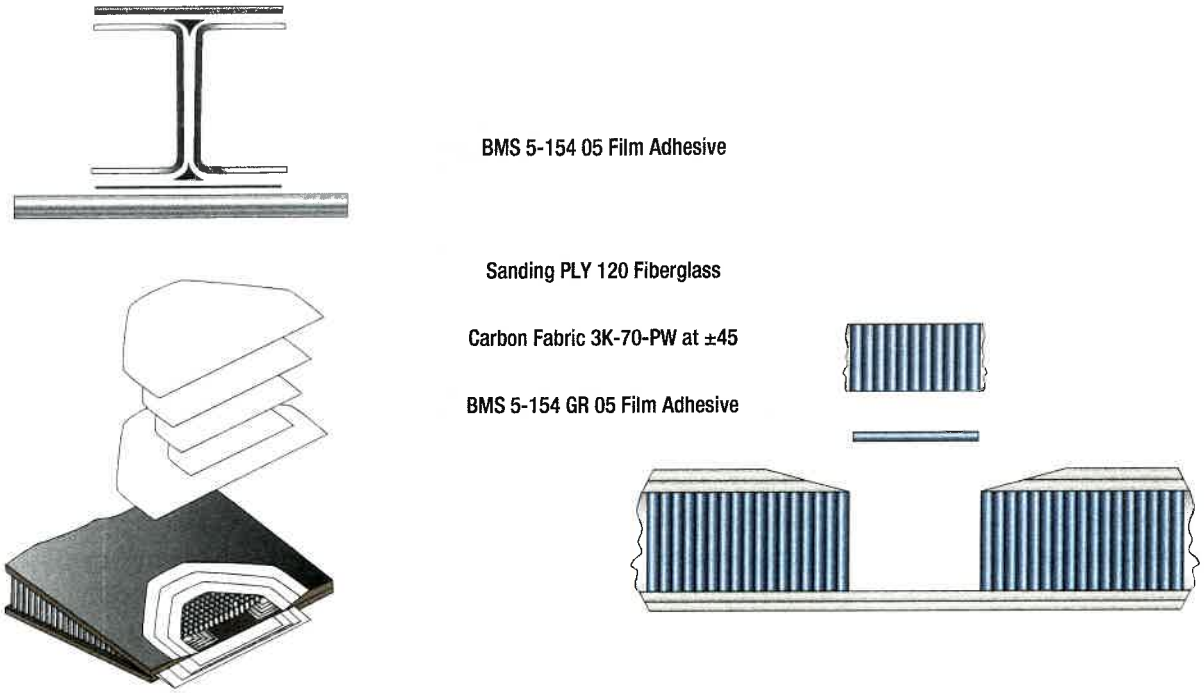


Figure 3-14. The use of film adhesive, Kevlar®, fiberglass, and carbon fiber.

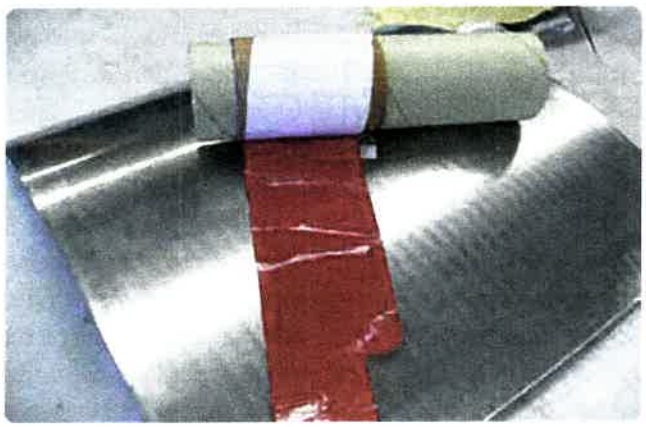


Figure 3-15. A roll of film adhesive.



Figure 3-16. Two-part paste adhesive.

Paste Adhesives

Paste adhesives are used as an alternative to film adhesive. These are often used to secondary bond repair patches to damaged parts and also used in places where film adhesive is difficult to apply. Paste adhesives for structural bonding are made mostly from epoxy. One part and two part systems are available. The advantages of paste adhesives are that they can be stored at room temperature and have a long shelf life. The disadvantage is that the bond line thickness is hard to control, which affects the strength of the bond. A scrim cloth can be used to maintain adhesive in the bondline when bonding patches with paste adhesive. (***Figure 3-16***)

Foaming Adhesives

Most foaming adhesives are 0.025 inch to 0.10 inch thick sheets of B staged epoxy. Foam adhesives cure at 121°C or 177°C. During the cure cycle, foaming adhesives expand. Foaming adhesives need to be stored in the freezer just like prepregs, and they have only a limited storage life. Foaming adhesives are used to splice pieces of honeycomb together in a sandwich construction and to bond repair plugs to existing core during a prepreg repair. (***Figure 3-17***)

SANDWICH STRUCTURES

A sandwich construction is a structural panel concept that consists in its simplest form of two relatively thin, parallel face sheets bonded to and separated by

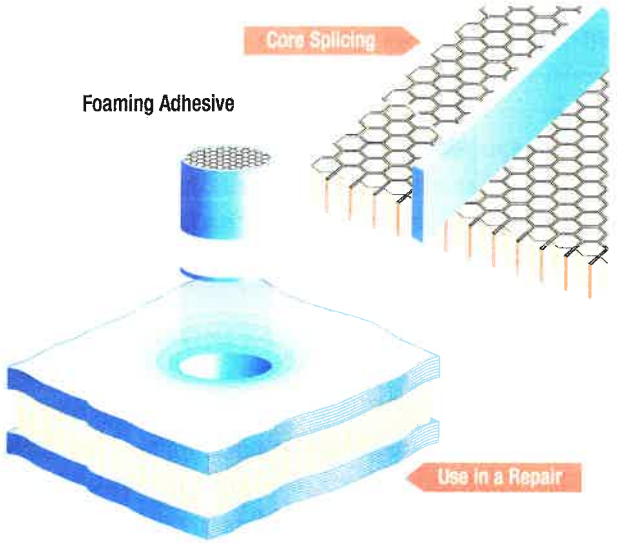


Figure 3-17. The use of foaming adhesives.

a relatively thick, lightweight core. The core supports the face sheets against buckling and resists out-of-plane shear loads. The core must have high shear strength and compression stiffness.

Composite sandwich construction is most often fabricated using autoclave cure, press cure, or vacuum bag cure. Skin laminates may be precured and subsequently bonded to core, co-cured to core in one operation, or a combination of the two methods. Examples of honeycomb structure are: wing spoilers, fairings, ailerons, flaps, nacelles, floor boards, and rudders. (***Figure 3-18***)

Properties

Sandwich construction has high bending stiffness at minimal weight in comparison to aluminum and composite laminate construction. Most honeycombs are anisotropic; that is, properties are directional.

Figure 3-19 illustrates the advantages of using a honeycomb construction. Increasing the core thickness greatly increases the stiffness of the honeycomb construction, while the weight increase is minimal. Due to the high stiffness of a honeycomb construction, it is not necessary to use external stiffeners, such as stringers and frames.

Facing Materials

Most honeycomb structures used in aircraft construction have aluminum, fiberglass, Kevlar®, or carbon fiber face sheets. Carbon fiber face sheets cannot be used with aluminum honeycomb core material, because it causes

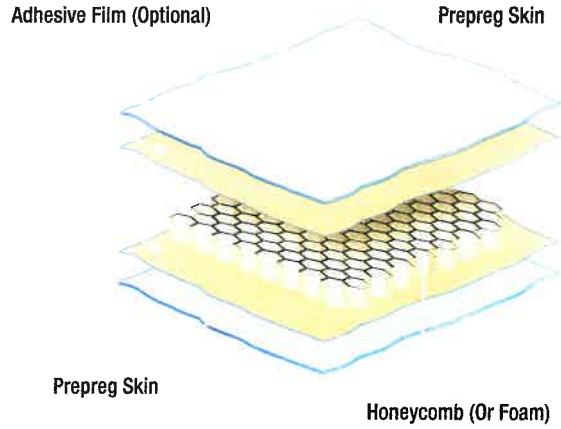


Figure 3-18. Honeycomb sandwich construction.

	Solid Material	Core Thickness t	Core Thickness 3t
	t	2t	4t
Thickness	1.0	7.0	37.0
Flexural Strength	1.0	3.5	9.2
Weight	1.0	1.03	1.06

Figure 3-19. Strength and stiffness of honeycomb sandwich material compared to a solid laminate.

the aluminum to corrode. Titanium and steel are used for specialty applications in high temperature constructions. The face sheets of many components, such as spoilers and flight controls, are very thin—sometimes only 3 or 4 plies. Field reports have indicated that these face sheets do not have a good impact resistance.

Core Materials

Honeycomb

Each honeycomb material provides certain properties and has specific benefits. (***Figure 3-20***) The most common core material used for aircraft honeycomb structures is aramid paper (Nomex® or Korex®). Fiberglass is used for higher strength applications.

- Kraft paper—relatively low strength, good insulating properties, is available in large quantities, and has a low cost.
- Thermoplastics—good insulating properties, good energy absorption and/or redirection, smooth cell walls, moisture and chemical resistance, are environmentally compatible, aesthetically pleasing, and have a relatively low cost.

- Aluminum—best strength-to-weight ratio and energy absorption, has good heat transfer properties, electromagnetic shielding properties, has smooth, thin cell walls, is machinable, and has a relatively low cost.
- Steel—good heat transfer properties, electromagnetic shielding properties, and heat resistant.
- Specialty metals (titanium)—relatively high strength to weight ratio, good heat transfer properties, chemical resistance, and heat resistant to very high temperatures.
- Aramid paper—flame resistant, fire retardant, good insulating properties, low dielectric properties, and good formability.
- Fiberglass—tailorable shear properties by layup, low dielectric properties, good insulating properties, and good formability.
- Carbon—good dimensional stability and retention, high-temperature property retention, high stiffness, very low coefficient of thermal expansion, tailorable thermal conductivity, relatively high shear modulus, and very expensive.
- Ceramics—heat resistant to very high temperatures, good insulating properties, is available in very small cell sizes, and very expensive. (***Figure 3-20***)

Honeycomb core cells for aerospace applications are usually hexagonal. The cells are made by bonding stacked sheets at special locations. The stacked sheets are expanded to form hexagons. The direction parallel to the sheets is called ribbon direction.

Bisected hexagonal core has another sheet of material cutting across each hexagon. Bisected hexagonal honeycomb is stiffer and stronger than hexagonal core.

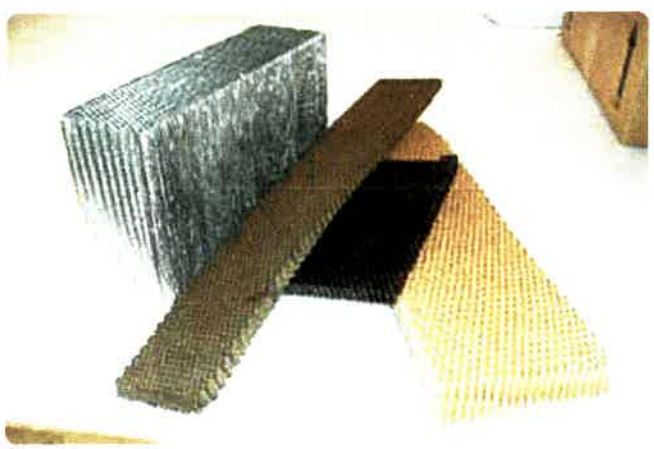
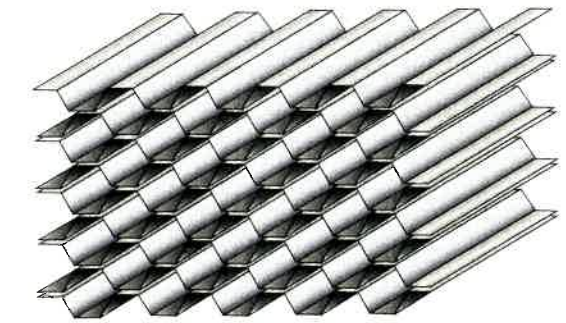


Figure 3-20. Honeycomb core materials.

Overexpanded core is made by expanding the sheets more than is needed to make hexagons. The cells of overexpanded core are rectangular. Overexpanded core is flexible perpendicular to the ribbon direction and is used in panels with simple curves. Bell shaped core, or flexicore, has curved cell walls, that make it flexible in all directions. Bell shaped core is used in panels with complex curves.

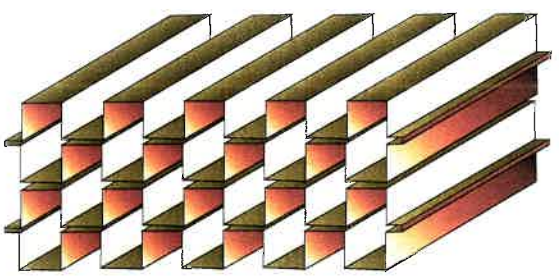
Honeycomb core is available with different cell sizes. Small sizes provide better support for sandwich face sheets. Honeycomb is also available in different densities. Higher density core is stronger and stiffer than lower density core. (***Figure 3-21***)



Hexagonal Honeycomb Core



Flexicore



Overexpanded Core

Figure 3-21. Honeycomb density.

Foam

Foam cores are used on homebuilts and lighter aircraft to give strength and shape to wing tips, flight controls, fuselage sections, wings, and wing ribs. Foam cores are not commonly used on commercial type aircraft. Foams are typically heavier than honeycomb and not as strong.

A variety of foams can be used as core material including:

- Polystyrene (better known as styrofoam)—aircraft grade styrofoam with a tightly closed cell structure and no voids between cells; high compressive strength and good resistance to water penetration; can be cut with a hot wire to make airfoil shapes.
- Phenolic—very good fire-resistant properties and can have very low density, but relatively low mechanical properties.
- Polyurethane—used for producing fuselage, wing tips, and other curved parts of small aircraft; relatively inexpensive, fuel resistant, and compatible with most adhesives; do not use hot wire to cut polyurethane foam; easily contoured with a large knife and sanding equipment.
- Polypropylene—used to make airfoil shapes; can cut with a hot wire; compatible with most adhesives and epoxy resins; not for use with polyester resins, dissolves in fuels and solvents.
- Polyvinyl chloride (PVC) (Divinycell, Klegecell, and Airex)—a closed cell medium to high density foam with high compression strength, durability, and excellent fire resistance; can be vacuum formed to compound shapes and be bent using heat; compatible with polyester, vinyl ester, and epoxy resins.
- Polymethacrylimide (Rohacell)—a closed-cell foam used for lightweight sandwich construction; excellent mechanical properties, high dimensional stability under heat, good solvent resistance, and outstanding creep compression resistance; more expensive than the other types of foams, but has greater mechanical properties.

Balsa Wood

Balsa is a natural wood product with elongated closed cells; it is available in a variety of grades that correlate to the structural, cosmetic, and physical characteristics. The density of balsa is less than one half of the density of conventional wood products. However, balsa has a considerably higher density than the other types of structural cores.

PLASTICS

Plastics are used in many applications throughout modern aircraft. These applications range from structural components of thermosetting plastics reinforced with fiberglass to decorative trim of thermoplastic materials to windows.

In addition to the above classes, transparent plastics are manufactured in two forms: monolithic (solid) and laminated. Laminated transparent plastics are made from transparent plastic face sheets bonded by an inner layer material, usually polyvinyl butyryl. Because of its shatter resistant qualities, laminated plastic is superior to solid plastics and is used in many pressurized aircraft.

Most of the transparent sheet used in aviation is manufactured in accordance with various military specifications. A new development in transparent plastics is stretched acrylic. Stretched acrylic is a type of plastic which, before being shaped, is pulled in both directions to rearrange its molecular structure. Stretched acrylic panels have a greater resistance to impact and are less subject to shatter; its chemical resistance is greater, edging is simpler, and crazing and scratches are less detrimental.

REINFORCED PLASTIC

Reinforced plastic is a thermosetting material used in the manufacture of radomes, antenna covers, and wingtips, and as insulation for various pieces of electrical equipment and fuel cells. It has excellent dielectric characteristics which make it ideal for radomes; however, its high strength-to-weight ratio, resistance to mildew, rust, and rot, and ease of fabrication make it equally suited for other parts of the aircraft.

Reinforced plastic components of aircraft are formed of either solid laminates or sandwich type laminates. Resins used to impregnate glass cloths are of the contact pressure type (requiring little or no pressure during cure). These resins are supplied as a liquid which can vary in viscosity from a waterlike consistency to a thick syrup. Cure or polymerization is effected by the use of a catalyst, usually benzoyl peroxide.

Solid laminates are constructed of three or more layers of resin impregnated cloths "wet laminated" together to form a solid sheet facing or molded shape. Sandwich-type laminates are constructed of two or more solid sheet facings or a molded shape enclosing a fiberglass

honeycomb or foam type core. Honeycomb cores are made of glass cloths impregnated with a polyester or a combination of nylon and phenolic resins. The specific density and cell size of honeycomb cores varies over a considerable latitude. Honeycomb cores are normally fabricated in blocks that are later cut to the desired thickness on a bandsaw.

Foam type cores are formulated from combinations of alkyd resins and metatoluene diisocyanate. Sandwich type fiberglass components filled with foam type cores are manufactured to exceedingly close tolerances on overall thickness of the molded facing and core material. To achieve this accuracy, the resin is poured into a close tolerance, molded shape. The resin formulation immediately foams up to fill the void in the molded shape and forms a bond between the facing and the core.

TRANSPARENT PLASTICS

Plastics cover a broad field of organic synthetic resin and may be divided into two main classifications: thermoplastics and thermosetting plastics.

- Thermoplastics—may be softened by heat and can be dissolved in various organic solvents. Acrylic plastic is commonly used as a transparent thermoplastic material for windows, canopies, etc. Acrylic plastics are known by the trade names of Lucite® or Plexiglas® and by the British as Perspex®, and meet the military specifications of MIL-P-5425 for regular acrylic and MIL-P-8184 for craze resistant acrylic.
- Thermosetting plastics—do not soften appreciably under heat but may char and blister at temperatures of 240–260°C. Most of the molded products of synthetic resin composition, such as phenolic, urea-formaldehyde, and melamine formaldehyde resins, belong to the thermosetting group. Once the plastic becomes hard, additional heat does not change it back into a liquid as it would with a thermoplastic.

OPTICAL CONSIDERATIONS

Scratches and other types of damage that obstruct the vision of the pilots are not acceptable. Some types of damage might be acceptable at the edges of the windshield.

STORAGE AND HANDLING

Because transparent thermoplastic sheets soften and deform when they are heated, they must be stored where the temperature never becomes excessive. Store them in a cool, dry location away from heating coils, radiators, or steam pipes, and away from such fumes as are found in paint spray booths or paint storage areas. Keep paper masked transparent sheets out of the direct rays of the sun, because sunlight accelerates deterioration of the adhesive, causing it to bond to the plastic, and making it difficult to remove.

Store plastic sheets with the masking paper in place, in bins that are tilted at a 10° angle from the vertical to prevent buckling. If the sheets are stored horizontally, take care to avoid getting dirt and chips between them. Stacks of sheets must never be over 18 inches high, with the smallest sheets stacked on top of the larger ones so there is no unsupported overhang. Leave the masking paper on the sheets as long as possible, and take care not to scratch or gouge the sheets by sliding them against each other or across rough or dirty tables.

Store formed sections with ample support so they do not lose their shape. Vertical nesting should be avoided. Protect formed parts from temperatures higher than 49°C, and leave their protective coating in place until they are installed on the aircraft.

FORMING PROCEDURES AND TECHNIQUES

Transparent acrylic plastics get soft and pliable when they are heated to their forming temperatures and can be formed to almost any shape. When they cool, they retain the shape to which they were formed. Acrylic plastic may be cold bent into a single curvature if the material is thin and the bending radius is at least 180 times the thickness of the sheet. Cold bending beyond these limits impose so much stress on the surface of the plastic that tiny fissures or cracks, called crazing, form.

Heating

Wear cotton gloves when handling the plastic to eliminate finger marks on the soft surface. Before heating any transparent plastic material, remove all of the masking paper and adhesive from the sheet. If the sheet is dusty or dirty, wash it with clean water and rinse it well. Dry the sheet thoroughly by blotting it with soft absorbent paper towels.

For the best results when hot forming acrylics, adhere to the temperatures recommended by the manufacturer. Use a forced air oven that can operate over a temperature range of 49–190°C. If the part gets too hot during the forming process, bubbles may form on the surface and impair the optical qualities of the sheet.

For uniform heating, it is best to hang the sheets vertically by grasping them by their edges with spring clips and suspending the clips in a rack. (***Figure 3-22***) If the piece is too small to hold with clips, or if there is not enough trim area, lay the sheets on shelves or racks covered with soft felt or flannel. Be sure there is enough open space to allow the air to circulate around the sheet and heat it evenly.

Small forming jobs, such as landing light covers, may be heated in a kitchen baking oven. Infrared heat lamps may be used if they are arranged on 7 to 8 inch centers and enough of them are used in a bank to heat the sheet evenly. Place the lamps about 18 inches from the material. Never use hot water or steam directly on the plastic to heat it because this likely causes the acrylic to become milky or cloudy.

Forms

Heated acrylic plastic molds with almost no pressure, so the forms used can be of very simple construction. Forms made of pressed wood, plywood, or plaster are adequate to form simple curves, but reinforced plastic or plaster may be needed to shape complex or compound curves. Since hot plastic conforms to any waviness or unevenness, the form used must be completely smooth. To ensure this, sand the form and cover it with soft cloth, such as outing flannel or billiard felt. The mold should be large enough to extend beyond the trim line of the part, and provisions should be made for holding the hot plastic snug against the mold as it cools.

A mold can be made for a complex part by using the damaged part itself. If the part is broken, tape the pieces together, wax or grease the inside so the plaster does not stick to it, and support the entire part in sand. Fill the part with plaster and allow it to harden, and then remove it from the mold. Smooth out any roughness and cover it with soft cloth. It is now ready to use to form the new part.

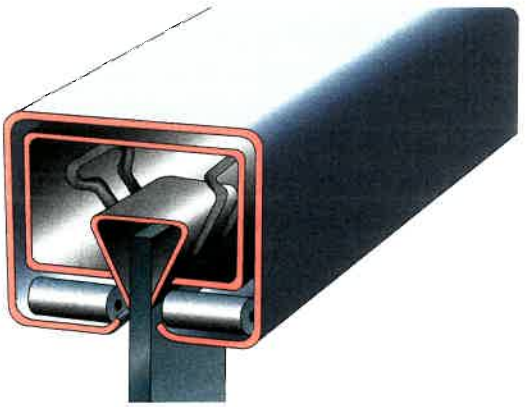


Figure 3-22. Hanging an acrylic sheet.

FORMING METHODS

Simple Curve Forming

Heat the plastic material to the recommended temperature, remove it from the heat source, and carefully drape it over the prepared form. Carefully press the hot plastic to the form and either hold or clamp the sheet in place until it cools. This process may take from 10–30 minutes. Do not force cool it.

Compound Curve Forming

Compound curve forming is normally used for canopies or complex wingtip light covers, and it requires a great deal of specialized equipment. There are four commonly used methods, each having its advantages and disadvantages.

Stretch Forming

Preheated acrylic sheets are stretched mechanically over the form in much the same way as is done with the simple curved piece. Take special care to preserve uniform thickness of the material, since some parts must stretch more than others.

Male and Female Die Forming

Male and female die forming requires expensive matching male and female dies. The heated plastic sheet is placed between the dies that are then mated. When the plastic cools, the dies are opened.

Vacuum Forming Without Forms

Many aircraft canopies are formed by this method. In this process, a panel, which has cut into it the outline of the desired shape, is attached to the top of a vacuum box. The heated and softened sheet of plastic is then clamped on top of the panel. When the air in the box is

evacuated, the outside air pressure forces the hot plastic through the opening and forms the concave canopy. It is the surface tension of the plastic that shapes the canopy.

Vacuum Forming With a Female Form

If the shape needed is other than that which would be formed by surface tension, a female mold, or form must be used. It is placed below the plastic sheet and the vacuum pump is connected. When air from the form is evacuated, the outside air pressure forces the hot plastic sheet into the mold and fills it.

SAWING

Several types of saws can be used with transparent plastics; however, circular saws are the best for straight cuts. The blades should be hollow ground or have some set to prevent binding. After the teeth are set, they should be side dressed to produce a smooth edge on the cut. Band saws are recommended for cutting flat acrylic sheets when the cuts must be curved or where the sheet is cut to a rough dimension to be trimmed later. Close control of size and shape may be obtained by band sawing a piece to within 1/16 inch of the desired size, as marked by a scribed line on the plastic, and then sanding it to the correct size with a drum or belt sander.

DRILLING

Unlike soft metal, acrylic plastic is a very poor conductor of heat. Make provisions for removing the heat when drilling. Deep holes need cooling, and water soluble cutting oil is a satisfactory coolant since it has no tendency to attack the plastic.

The drill used on acrylics must be carefully ground and free from nicks and burrs that would affect the surface finish. (***Figure 3-23***) Grind the drill with a greater included angle than would be used for soft metal. The rake angle should be zero in order to scrape, and not cut. The patented Unibit® is good for drilling small holes in aircraft windshields and windows. (***Figure 3-24***) It can cut holes from 1/8 to 1/2 inch in 1/32 inch increments and produces good smooth holes with no stress cracks around their edges.

CEMENTING

Polymerizable cements are those in which a catalyst is added to an already thick monomer-polymer syrup to promote rapid hardening. Cement PS-30® and Weld-On 40® are polymerizable cements of this type. They

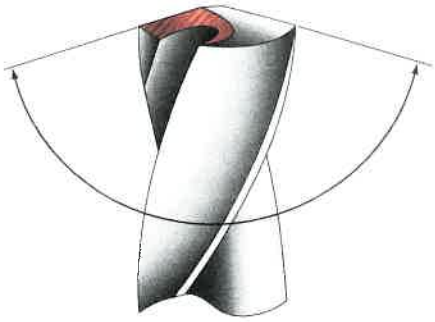


Figure 3-23. A twist drill with an included angle of 150° is used to drill acrylic plastics.

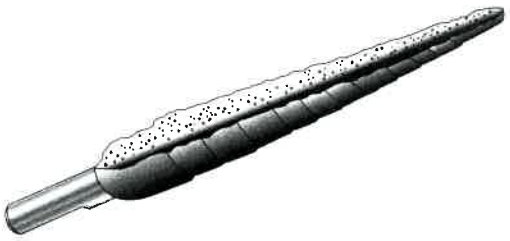


Figure 3-24. Unibit® drill for drilling acrylic plastics.

are suitable for cementing all types of plexiglas acrylic cast sheet and parts molded from plexiglas molding pellets. At room temperature, the cements harden (polymerize) in the container in about 45 minutes after mixing the components.

They harden more rapidly at higher temperatures. The cement joints are usually hard enough for handling within 4 hours after assembly. The joints may be machined within 4 hours after assembly, but it is better to wait 24 hours.

Application of Cement

PS-30® and Weld-On 40® joints retain excellent appearance and color stability after outdoor exposure. These cements produce clear, transparent joints and should be used when the color and appearance of the joints are important. PS-30® and Weld-On 40® should be used at temperatures no lower than 18°C. If cementing is done in a room cooler than 18°C, it requires a longer time to harden and the joint strength is reduced.

The cement should be prepared with the correct proportions of components as given in the manufacturer's instructions and thoroughly mixed, making sure neither the mixing container nor mixing paddle adds color or effects the hardening of the cement. Clean glass or polyethylene mixing containers are preferred. Because of their short pot life (approximately 45 minutes), Cement

PS-30® and Weld-On 40® must be used quickly once the components are mixed. Time consumed in preparation shortens the effective working time, making it necessary to have everything ready to be cemented before the cements are mixed.

For better handling, pour cement within 20 minutes of mixing. For maximum joint strength, the final cement joint should be free of bubbles. It is usually sufficient to allow the mixed cement to stand for 10 minutes before cementing to allow bubbles to rise to the surface. The gap joint technique can only be used with colorless plexiglas acrylic or in cases where joints are hidden. If inconspicuous joints in colored plexiglas acrylic are needed, the parts must be fitted closely, using closed V groove, butt, or arc joints.

Cement forms, or dams, may be made with masking tape as long as the adhesive surface does not contact the cement. This is easily done with a strip of cellophane tape placed over the masking tape adhesive. The tape must be chosen carefully. The adhesive on ordinary cellophane tape prevents the cure of PS-30® and Weld-On 40®. Before actual fabrication of parts, sample joints should be tried to ensure that the tape system used does not harm the cement. Since it is important for all of the cement to remain in the gap, only contact pressure should be used.

Bubbles tend to float to the top of the cement bead in a gap joint after the cement is poured. These cause no problem if the bead is machined off. A small wire (not copper) or similar object may be used to lift some bubbles out of the joint; however, the cement joint should be disturbed as little as possible.

Polymerizable cements shrink as the cement hardens. Therefore, the freshly poured cement bead should be left above the surfaces being cemented to compensate for the shrinkage. If it is necessary for appearances, the bead may be machined off after the cement has set.

REPAIRS

Whenever possible, replace, rather than repair, extensively damaged transparent plastic. A carefully patched part is not the equal of a new section, either optically or structurally. At the first sign of crack development, drill a small hole with a #30 or a ⅛ inch drill at the extreme ends of the cracks. (***Figure 3-25***)

This serves to localize the cracks and to prevent further splitting by distributing the strain over a large area. If the cracks are small, stopping them with drilled holes usually suffices until replacement or more permanent repairs can be made.

CLEANING

Plastics have many advantages over glass for aircraft use, but they lack the surface hardness of glass and care must be exercised while servicing the aircraft to avoid scratching or otherwise damaging the surface. Clean the plastic by washing it with plenty of water and mild soap, using a clean, soft, grit free cloth, sponge, or bare hands. Do not use gasoline, alcohol, benzene, acetone, carbon tetrachloride, fire extinguisher or deicing fluids, lacquer thinners, or window cleaning sprays. These soften the plastic and cause crazing.

Plastics should not be rubbed with a dry cloth since it is likely to cause scratches and to build up an electrostatic charge that attracts dust particles to the surface. If, after removing dirt and grease, no great amount of scratching

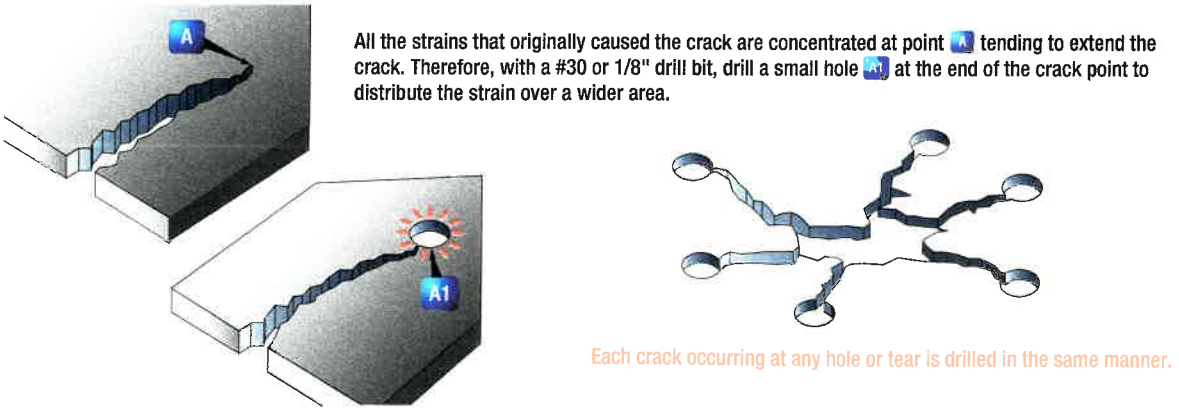


Figure 3-25. Stop drilling of cracks.

is visible, finish the plastic with a good grade of commercial wax. Apply the wax in a thin even coat and bring to a high polish by rubbing lightly with a soft cloth.

POLISHING

Do not attempt hand polishing or buffing until the surface is clean. A soft, open type cotton or flannel buffing wheel is suggested. Minor scratches may be removed by vigorously rubbing the affected area by hand, using a soft clean cloth dampened with a mixture of turpentine and chalk, or by applying automobile cleanser with a damp cloth. Remove the cleaner and polish with a soft, dry cloth. Acrylic and cellulose acetate plastics are thermoplastic. Friction created by buffing or polishing too long in one spot can generate sufficient heat to soften the surface. This condition produces visual distortion and should be avoided.

WINDSHIELD INSTALLATION

Use material equivalent to that originally used by the manufacturer of the aircraft for replacement panels. There are many types of transparent plastics on the market. Their properties vary greatly, particularly expansion characteristics, brittleness under low temperatures, resistance to discoloration when exposed to sunlight, surface checking, etc. These properties are considered by aircraft manufacturer's in selecting materials to be used in their designs and the use of substitutes having different characteristics may result in subsequent difficulties.

Installation Procedures

When installing a replacement panel, use the same mounting method employed by the manufacturer of the aircraft. While the actual installation varies from one type of aircraft to another, consider the following major principles when installing a panel.

1. Never force a plastic panel out of shape to make it fit a frame. If a replacement panel does not fit easily into the mounting, obtain a new replacement or heat the whole panel and re-form. When possible, cut and fit a new panel at ordinary room temperature.
2. In clamping or bolting plastic panels into their mountings, do not place the plastic under excessive compressive stress. It is easy to develop more than 1 000 psi on the plastic by overtorquing a nut and bolt. Tighten each nut to a firm fit, and then back the nut off one full turn (until they are snug and can still be rotated with the fingers).
3. In bolted installations, use spacers, collars, shoulders, or stop-nuts to prevent tightening the bolt excessively. Whenever such devices are used by the aircraft manufacturer, retain them in the replacement installation. It is important that the original number of bolts, complete with washers, spacers, etc., be used. When rivets are used, provide adequate spacers or other satisfactory means to prevent excessive tightening of the frame to the plastic.
4. Mount plastic panels between rubber, cork, or other gasket material to make the installation waterproof, to reduce vibration, and to help to distribute compressive stresses on the plastic.
5. Plastics expand and contract considerably more than the metal channels in which they are mounted. Mount windshield panels to a sufficient depth in the channel to prevent it from falling out when the panel contracts at low temperatures or deforms under load. When the manufacturer's original design permits, mount panels to a minimum depth of 1⅛ inches, and with a clearance of ⅛ inch between the plastic and bottom of the channel.
6. In installations involving bolts or rivets, make the holes through the plastic oversize by ⅛ inch and center so that the plastic does not bind or crack at the edge of the holes. The use of slotted holes is also recommended.

6.3.1(B) - DETECTION OF DEFECTS AND DETERIORATION IN COMPOSITE MATERIAL

MANUFACTURING DEFECTS INCLUDE:

- Delamination
- Resin starved areas
- Resin rich areas
- Thermal decomposition
- Blisters, air bubbles
- Wrinkles
- Voids

Manufacturing damage includes anomalies, such as porosity, microcracking, and delaminations resulting from processing discrepancies. It also includes such items as inadvertent edge cuts, surface gouges and scratches, damaged fastener holes, and impact damage. Examples of flaws occurring in manufacturing include a contaminated bondline surface or inclusions, such as prepreg backing paper or separation film, that is inadvertently left between plies during layup. Inadvertent (nonprocess) damage can occur in detail parts or components during assembly or transport or during operation.

A part is resin rich if too much resin is used. For non-structural applications this is not necessarily bad, but it adds weight. A part is called resin starved if too much resin is bled off during the curing process or if not enough resin is applied during the wet layup process. Resin-starved areas are indicated by fibers that show to the surface. The ratio of 60:40 fiber to resin ratio is considered optimum.

Sources of manufacturing defects include:

- Improper cure or processing
- Improper machining
- Mishandling
- Improper drilling
- Tool drops
- Contamination
- Improper sanding
- Substandard material
- Inadequate tooling
- Mislocation of holes or details

Damage can occur at several scales within the composite material and structural configuration. This ranges from damage in the matrix and fiber to broken elements and failure of bonded or bolted attachments. The extent of damage controls repeated load life and residual strength and is critical to damage tolerance.

FIBER BREAKAGE

Fiber breakage can be critical because structures are typically designed to be fiber dominant (i.e., fibers carry most of the loads). Fortunately, fiber failure is typically limited to a zone near the point of impact and is constrained by the impact object size and energy. Only a few of the service related events listed in the previous section could lead to large areas of fiber damage.

MATRIX IMPERFECTIONS

Matrix imperfections usually occur on the matrix-fiber interface or in the matrix parallel to the fibers. These imperfections can slightly reduce some of the material properties but are seldom critical to the structure, unless the matrix degradation is widespread. Accumulation of matrix cracks can cause the degradation of matrix-dominated properties.

For laminates designed to transmit loads with their fibers (fiber dominant), only a slight reduction of properties is observed when the matrix is severely damaged. Matrix cracks, or microcracks, can significantly reduce properties dependent on the resin or the fiber-resin interface, such as interlaminar shear and compression strength. Microcracking can have a very negative effect on properties of high-temperature resins. Matrix imperfections may develop into delaminations, which are a more critical type of damage.

DELAMINATION AND DEBONDS

Delaminations form on the interface between the layers in the laminate. Delaminations may form from matrix cracks that grow into the interlaminar layer or from low energy impact. Debonds can also form from production non adhesion along the bondline between two elements and initiate delamination in adjacent laminate layers. Under certain conditions, delaminations or debonds can grow when subjected to repeated loading and can cause catastrophic failure when the laminate is loaded in compression. The criticality of delaminations or debonds depend on:

- Dimensions.
- Number of delaminations at a given location.
- Location—in the thickness of laminate, in the structure, proximity to free edges, stress concentration region, geometrical discontinuities

- Loads—behavior of delaminations and debonds depend on loading type. They have little effect on the response of laminates loaded in tension. Under compression or shear loading, however, the sublaminates adjacent to the delaminations or debonded elements may buckle and cause a load redistribution mechanism that leads to structural failure.

COMBINATIONS OF DAMAGES

In general, impact events cause combinations of damages. High energy impacts by large objects (e.g., turbine blades) may lead to broken elements and failed attachments. The resulting damage may include significant fiber failure, matrix cracking, delamination, broken fasteners, and debonded elements. Damage caused by low-energy impact is more contained, but may also include a combination of broken fibers, matrix cracks, and multiple delaminations.

FLAWED FASTENER HOLES

Improper hole drilling, poor fastener installation, and missing fasteners may occur in manufacturing. Hole elongation can occur due to repeated load cycling in service.

IN-SERVICE DEFECTS

In-service defects include:

- Environmental degradation
- Impact damage
- Fatigue
- Cracks from local overload
- Debonding
- Delamination
- Fiber fracturing
- Erosion

Many honeycomb structures, such as wing spoilers, fairings, flight controls, and landing gear doors, have thin face sheets which have experienced durability problems that could be grouped into three categories: low resistance to impact, liquid ingression, and erosion. These structures have adequate stiffness and strength but low resistance to a service environment in which parts are crawled over, tools dropped, and service personnel are often unaware of the fragility of thin skinned sandwich parts. Damages to these components, such as core crush, impact damages, and disbonds, are quite often easy to detect with a visual inspection due to their thin face sheets. However, they are sometimes overlooked or damaged by service personnel who do not want to delay aircraft departure or bring attention to their accidents, which might reflect poorly on their performance record. Therefore, damages are sometimes allowed to go unchecked, often resulting in growth of the damage due to liquid ingression into the core. Non-durable design details (e.g., improper core edge close-outs) also lead to liquid ingression.

The repair of parts due to liquid ingression can vary depending on the liquid, most commonly water or Skydrol® (hydraulic fluid). Water tends to create additional damage in repaired parts when cured unless all moisture is removed from the part. Most repair material systems cure at temperatures above the boiling point of water, which can cause a disbond at the skin-to-core interface wherever trapped water resides. For this reason, core drying cycles are typically included prior to performing any repair. Some operators take the extra step of placing a damaged but unrepaired part in the autoclave to dry to preclude any additional damage from occurring during the cure of the repair. Skydrol® presents a different problem. Once the core of a sandwich part is saturated, complete removal of Skydrol® is almost impossible. The part continues to weep the liquid even in cure until bondlines can become contaminated and full bonding does not occur. Removal of contaminated core and adhesive as part of the repair is highly recommended. (***Figure 3-26***)

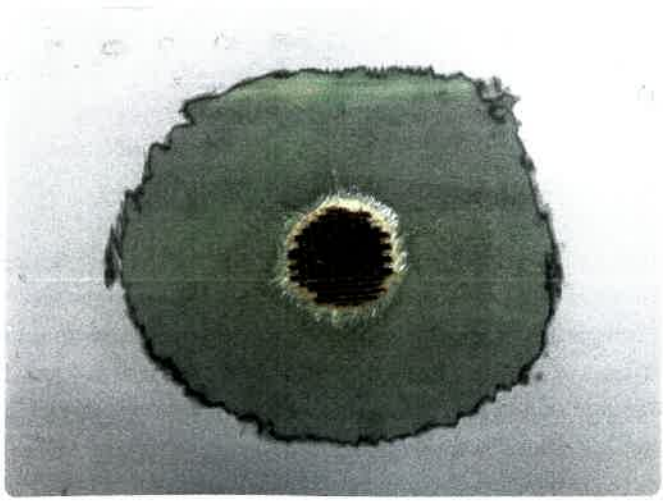


Figure 3-26. Damage to fuselage skin, prepped for core replacements and layup.

Erosion capabilities of composite materials have been known to be less than that of aluminum and, as a result, their application in leading-edge surfaces has been generally avoided. However, composites have been used in areas of highly complex geometry, but generally with an erosion coating. The durability and maintainability of some erosion coatings are less than ideal. Another problem, not as obvious as the first, is that edges of doors or panels can erode if they are exposed to the air stream. This erosion can be attributed to improper design or installation/fit-up. On the other hand, metal structures in contact or in the vicinity of these composite parts may show corrosion damage due to inappropriate choice of aluminum alloy, damaged corrosion sealant of metal parts during assembly or at splices, or insufficient sealant and/or lack of glass fabric isolation plies at the interfaces of spars, ribs, and fittings. (***Figure 3-27***)

CORROSION

Many fiberglass and Kevlar® parts have a fine aluminum mesh for lightning protection. This aluminum mesh often corrodes around the bolt or screw holes. The corrosion affects the electrical bonding of the panel, and the aluminum mesh needs to be removed and new mesh installed to restore the electrical bonding of the panel. (***Figure 3-28***)

Ultraviolet (UV) light affects the strength of composite materials. Composite structures need to be protected by a top coating to prevent the effects of UV light. Special UV primers and paints have been developed to protect composite materials.

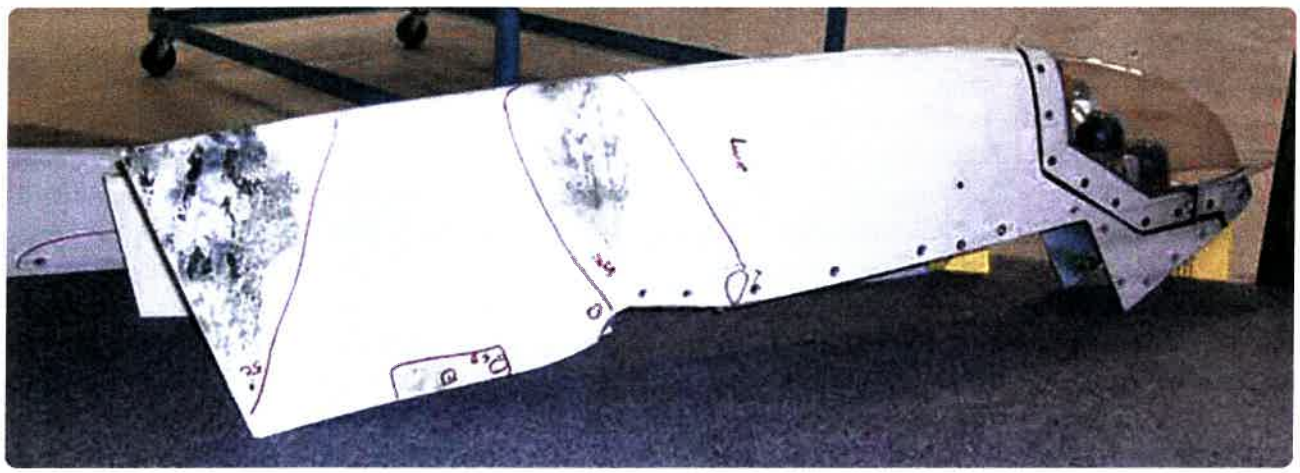


Figure 3-27. Erosion damage to wingtip.

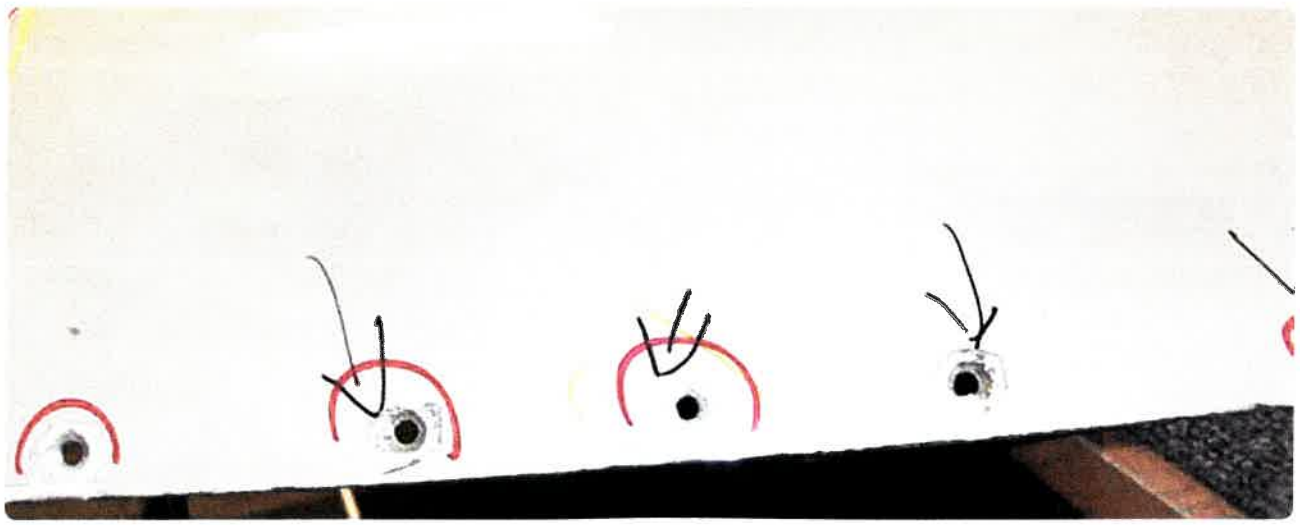


Figure 3-28. Corrosion of aluminum lightning protection mesh.

NON-DESTRUCTIVE INSPECTION OF COMPOSITES

Visual Inspection

A visual inspection is the primary inspection method for inservice inspections. Most types of damage scorch, stain, dent, penetrate, abrade, or chip the composite surface, making the damage visible. Once damage is detected, the affected area needs to be inspected closer using flashlights, magnifying glasses, mirrors, and borescopes. These tools are used to magnify defects that otherwise might not be seen easily and to allow visual inspection of areas that are not readily accessible. Resin starvation, resin richness, wrinkles, ply bridging, discoloration (due to overheating, lightning strike, etc.), impact damage by any cause, foreign matter, blisters, and disbonding are some of the discrepancies that can be detected with a visual inspection. Visual inspection cannot find internal flaws in the composite, such as delaminations, disbonds, and matrix crazing. More sophisticated NDI techniques are needed to detect these types of defects.

Audible Sonic Testing (Coin Tapping)

Sometimes referred to as audio, sonic, or coin tap, this technique makes use of frequencies in the audible range (10 Hz to 20 Hz). A surprisingly accurate method in the hands of experienced personnel, tap testing is perhaps the most common technique used for the detection of delamination and/or disbond. The method is accomplished by tapping the inspection area with a solid round disk or lightweight hammer-like device and listening to the response of the structure to the hammer. (***Figure 3-29***)

A clear, sharp, ringing sound is indicative of a well bonded solid structure, while a dull or thud like sound indicates a discrepant area.

The tapping rate needs to be rapid enough to produce enough sound for any difference in sound tone to be discernable to the ear. Tap testing is effective on thin skin to stiffener bondlines, honeycomb sandwich with thin face sheets, or even near the surface of thick laminates, such as rotorcraft blade supports. Again, inherent in the method is the possibility that changes within the internal elements of the structure might produce pitch changes that are interpreted as defects, when in fact they are present by design. This inspection should be accomplished in as quiet an area as possible and by experienced personnel familiar with the part's internal configuration. This method is not reliable for structures with more than four plies. It is often used to map out the damage on thin honeycomb facesheets. (***Figure 3-29***)

Automated Tap Test

This test is very similar to the manual tap test except that a solenoid is used instead of a hammer. The solenoid produces multiple impacts in a single area. The tip of the impactor has a transducer that records the force versus time signal of the impactor. The magnitude of the force depends on the impactor, the impact energy, and the mechanical properties of the structure. The impact duration (period) is not sensitive to the magnitude of the impact force; however, this duration changes as the stiffness of the structure is altered. Therefore, the signal from an unflawed region is used for calibration, and any deviation from this unflawed signal indicates the existence of damage.

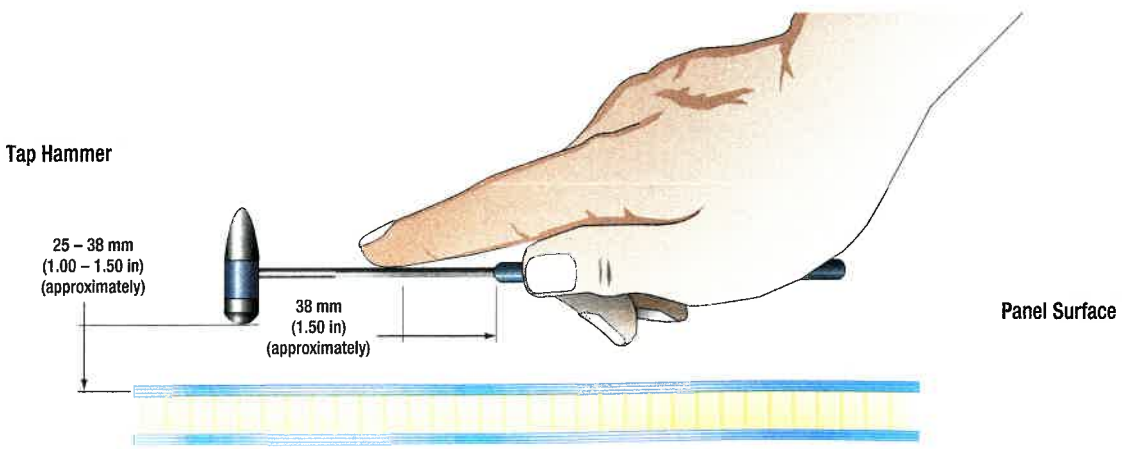


Figure 3-29. Tap test with tap hammer.

Ultrasonic Inspection

Ultrasonic inspection has proven to be a very useful tool for the detection of internal delaminations, voids, or inconsistencies in composite components not otherwise discernible using visual or tap methodology. There are many ultrasonic techniques; however, each technique uses sound wave energy with a frequency above the audible range. (***Figure 3-30***)

A high-frequency (usually several MHz) sound wave is introduced into the part and may be directed to travel normal to the part surface, or along the surface of the part, or at some predefined angle to the part surface. You may need to try different directions to locate the flow. The introduced sound is then monitored as it travels its assigned route through the part for any significant change. Ultrasonic sound waves have properties similar to light waves. When an ultrasonic wave strikes an interrupting object, the wave or energy is either absorbed or reflected back to the surface. The disrupted or diminished sonic energy is then picked up by a receiving transducer and converted into a display on an oscilloscope or a chart recorder. The display allows the operator to evaluate the discrepant indications comparatively with those areas known to be good. To facilitate the comparison, reference standards are established and utilized to calibrate the ultrasonic equipment.

The repair technician must realize that the concepts outlined here work fine in the repetitious manufacturing environment, but are likely to be more difficult to implement in a repair environment given the vast number of different composite components installed on the aircraft and the relative complexity of their construction. The reference standards would also have to take into account the transmutations that take place when a composite component is exposed to an in-service environment over a prolonged period or has been the subject of repair activity or similar restorative action. The four most common ultrasonic techniques are discussed next.

Through Transmission Ultrasonic Inspection

Through transmission ultrasonic inspection uses two transducers, one on each side of the area to be inspected. The ultrasonic signal is transmitted from one transducer to the other transducer. The loss of signal strength is then measured by the instrument. The instrument shows the loss as a percent of the original signal strength or the loss in decibels. The signal loss is compared to a reference standard. Areas with a greater loss than the reference standard indicate a defective area.

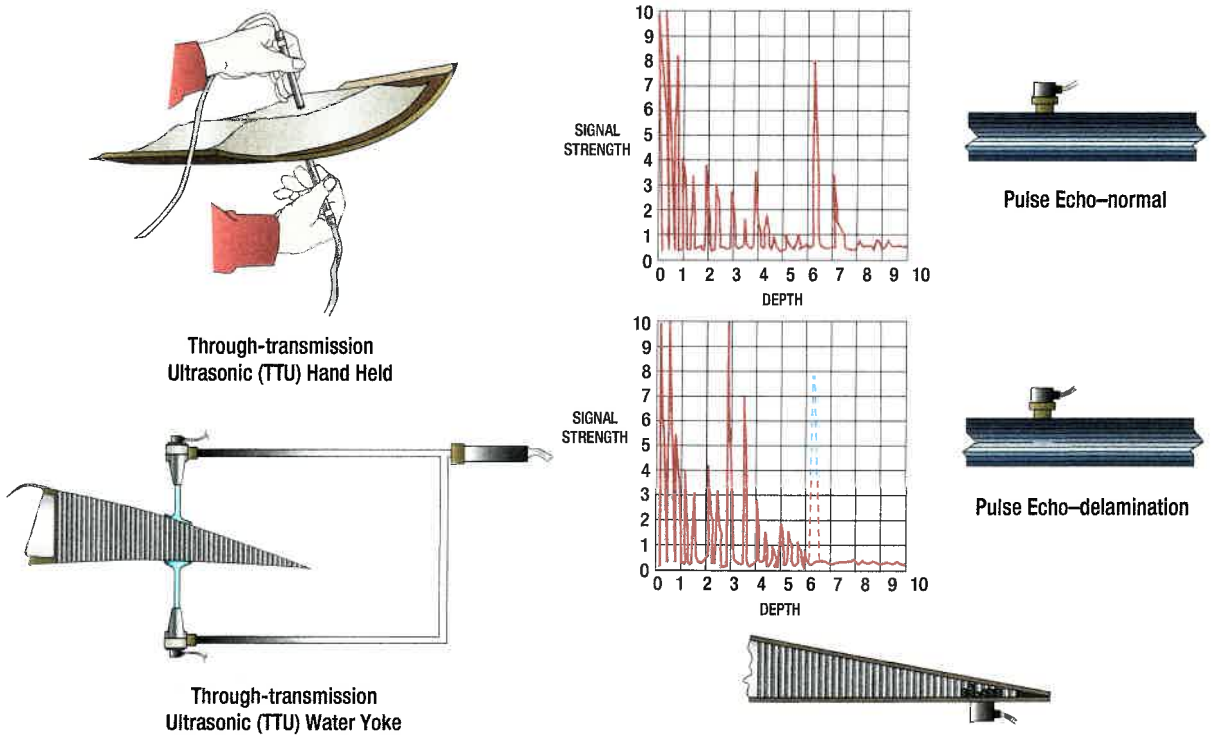


Figure 3-30. Ultrasonic testing methods.

Pulse Echo Ultrasonic Inspection

Single-side ultrasonic inspection may be performed using pulse echo techniques. In this method, a single search unit is working as a transmitting and a receiving transducer that is excited by high voltage pulses. Each electrical pulse activates the transducer element. This element converts the electrical energy into mechanical energy in the form of an ultrasonic sound wave. The sonic energy travels through a Teflon® or methacrylate contact tip into the test part. A waveform is generated in the test part and is picked up by the transducer element. Any change in amplitude of the received signal, or time required for the echo to return to the transducer, indicates the presence of a defect. Pulse echo inspections are used to find delaminations, cracks, porosity, water, and disbonds of bonded components. Pulse echo does not find disbonds or defects between laminated skins and honeycomb core. (***Figure 3-31***)



Figure 3-31. Pulse echo test equipment.

Ultrasonic Bond Tester Inspection

Low frequency and high frequency bondtesters are used for ultrasonic inspections of composite structures. These bondtesters use an inspection probe that has one or two transducers. The high frequency bondtester is used to detect delaminations and voids. It cannot detect a skin-to honeycomb core disbond or porosity. It can detect defects as small as 0.5 inch in diameter. The low frequency bondtester uses two transducers and is used to detect delamination, voids, and skin to honeycomb core disbands. This inspection method does not detect which side of the part is damaged, and cannot detect defects smaller than 1.0 inch. (***Figure 3-32***)



Figure 3-32. Bond tester.

Phased Array Inspection

Phased array inspection is one of the latest ultrasonic instruments to detect flaws in composite structures. It operates under the same principle of operation as pulse echo, but it uses 64 sensors at the same time, which speeds up the process. (***Figure 3-33***)



Figure 3-33. Phased array testing equipment.

Radiography

Radiography, often referred to as X-ray, is a very useful NDI method because it essentially allows a view into the interior of the part. This inspection method is accomplished by passing X-rays through the part or assembly being tested while recording the absorption of the rays onto a film sensitive to X-rays. The exposed film, when developed, allows the inspector to analyze variations in the opacity of the exposure recorded

onto the film, in effect creating a visualization of the relationship of the component's internal details. Since the method records changes in total density through its thickness, it is not a preferred method for detecting defects such as delaminations that are in a plane that is normal to the ray direction. It is a most effective method, however, for detecting flaws parallel to the X-ray beam's centerline. Internal anomalies, such as delaminations in the corners, crushed core, blown core, water in core cells, voids in foam adhesive joints, and relative position of internal details, can readily be seen via radiography.

Most composites are nearly transparent to X-rays, so low energy rays must be used. Because of safety concerns, it is impractical to use around aircraft. Operators should always be protected by sufficient lead shields, as the possibility of exposure exists either from the X-ray tube or from scattered radiation. Maintaining a minimum safe distance from the X-ray source is always essential.

Thermography

Thermal inspection comprises all methods in which heat sensing devices are used to measure temperature variations for parts under inspection. The basic principle of thermal inspection consists of measuring or mapping of surface temperatures when heat flows from, to, or through a test object. All thermographic techniques rely on differentials in thermal conductivity between normal, defect free areas, and those having a defect. Normally, a heat source is used to elevate the temperature of the part being examined while observing the surface heating effects. Because defect free areas conduct heat more efficiently than areas with defects, the amount of heat that is either absorbed or reflected indicates the quality of the bond. The type of defects that affect the thermal properties include debonds, cracks, impact damage, panel thinning, and water ingress into composite materials and honeycomb core. Thermal methods are most effective for thin laminates or for defects near the surface.

Neutron Radiography

Neutron radiography is a nondestructive imaging technique that is capable of visualizing the internal characteristics of a sample. The transmission of neutrons through a medium is dependent upon the neutron cross sections for the nuclei in the medium. Differential attenuation of neutrons through a medium may be measured, mapped, and then visualized. The resulting image may then be utilized to analyze internal characteristics of the sample. Neutron radiography is a complementary technique to X-ray radiography. Both techniques visualize the attenuation through a medium. The major advantage of neutron radiography is its ability to reveal light elements such as hydrogen found in corrosion products and water.

Moisture Detector

A moisture meter can be used to detect water in sandwich honeycomb structures. This meter measures the RF power loss caused by the presence of water. The moisture meter is often used to detect moisture in nose radomes. (***Figure 3-34***) ***Figure 3-35*** provides a comparison of NDI testing equipment.



Figure 3-34. Moisture tester equipment.

REPAIR OF COMPOSITE MATERIAL

LAYUP MATERIALS AND TOOLS

Prepreg and dry fabrics can be cut with hand tools, such as scissors, pizza cutters, and knives. Materials made from Kevlar® are more difficult to cut than fiberglass or carbon and tools wear quicker. A squeegee and a brush are used to impregnate dry fibers with resin for wet layup. Markers, rulers, and circle templates are used to make a repair layout. (***Figure 3-36***)

Air Tools

Air-driven power tools, such as drill motors, routers, and grinders, are used for composite materials. Electric motors are not recommended, because carbon is a

Method of Inspection	Type of Defect							
	Disbond	Delamination	Dent	Crack	Hole	Water Ingestion	Overheat and Burns	Lightning Strike
Visual	X (1)	X (1)	X	X	X		X	X
X-Ray	X (1)	X (1)		X (1)		X		
Ultrasonic TTU	X	X						
Ultrasonic pulse echo		X				X		
Ultrasonic bondtester	X	X						
Tap test	X (2)	X (2)						
Infrared thermography	X (3)	X (3)				X		
Dye penetrant				X (4)				
Eddy current				X (4)				
Shearography	X (3)	X (3)						

Notes:
(1) For defects that open to the surface
(2) For thin structure (3 plies or less)
(3) The procedures for this type of inspection are being developed
(4) This procedure is not recommended

Figure 3-35. Moisture tester equipment.

conductive material that can cause an electrical short circuit. If electric tools are used, they need to be of the totally enclosed type. (***Figure 3-37***)

Caul Plate

A caul plate made from aluminum is often used to support the part during the cure cycle. A mold release agent, or parting film, is applied to the caul plate so that the part does not attach to the caul plate. A thin caul plate is also used on top of the repair when a heat bonder is used. The caul plate provides a more uniform heated area and it leaves a smoother finish of the composite laminate.

Support Tooling and Molds

Certain repairs require tools to support the part and/or maintain surface contour during cure. A variety of materials can be used to manufacture these tools.

The type of material depends on the type of repair, cure temperature, and whether it is a temporary or permanent tool. Support tooling is necessary for oven and autoclave cure due to the high cure temperature. The parts deform if support tooling is not used. There are many types of tooling material available. Some are molded to a specific part contour and others are used as rigid supports to maintain the contour during cure.

Plaster is an inexpensive and easy material for contour tooling. It can be filled with fiberglass, hemp, or other material. Plaster is not very durable, but can be used for



Figure 3-36. Hand tools for laminating.



Figure 3-37. Air tools used for composite repair.

temporary tools. Often, a layer of fiberglass reinforced epoxy is placed on the tool side surface to improve the finish quality. Tooling resins are used to impregnate fiberglass, carbon fiber, or other reinforcements to make permanent tools. Complex parts are made from metal or high-temperature tooling boards that are machined with 5-axis CNC equipment to make master tools that can be used to fabricate aircraft parts. (***Figures 3-38 and 3-39***)

VACUUM BAG MATERIALS

Repairs of composite aircraft components are often performed with a technique known as vacuum bagging. A plastic bag is sealed around the repair area. Air is then removed from the bag, which allows repair plies to be drawn together with no air trapped in between. Atmospheric pressure bears on the repair and a strong, secure bond is created.

Several processing materials are used for vacuum bagging a part. These materials do not become part of the repair and are discarded after the repair process.

Release Agents

Release agents, also called mold release agents, are used so that the part comes off the tool or caul plate easily after curing.

Bleeder Ply

The bleeder ply creates a path for the air and volatiles to escape from the repair. Excess resin is collected in the bleeder. Bleeder material could be made of a layer of fiberglass, nonwoven polyester, or it could be a perforated Teflon® coated material. The Structural Repair Manual (SRM) indicates what type and how many plies of bleeder are required. As a general rule, the thicker the laminate, the more bleeder plies are required.

Peel Ply

Peel plies are often used to create a clean surface for bonding purposes. A thin layer of fiberglass is cured with the repair part. Just before the part is bonded to another structure, the peel ply is removed. The peel ply is easy to remove and leaves a clean surface for bonding. Peel plies are manufactured from polyester, nylon, Flouronated Ethylene Propylene (FEP), or coated fiberglass. They can be difficult to remove if overheated. Some coated peel plies can leave an undesirable contamination on the surface. The preferred peel ply material is polyester that has been heat-set to eliminate shrinkage.



Figure 3-38. Five-axis CNC equipment for tool and mold making.



Figure 3-39. A mold of an inlet duct.

Layup Tapes

Vacuum bag sealing tape, also called sticky tape, is used to seal the vacuum bag to the part or tool. Always check the temperature rating of the tape before use to ensure that you use appropriately rated tape.

Perforated Release Film

Perforated parting film is used to allow air and volatiles out of the repair, and it prevents the bleeder ply from sticking to the part or repair. It is available with different size holes and hole spacing depending on the amount of bleeding required.

Solid Release Film

Solid release films are used so that the prepreg or wet layup plies do not stick to the working surface or caul plate. Solid release film is also used to prevent the resins from bleeding through and damaging the heat blanket or caul plate if they are used.

Breather Material

The breather material is used to provide a path for air to get out of the vacuum bag. The breather must contact the bleeder. Typically, polyester is used in either 4 ounce or 10 ounce weights. Four ounces is used for applications below 50 pounds per square inch (psi) and 10 ounces is used for 50–100 psi.

Vacuum Bag

The vacuum bag material provides a tough layer between the repair and the atmosphere. The vacuum bag material is available in different temperature ratings, so make sure that the material used for the repair can handle the cure temperature. Most vacuum bag materials are one time use, but material made from flexible silicon rubber is reusable. Two small cuts are made in the bagging material so that the vacuum probe valve can be installed. The vacuum bag is not very flexible and plies need to be made in the bag if complex shapes are to be bagged. Sometimes, an envelope type bag is used, but the disadvantage of this method is that the vacuum pressure might crush the part. Reusable bags made from silicon rubber are available that are more flexible. Some have a built-in heater blanket that simplifies the bagging task. (***Figures 3-40, 3-41, and 3-42***)

VACUUM EQUIPMENT

A vacuum pump is used to evacuate air and volatiles from the vacuum bag so that atmospheric pressure consolidates the plies. A dedicated vacuum pump is used in a repair shop. For repairs on the aircraft, a mobile vacuum pump could be used. Most heat bonders have a built-in vacuum pump. Special air hoses are used as vacuum lines, because regular air hoses might collapse when a vacuum is applied. The vacuum lines that are used in the oven or autoclave need to be able to withstand the high temperatures in the heating device. A vacuum pressure regulator is sometimes used to lower the vacuum pressure during the bagging process.

Vacuum Compaction Table

A vacuum compaction table is a convenient tool for debulking composite layups with multiple plies. Essentially a reusable vacuum bag, a compaction table consists of a metal table surface with a hinged cover. The cover includes a solid frame, a flexible membrane, and a vacuum seal. Repair plies are laid up on the table surface and sealed beneath the cover with vacuum to remove entrapped air. Some compaction tables are heated.



Figure 3-40. Bagging materials.



Figure 3-41. Bagging of complex part.

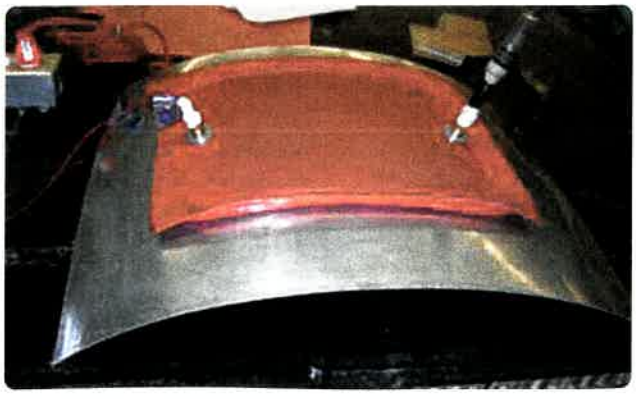


Figure 3-42. Self-sealing vacuum bag with heater element.

Oven

Composite materials can be cured in ovens using various pressure application methods. (***Figure 3-43***) Typically, vacuum bagging is used to remove volatiles and trapped air and utilizes atmospheric pressure for consolidation. Another method of pressure application for oven cures is the use of shrink wrapping or shrink tape. The oven uses heated air circulated at high speed to cure the material system. Typical oven cure temperatures are 120°C and 175°C. Ovens have a temperature sensor to feed temperature data back to the oven controller. The oven temperature can differ from the actual part temperature depending upon the location of the oven sensor and the location of the part in the oven. The thermal mass of the part in the oven is generally greater than the surrounding oven and during rise to temperature, the part temperature can lag the oven temperature by a considerable amount.



Figure 3-43. Walk-in curing oven.

To deal with these differences, at least two thermocouples must be placed on the part and connected to a temperature sensing device (separate chart recorder, hot bonder, etc.) located outside the oven. Some oven controllers can be controlled by thermocouples placed on the repair part.

Autoclave

An autoclave system allows a complex chemical reaction to occur inside a pressure vessel according to a specified time, temperature, and pressure profile in order to process a variety of materials. (***Figure 3-44***) The evolution of materials and processes has taken autoclave operating conditions from 120°C (250°F) and 275 kPa (40 psi) to well over 760°C (1 400°F) and 69 000 kPa (10 000 psi). Autoclaves that are operated at lower temperatures and pressures can be pressurized by air, but if higher temperatures and pressures are required for cure cycle, a 50/50 mixture of air and nitrogen or 100 percent nitrogen should be used to reduce the chance of an autoclave fire.

The major elements of an autoclave system are a vessel to contain pressure, sources to heat the gas stream and circulate it uniformly within the vessel, a subsystem to apply vacuum to parts covered by a vacuum bag, a subsystem to control operating parameters, and a subsystem to load the molds into the autoclave. Modern autoclaves are computer controlled and the operator can

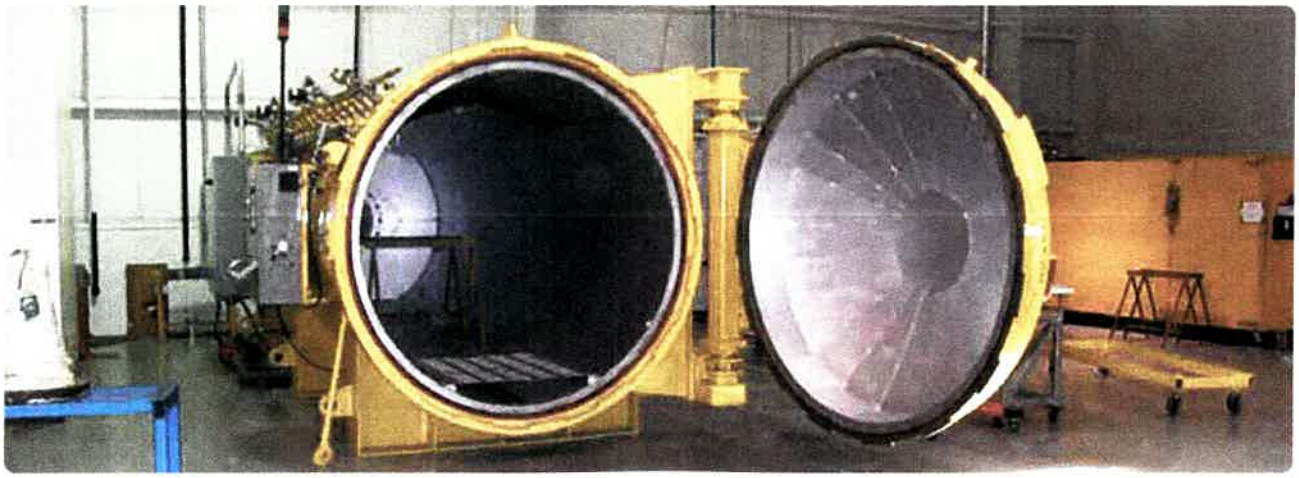


Figure 3-44. Autoclave.

write and monitor all types of cure cycle programs. The most accurate way to control the cure cycle is to control the autoclave controller with thermocouples that are placed on the actual part.

Most parts processed in autoclaves are covered with a vacuum bag that is used primarily for compaction of laminates and to provide a path for removal of volatiles. The bag allows the part to be subjected to differential pressure in the autoclave without being directly exposed to the autoclave atmosphere. The vacuum bag is also used to apply varying levels of vacuum to the part.

On Aircraft Heating

Typical on-aircraft heating methods include electrical resistance heat blankets, infrared heat lamps, and hot air devices. All heating devices must be controlled by some means so that the correct amount of heat can be applied. This is particularly important for repairs using prepreg material and adhesives, because controlled heating and cooling rates are usually prescribed.

Heat Bonder

A heat bonder is a portable device that automatically controls heating based on temperature feedback from the repair area. Heat bonders also have a vacuum pump that supplies and monitors the vacuum in the vacuum bag. The heat bonder controls the cure cycle with thermocouples that are placed near the repair. Some repairs require up to 10 thermocouples. Modern heat bonders can run many different types of cure programs and cure cycle data can be printed out or uploaded to a computer. (***Figure 3-45***)



Figure 3-45. Heat bonder equipment.

Heat Blanket

A heat blanket is a flexible heater. It is made of two layers of silicon rubber with a metal resistance heater between the two layers of silicon. Heat blankets are a common method of applying heat for repairs on the aircraft. Heat blankets may be controlled manually; however, they are usually used in conjunction with a heat bonder. Heat is transferred from the blanket via conduction. Consequently, the heat blanket must conform to and be in 100 percent contact with the part, which is usually accomplished using vacuum bag pressure. (***Figure 3-46***)

Heat Lamp

Infrared heat lamps can also be used for elevated temperature curing of composites if a vacuum bag is not utilized. However, they are generally not effective for producing curing temperatures above 65°C, or for areas larger than two square feet. It is also difficult to control the heat applied with a lamp, and lamps tend to generate high surface temperatures quickly. If controlled by thermostats, heat lamps can be useful in applying curing heat to large or irregular surfaces. Heat bonders can be used to control heat lamps.

Hot Air System

Hot air systems can be used to cure composite repairs, and are mainly restricted to small repairs and for drying the repair area. A heat generator supplies hot air that is directed into an insulated enclosure set up around the repair area after vacuum bagging has been deployed. The hot air surrounds the repair for even temperature rise.

HEAT PRESS FORMING

During the press forming process, flat stacked thermoplastic prepreg is heated above melt temperature

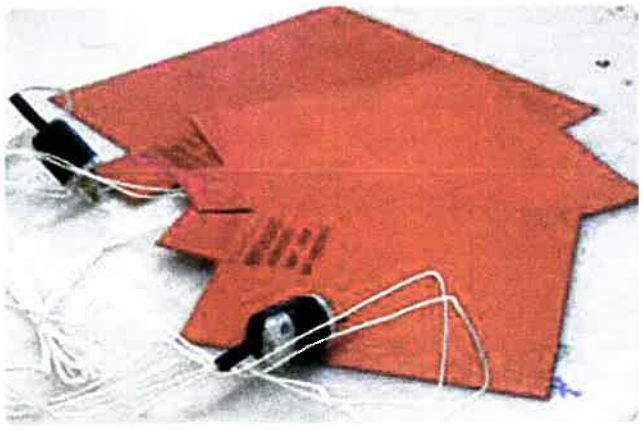


Figure 3-46. Heat blankets.

(340–430°C) in an oven, rapidly (1–10 seconds) shuttled to a forming die, pressed to shape, and consolidated and cooled under pressure (700–7 000 kPa, or 100–1 000 psi). (***Figure 3-47***)

In production, press forming dies usually are matched male-female sets constructed of steel or aluminum. However, rubber, wood, phenolics, and so on can be used during prototyping. The die set can be maintained at room temperature throughout the forming consolidation cycle. But, the use of a hot die 120–200°C allows control of the cooling down rate (avoiding part warpage and controlling morphology in semicrystalline thermoplastic prepreg, such as PEEK and polyphenylene sulfide) and extends the forming window promoting better ply slip.

The main disadvantage with this method is that the press only applies pressure in one direction, and hence, it is difficult to make complex shaped (e.g., beads, closed corners) parts or parts with legs that approach vertical. Since the temperature of the die set need not be cycled with each part, rapid forming times of between 10 minutes and 2 hours are achievable with press forming.



Figure 3-47. Heat press.

THERMOCOUPLES

A Thermocouple (TC) is a thermoelectric device used to accurately measure temperatures. It may be connected to a simple temperature reading device, or connected to a hot bonder, oven, or other type of controller that regulates the amount of heat. TCs consist of a wire with two leads of dissimilar metals that are joined at one end. Heating the joint produces an electric current, which is converted to a temperature reading with a TC monitor. Select the type of wire (J or K) and the type of connector that are compatible with the local temperature monitoring equipment (hot bonder, oven, autoclave, etc.). TC wire is available with different types of insulation; check the manufacturer's product data sheets to ensure the insulation withstands the highest cure temperature. Teflon insulated wire is generally good for 200°C and lower cures; Kapton-insulated wire should be used for higher temperatures.

Thermocouple Placement

Thermocouple placement is the key in obtaining proper cure temperatures throughout the repair. In general, the thermocouples used for temperature control should be placed as close as possible to the repair material without causing it to become embedded in the repair or producing indentations in the repair. They should also be placed in strategic hot or cold locations to ensure the materials are adequately cured but not exposed to excessively high temperatures that could degrade the material structural properties. The thermocouples should be placed as close as practical to the area that needs to be monitored. The following steps should be taken when using thermocouples:

- Never use fewer than three thermocouples to monitor a heating cycle.
- If bonding a precured patch, place the thermocouple near the center of the patch.
- A control thermocouple may be centered over a low temperature (95°C or lower) co-cured patch as long as it is placed on top of a thin metallic sheet to prevent a thermocouple indentation onto the patch. This may allow for a more accurate control of the patch temperature.
- The thermocouples installed around the perimeter of the repair patch should be placed approximately 0.5 inch away from the edge of the adhesive line.
- Place flash tape below and above the thermocouple tips to protect them from resin flash and to protect the control unit from electrical shorts.
- Do not place the thermocouple under the vacuum port as the pressure may damage the lead and cause erroneous readings to occur.
- Do not place thermocouple wires adjacent to or crossing the heat blanket power cord to prevent erroneous temperature readings caused by magnetic flux lines.

- Do not place any control thermocouple beyond the heat blanket's two inch overlap of the repair to prevent the controller from trying to compensate for the lower temperature.
- Always leave slack in the thermocouple wire under the vacuum bag to prevent the thermocouple from being pulled away from the area to be monitored as vacuum is applied.

THERMAL SURVEY OF REPAIR AREA

In order to achieve maximum structural bonded composite repair, it is essential to cure these materials within the recommended temperature range. Failure to cure at the correct temperatures can produce weak patches and/or bonding surfaces and can result in a repair failure during service. A thermal survey should be performed prior to installing the repair to ensure proper and uniform temperatures can be achieved. The thermal survey determines the heating and insulation requirements, as well as thermocouple locations for the repair area. The thermal survey is especially useful for determining the methods of heating (hot air modules, heat lamps, heat blanket method and monitoring requirements in cases where heat sinks (substructure for instance) exist in the repair area). It should be performed for all types of heating methods to preclude insufficient, excessive, or uneven heating of the repair area.

Temperature Variations in Repair Zone

Thermal variations in the repair area occur for many reasons. Primary among these are material type, material thickness, and underlying structure in the repair zone. For these reasons, it is important to know the structural composition of the area to be repaired. Substructure existing in the repair zone conducts heat away from the repair area, resulting in a cold spot directly above the structure. Thin skins heat quickly and can easily be overheated. Thick skin sections absorb heat slowly and take longer to reach soak temperature. The thermal survey identifies these problem areas and allows the technician to develop the heat and insulation setup required for even heating of the repair area.

Thermal Survey

During the thermal survey process, try to determine possible hot and cold areas in the repair zone. Temporarily attach a patch of the same material and thickness, several thermal couples, heating blanket, and a vacuum bag to the repair area. Heat the area and, after the temperature is stabilized, record the thermocouple temperatures. Add insulation if the temperature of the thermocouple varies more than 10 degrees from average. The areas with a stringer and rib indicate a lower temperature than the middle of the patch because they act as a heat sink. Add insulation to these areas to increase the temperature. (***Figure 3-48***)

Solutions to Heat Sink Problems

Additional insulation can be placed over the repair area. This insulation can also be extended beyond the repair area to minimize heat being conducted away. Breather materials and fiberglass cloths work well, either on top of the vacuum bag or within the vacuum bag or on the accessible backside of the structure. Place more insulation over cool spots and less insulation over hot spots. If access is available to the backside of the repair area, additional heat blankets could be placed there to heat more evenly.

TYPES OF LAYUPS

Wet Layups

During the wet layup process, a dry fabric is impregnated with a resin. Mix the resin system just before making the repair. Lay out the repair plies on a piece of fabric and impregnate the fabric with the resin. After the fabric is impregnated, cut the repair plies, stack in the correct ply orientation, and vacuum bag. Wet layup repairs are often used with fiberglass for nonstructural applications. Carbon and Kevlar® dry fabric could also be used with a wet layup resin system. Many resin systems used with wet layup cure at room temperature, are easy to accomplish, and the materials can be stored at room temperature for long period of times. The disadvantage of room temperature wet layup is that it does not restore the strength and durability of the original structure and parts that were cured at 120°C or 200°C during manufacturing. Some wet layup resins use an elevated temperature cure and have improved properties. In general, wet layup properties are less than properties of prepreg material.

Epoxy resins may require refrigeration until they are used. This prevents the aging of the epoxy. The label on the container states the correct storage temperature for each component. The typical storage temperature is between 4°C and 27°C for most epoxy resins. Some resin systems require storage below 4°C.

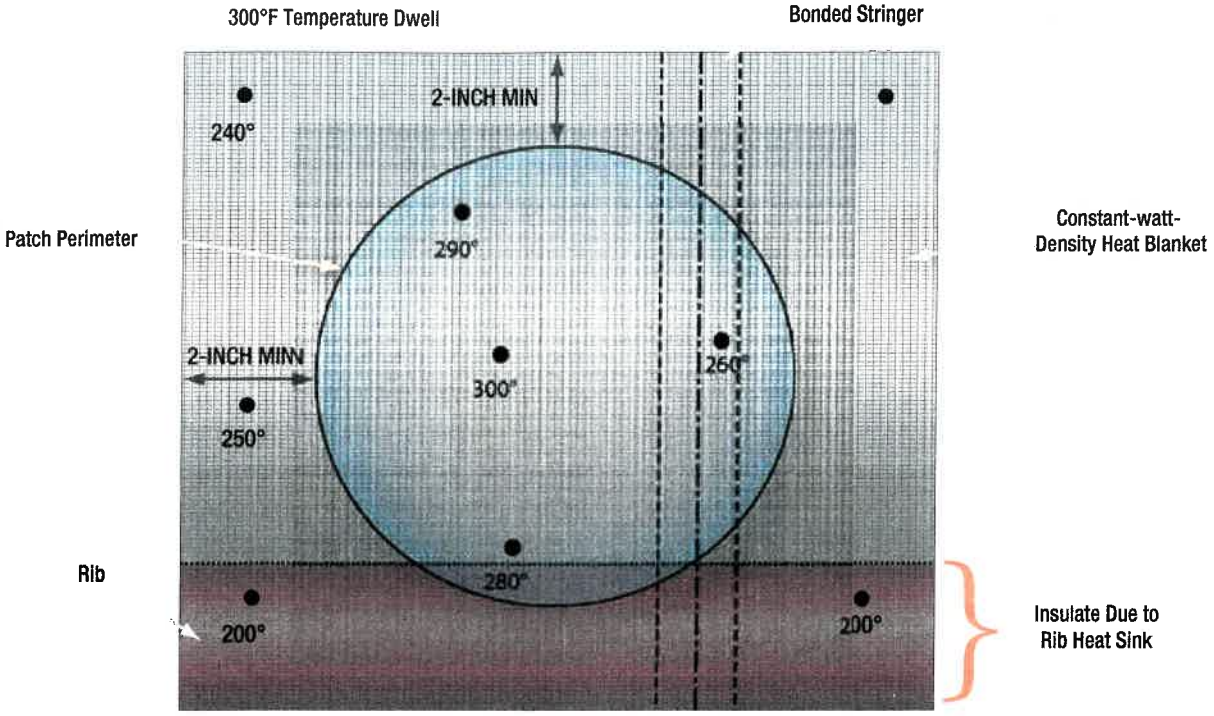


Figure 3-48. Thermal survey example.

Prepregs

Prepreg is a fabric or tape that is impregnated with a resin during the manufacturing process. The resin system is already mixed and is in the B stage cure. Store the prepreg material in a freezer below -18°C to prevent further curing of the resin. The material is typically placed on a roll and a backing material is placed on one side of the material so that the prepreg does not stick together. The prepreg material is sticky and adheres to other plies easily during the stack-up process. You must remove the prepreg from the freezer and let the material thaw, which might take 8 hours for a full roll. Store the prepreg materials in a sealed, moisture proof bag. Do not open these bags until the material is completely thawed, to prevent contamination of the material by moisture. After the material is thawed and removed from the backing material, cut it in repair plies, stack in the correct ply orientation, and vacuum bag. Do not forget to remove the backing material when stacking the plies. Cure prepregs at an elevated cure cycle; the most common temperatures used are 120°C and 200°C. Autoclaves, curing ovens, and heat bonders can be used to cure the prepreg material.

Consolidation is necessary if parts are made from several layers of prepreg, because large quantities of air can be trapped between each prepreg layer. Remove this trapped air by covering the prepreg with a perforated release film and a breather ply, and apply a vacuum bag. Apply the vacuum for 10 to 15 minutes at room temperature. Typically, attach the first consolidated ply to the tool face and repeat this process after every 3 or 5 layers depending on the prepreg thickness and component shape.

Store prepreg, film adhesive, and foaming adhesives in a freezer at a temperature below -18°C. If these types of materials need to be shipped, place them in special containers filled with dry ice. The freezer must not be of the automatic defrost type; the auto defrost cycle periodically warms the inside of the freezer, which can reduce the shelf life and consume the allowable out time of the composite material. Freezers must be capable of maintaining -18°C or below; most household freezers meet this level. Walk-in freezers can be used for large volume cold storage. If usage is small, a chest type freezer may suffice. Refrigerators are used to store laminating and paste adhesives and should be kept near 4°C. (***Figure 3-49***)

Uncured prepreg materials have time limits for storage and use. (***Figure 3-50***) The maximum time allowed for storing of a prepreg at low temperature is called the storage life, which is typically 6 months to a year. The material can be tested, and the storage life could be extended by the material manufacturer. The maximum



Figure 3-49. Walk-in freezer for storing prepeg materials.

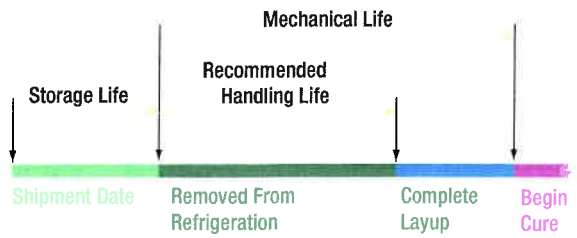


Figure 3-50. Storage life for prepeg materials.

time allowed for material at room temperature before the material cures is called the mechanical life. The recommended time at room temperature to complete layup and compaction is called the handling life. The handling life is shorter than the mechanical life. The mechanical life is measured from the time the material is removed from the freezer until the time the material is returned to the freezer. The operator must keep records of the time in and out of the freezer. Material that exceeds the mechanical life needs to be discarded.

Many repair facilities cut the material in smaller kits and store them in moisture proof bags that thaw quicker when removed from the freezer. This also limits the time out of the freezer for a big roll.

All frozen prepreg materials need to be stored in moisture proof back to avoid moisture contamination. All prepreg material should be protected from dust, oil, vapors, smoke, and other contaminants. A clean room for repair layup would be best, but if a clean room is not available, the prepreg should be protected by storing them in bags or keeping them covered with plastic. Before starting the layup, cover the unprotected sides of the prepreg with parting film, and clean the area being repaired immediately before laying up the repair plies.

Prepreg material is temperature sensitive. Excessively high temperatures cause the material to begin curing, and excessively low temperatures make the material difficult to handle. For repairs on aircraft in very cold or very hot climates, the area should be protected by a tent around the repair area. Prepare the prepreg repair plies in a controlled temperature environment and bring them to the repair area immediately before using them.

Co-Curing

Co-curing is a process wherein two parts are simultaneously cured. The interface between the two parts may or may not have an adhesive layer. Co-curing often results in poor panel surface quality, which is prevented by using a secondary surfacing material co-cured in the standard cure cycle or a subsequent fill-and-fair operation. Co-cured skins may also have poorer mechanical properties, requiring the use of reduced design values.

A typical co-cure application is the simultaneous cure of a stiffener and a skin. Adhesive film is frequently placed into the interface between the stiffener and the skin to increase fatigue and peel resistance. Principal advantages derived from the co-cure process are excellent fit between bonded components and guaranteed surface cleanliness.

Secondary Bonding

Secondary bonding utilizes precured composite detail parts, and uses a layer of adhesive to bond two precured composite parts. Honeycomb sandwich assemblies commonly use a secondary bonding process to ensure optimal structural performance. Laminates co-cured over honeycomb core may have distorted plies that have dipped into the core cells. As a result, compressive stiffness and strength can be reduced as much as 10 and 20 percent, respectively.

Precured laminates undergoing secondary bonding usually have a thin nylon or fiberglass peel ply cured onto the bonding surfaces. While the peel ply sometimes hampers nondestructive inspection of the precured laminate, it has been found to be the most effective means of ensuring surface cleanliness prior to bonding. When the peel ply is stripped away, a pristine surface becomes available. Light scuff sanding removes high resin peak impressions produced by the peel ply weave which, if they fracture, create cracks in the bondline.

Composite materials can be used to structurally repair, restore, or enhance aluminum, steel, and titanium components. Bonded composite doublers have the ability to slow or stop fatigue crack growth, replace lost structural area due to corrosion grind-outs, and structurally enhance areas with small and negative margins. This technology has often been referred to as a combination of metal bonding and conventional on aircraft composite bonded repair. Boron prepreg tape with an epoxy resin is most often used for this application.

Co-Bonding

In the co-bonding process, one of the detail parts is precured with the mating part being cured simultaneously with the adhesive. Film adhesive is often used to improve peel strength.

WET LAMINATED LAYUP TECHNIQUES

Read the stuctural repair manual and determine the correct repair material, number of plies required for the repair, and the ply orientation. Dry the part, remove the damage, and taper sand the edges of damaged area. Use a piece of thin plastic, and trace the size of each repair ply from the damaged area. Indicate the ply orientation of each ply on the trace sheet. Copy the repair ply information to a piece of repair material that is large enough to cut all plies. Impregnate the repair material with resin, place a piece of transparent release film over the fabric, cut out the plies, and layup the plies in the damaged area. The plies are usually placed using the smallest ply first taper layup sequence, but an alternative method is to use the largest ply first layup sequence. In this sequence, the first layer of reinforcing fabric completely covers the work area, followed by successively smaller layers, and then is finished with an extra outer layer or two extending over the patch and onto the sound laminate for some distance. Both methods are illustrated in ***Figures 3-51 and 3-52***.

Bleedout Technique

The traditional bleedout using a vacuum bag technique places a perforated release film and a breather/bleeder ply on top of the repair. The holes in the release film allow air to breath and resin to bleed off over the entire repair area. The amount of resin bled off depends on the size and number of holes in the perforated release film, the thickness of the bleeder/breather cloth, the resin viscosity and temperature, and the vacuum pressure.

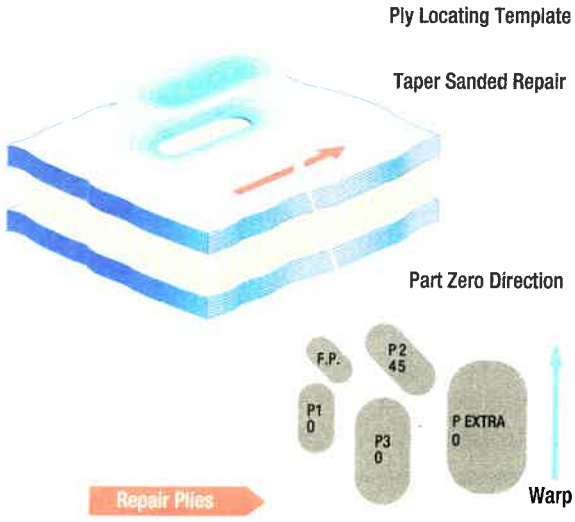


Figure 3-51. Repair layup process.

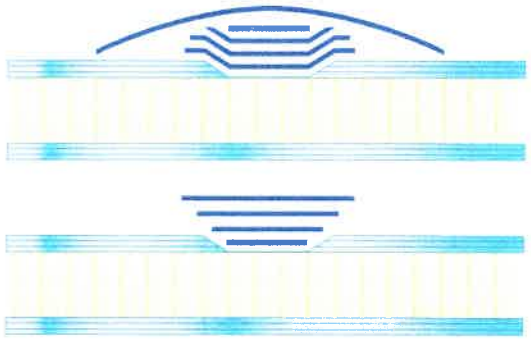


Figure 3-52. Different lay-up techniques.

Controlled bleed allows a limited amount of resin to bleed out in a bleeder ply. Place a piece of perforated release film on top of the prepreg material, a bleeder ply on top of the perforated release film, and a solid release film on top of the bleeder. Use a breather and a vacuum bag to compact the repair. The breather allows the air to escape. The bleeder can only absorb a limited amount of resin, and the amount of resin that is bled can be controlled by using multiple bleeder plies. Too many bleeder plies can result in a resin starved repair. Always consult the maintenance manual or manufacturer tech sheets for correct bagging and bleeding techniques.

No Bleedout

Prepreg systems with 32-35 percent resin content are typically no-bleed systems. These prepregs contain exactly the amount of resin needed in the cured laminate; therefore, resin bleedoff is not desired. Bleedout of these prepregs results in a resin starved repair or part. Many high strength prepregs in use today are no-bleed systems. No bleeder is used, and the resin is trapped/sealed so that none bleeds away. Consult the

maintenance manual to determine if bleeder plies are required for the repair. A sheet of solid release film (no holes) is placed on top of the prepreg and taped off at the edges with flash tape. Small openings are created at the edges of the tape so that air can escape. A breather and vacuum bag are installed to compact the prepreg plies. The air can escape on the edge of the repair but no resin can bleed out. (***Figure 3-53***)

Horizontal (or edge) bleedout is used for small room temperature wet layup repairs. A 2 inch strip of breather cloth is placed around the repair or part (edge breather). There is no need for a release film because there is no bleeder/breather cloth on top of the repair. The part is impregnated with resin, and the vacuum bag is placed over the repair. A vacuum is applied and a squeegee is used to remove air and excess resin to the edge breather.

Ply Orientation Warp Clock

In order to minimize any residual thermal stresses caused during cure of the resin, it is always good practice to design a symmetrical, or balanced, laminate. Examples of balance laminates are presented in ***Figure 3-54***. The first example uses unidirectional tape, and examples 2 and 3 are typical quasi-isotropic laminates fabricated from woven cloth.

Figure 3-55 presents examples of the effects caused by non-symmetrical laminates. These effects are most pronounced in laminates that are cured at high temperature in an autoclave or oven due to the thermal stresses developed in the laminate as the laminate cools down from the cure temperature to room temperature. Laminates cured at room temperature using typical wet layup do not exhibit the same degree of distortion due to the much smaller thermal stresses.



Figure 3-53. Vacuum bagging of contoured part.

Example	Lamina	Written As
1	±45°, –45°, 0°, 0°, –45°, +45°	(+45, –45, 0) S
2	±45°, 0°/90°, ±45°, 0°/90°, 0°/90°, ±45°, 0°/90°, ±45°	(±45, 0/90)2S
3	±45°, ±45°, 0°/90°, 0°/90°, ±45°, ±45°	([±45] 2, 0/90) S

Figure 3-54. Examples of balance laminates.

Type	Example	Comments
Symmetrical, balanced	(+45, –45, 0, 0, –45, +45)	Flat, constant midplane stress
Nonsymmetrical, balanced	(90, +45, 0, 90, –45, 0)	Induces curvature
Symmetrical, nonbalanced	(–45, 0, 0, –45)	Induces twist
Nonsymmetrical, nonbalanced	(90, –45, 0, 90, –45, 0)	Induces twist and curvature

Figure 3-55. Examples of the effects caused by non-symmetrical laminates.

The strength and stiffness of a composite buildup depends on the ply orientation. The practical range of strength and stiffness of carbon epoxy extends from values as low as those provided by fiberglass to as high as those provided by titanium. This range of values is determined by the orientation of the plies to the applied load. Because the strength design requirement is a function of the applied load direction, ply orientation and ply sequence must be correct. It is critical during a repair operation to replace each damaged ply with a ply of the same material and orientation or an approved substitute.

Warp is the longitudinal fibers of a fabric. The warp is the high strength direction due to the straightness of the fibers. A warp clock is used to describe direction of fibers on a diagram, spec sheet, or manufacturer's sheets. If the warp clock is not available on the fabric, the orientation is defaulted to zero as the fabric comes off the roll. Therefore, 90° to zero is across the width of the fabric. 90° to zero is also called the fill direction.

Mixing Resins

Epoxy resins, like all multi-part materials, must be thoroughly mixed. Some resin systems have a dye added to aid in seeing how well the material is mixed. Since many resin systems do not have a dye, the resin must

be mixed slowly and fully for three minutes. Air enters into the mixture if the resin is mixed too fast. If the resin system is not fully mixed, the resin may not cure properly. Make sure to scrape the edges and bottom of the mixing cup to ensure that all resin is mixed correctly Do not mix large quantities of quick curing resin. These types of resins produce heat after they are mixed. Smoke can burn or poison you when the resin overheats. Mix only the amount of material that is required. Mix more than one batch if more material is needed than the maximum batch size.

Saturation Techniques

For wet layup repair, impregnate the fabric with resin. It is important to put the right amount of resin on the fabric. Too much or too little resin affects the strength of the repair. Air that is put into the resin or not removed from the fabric also reduces the repair strength.

Impregnation With a Brush or Squeegee

The traditional way of impregnating the fabric is by using a brush or squeegee. The technician puts a mold release compound or a release film on a caul plate so that the plies will not adhere to the caul plate. Place a sheet of fabric on the caul plate and apply resin in the middle of the sheet. Use a brush or squeegee to thoroughly wet the fabric. More plies of fabric and resin are added and the process is repeated until all plies are impregnated. A vacuum bag will be used to consolidate the plies and to bleed off excess resin and volatiles. Most wet layup processes have a room temperature cure but extra heat, up to 65°C, are used to speed up the curing process. (***Figure 3-56***)

VACUUM BAGGING TECHNIQUES

Vacuum bag molding is a process in which the layup is cured under pressure generated by drawing a vacuum in the space between the layup and a flexible sheet placed over it and sealed at the edges. In the vacuum bag molding process, the plies are generally placed in the mold by hand layup using prepreg or wet layup. High flow resins are preferred for vacuum bag molding.

Single Side Vacuum Bagging

This is the preferred method if the repair part is large enough for a vacuum bag on one side of the repair.



Figure 3-56. Examples of the effects caused by nonsymmetrical laminates.

The vacuum bag is taped in place with tacky tape and a vacuum port is placed through the bag to create the vacuum.

Envelope Bagging

Envelope bagging is a process in which the part to be repaired is completely enclosed in a vacuum bag or the bag is wrapped around the end of the component to obtain an adequate seal. It is frequently used for removable aircraft parts, such as flight controls, access panels, etc., and when a part's geometry and/or the repair location makes it very difficult to properly vacuum bag and seal the area in a vacuum. In some cases, a part may be too small to allow installation of a single-side bag vacuum. Other times, the repair is located on the end of a large component that must have a vacuum bag wrapped around the ends and sealed all the way around. (***Figure 3-57***)



Figure 3-57. Envelope bagging of repair.

Impregnation Using a Vacuum Bag

The vacuum assisted impregnation method is used to impregnate repair fabric with a two part resin while enclosed inside a vacuum bag. This method is preferred for tight-knit weaves and when near optimum resin to fiber ratio is required. Compared to squeegee impregnation, this process reduces the level of entrapped air within the fabric and offers a more controlled and contained configuration for completing the impregnation process. Vacuum assisted impregnation consists of the following steps:

1. Place vacuum bag sealing tape on the table surface around the area that is used to impregnate the material.
2. The area should be at least 4 inches larger than the material to be impregnated.
3. Place an edge breather cloth next to the vacuum bag sealing tape. The edge breather should be 1 to 2 inches wide.
4. Place a piece of solid parting film on the table. The sheet should be 2 inches larger than the material to be impregnated.
5. Weigh the fabric to find the amount of resin mix that is necessary to impregnate the material.
6. Lay the fabric on the parting film.
7. Put a piece of breather material between the fabric and the edge breather to provide an air path.
8. Pour the resin onto the fabric. The resin should be a continuous pool in the center area of the fabric.
9. Put vacuum probes on the edge breather.
10. Place a second piece of solid parting film over the fabric. This film should be the same size or larger than the first piece.
11. Place and seal the vacuum bag, and apply vacuum to the bag.
12. Allow 2 minutes for the air to be removed from the fabric.
13. Sweep the resin into the fabric with a squeegee. Slowly sweep the resin from the center to the edge of the fabric. The resin should be uniformly distributed over all of the fabric.
14. Remove the fabric and cut the repair plies.

ALTERNATE PRESSURE APPLICATION

Shrink Tape

Another method of pressure application for oven cures is the use of shrink wrapping or shrink tape. This method is commonly used with parts that have been filament wound, because some of the same rules for application apply. The tape is wrapped around the completed layup, usually with only a layer of release material between the tape and the layup. Heat is applied to the tape, usually using a heat gun to make the tape shrink, a process that can apply a tremendous amount of pressure to the layup. After shrinking, the part is placed in the oven for cure. High quality parts can be made inexpensively using shrink tape.

C-Clamps

Parts can also be pressed together with clamps. This technique is used for solid laminate edges of honeycomb panels. Clamps (e.g., C-clamps and spring clamps) are

used for pressing together the edges of components and/or repair details. Always use clamps with pressure distribution pads because damage to the part may occur if the clamping force is too high. Spring clamps can be used in applications where resin squeeze-out during cure would require C-clamps to be retightened periodically.

Shotbags and Weights

Shotbags and weights can be used also to provide pressure, but their use is limited due to the low level of pressure imposed.

CURING OF COMPOSITE MATERIALS

A cure cycle is the time/temperature/pressure cycle used to cure a thermosetting resin system or prepreg. The curing of a repair is as important as the curing of the original part material. Unlike metal repairs in which the materials are premanufactured, composite repairs require the technician to manufacture the material. This includes all storage, processing, and quality control functions. An aircraft repair's cure cycle starts with material storage. Materials that are stored incorrectly can begin to cure before they are used for a repair. All time and temperature requirements must be met and documented. Consult the aircraft structural repair manual to determine the correct cure cycle for the part that needs to be repaired.

Room Temperature Curing

Room temperature curing is the most advantageous in terms of energy savings and portability. Room temperature cure wet layup repairs do not restore either the strength or the durability of the original 120°C or 200°C cure components and are often used for wet layup fiberglass repairs for noncritical components. Room temperature cure repairs can be accelerated by the application of heat. Maximum properties are achieved at 65°C. A vacuum bag can be used to consolidate the plies and to provide a path for air and volatiles to escape.

Elevated Temperature Curing

All prepreg materials are cured with an elevated temperature cure cycle. Some wet layup repairs use an elevated cure cycle as well to increase repair strength and to speed up the curing process. The curing oven and heat bonder uses a vacuum bag to consolidate the plies and to provide a path for air and volatiles to escape. The autoclave uses vacuum and positive pressure to consolidate the plies and to provide a path for air and volatiles to escape. Most heating devices use a programmable computer control to run the cure cycles. The operator can select from a menu of available cure cycles or write his or her own program. Thermocouples are placed near the repair, and they provide temperature feedback for the heating device.

Typical curing temperature for composite materials is 120°C or 200°C. The temperature of large parts that are cured in an oven or autoclave might be different from that of an oven or autoclave during the cure cycle, because they act like a heat sink. The part temperature is most important for a correct cure, so thermocouples are placed on the part to monitor and control part temperature. The oven or autoclave air temperature probe that measures oven or autoclave temperature is not always a reliable device to determine part curing temperature. The oven temperature and the part temperature can be substantially different if the part or tool acts as a heat sink. The elevated cure cycle consists of at least three segments:

- Ramp up: The heating device ramps up at a set temperature typically between 2°C to 3°C per minute.
- Hold or soak: The heating device maintains the temperature for a predetermined period.
- Cool down: The heating device cools down at a set temperature. Cool down temperatures are typically below 3°C per minute. When the heating device is below 50°C, the part can be removed. When an autoclave is used for curing parts, make sure that the pressure in the autoclave is relieved before the door is opened. (***Figure 3-58***)

The curing process is accomplished by the application of heat and pressure to the laminate. The resin begins to soften and flow as the temperature is increased. At lower temperatures, very little reaction occurs. Any volatile contaminants, such as air and/or water, are drawn out of the laminate with vacuum during this time. The laminate is compacted by applying pressure, usually vacuum (atmospheric pressure); autoclaves apply additional pressure, typically 50–100 psi. As the temperature approaches the final cure temperature, the rate of reaction greatly increases, and the resin begins to gel and harden. The hold at the final cure lets the resin finish curing and attain the desired structural properties.

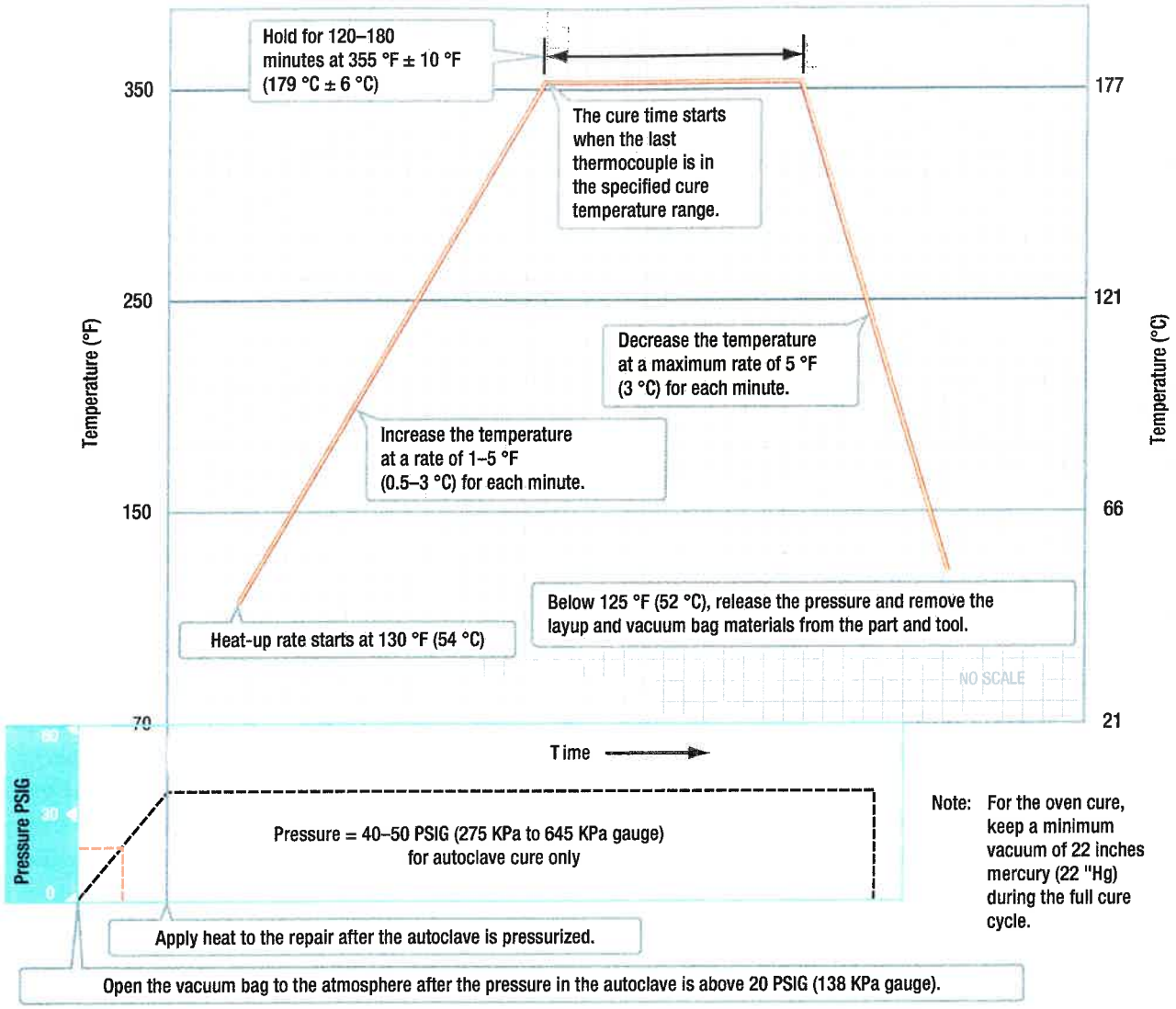


Figure 3-58. Autoclave cure.

COMPOSITE HONEYCOMB SANDWICH REPAIRS

A large proportion of current aerospace composite components are light sandwich structures that are susceptible to damage and are easily damaged. Because sandwich structure is a bonded construction and the face sheets are thin, damage to sandwich structure is usually repaired by bonding. Repairs to sandwich honeycomb structure use similar techniques for the most common types of face sheet materials, such as fiberglass, carbon, and Kevlar®. Kevlar® is often repaired with fiberglass. (***Figure 3-59***)

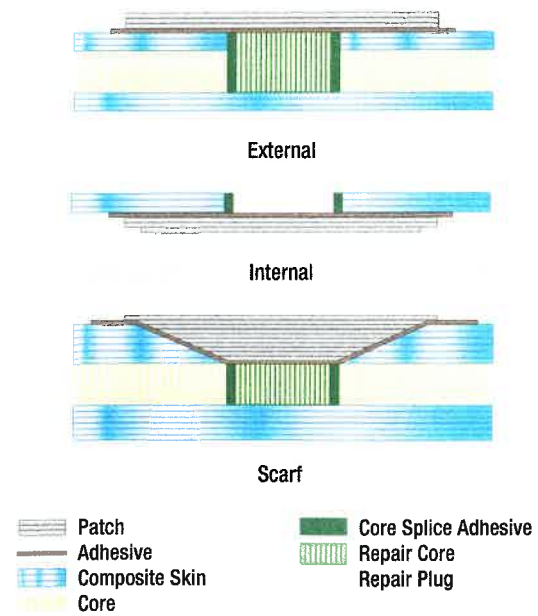


Figure 3-59. Typical repairs for honeycomb sandwich structure.

Damage Classification

A temporary repair meets the strength requirements, but is limited by time or flight cycles. At the end of the repair's life, the repair must be removed and replaced. An interim repair restores the required strength to

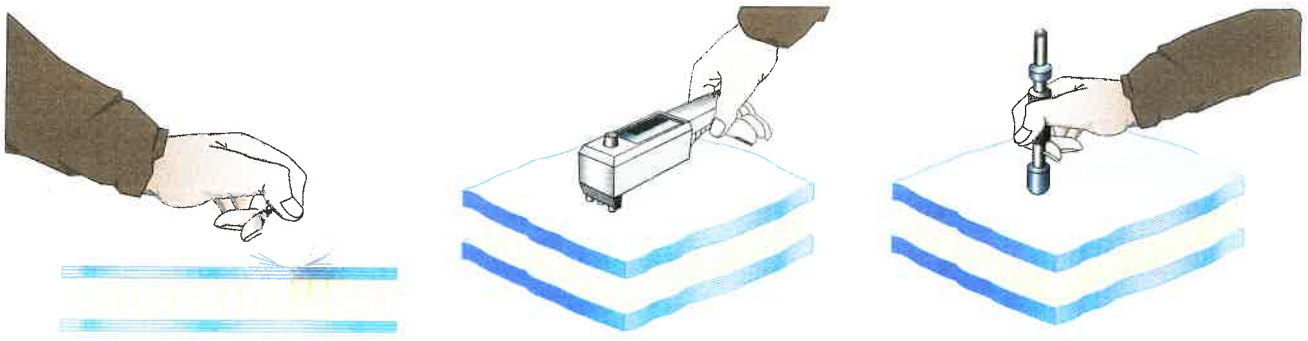


Figure 3-60. Tap testing techniques.

the component. However, this repair does not restore the required durability to the component. Therefore, it has a different inspection interval and/or method. A permanent repair is a repair that restores the required strength and durability to the component. The repair has the same inspection method and interval as the original component.

Minor Core Damage (Filler and Potting Repairs)

A potted repair can be used to repair damage to a sandwich honeycomb structure that is smaller than 0.5 inches. The honeycomb material could be left in place or could be removed and is filled up with a potting compound to restore some strength. Potted repairs do not restore the full strength of the part. Potting compounds are most often epoxy resins filled with hollow glass, phenolic or plastic microballoons, cotton, flux, or other materials. The potting compound can also be used as filler for cosmetic repairs to edges and skin panels. Potting compounds are also used in sandwich honeycomb panels as hard points for bolts and screws. The potting compound is heavier than the original core and this could affect flight control balance. The weight of the repair must be calculated and compared with flight control weight and balance limits set out in the Structural Repair Manual SRM.

Damage Requiring Core Replacement and Repair to One or Both Face Plates

Note: the following steps are not a substitution for the aircraft specific Structural Repair Manual (SRM). Do not assume that the repair methods used by one manufacturer are applicable to another manufacturer.

Step 1: Inspect the Damage

Thin laminates can be visually inspected and tap tested to map out the damage. (***Figure 3-60***) Thicker laminates need more in-depth NDI methods, such as ultrasonic inspection. Check in the vicinity of the damage for entry of water, oil, fuel, dirt, or other foreign matter. Water can be detected with X-ray, back light, or a moisture detector.

Step 2: Remove Water From Damaged Area

Water needs to be removed from the core before the part is repaired. (***Figure 3-61***) If the water is not removed, it boils during the elevated temperature cure cycle and the face sheets blow off the core, resulting in more damage. Water in the honeycomb core could also freeze at the low temperatures that exist at high altitudes, which could result in disbonding of the face sheets.

Step 3: Remove the Damage

Trim out the damage to the face sheet to a smooth shape with rounded corners, or a circular or oval shape. Do

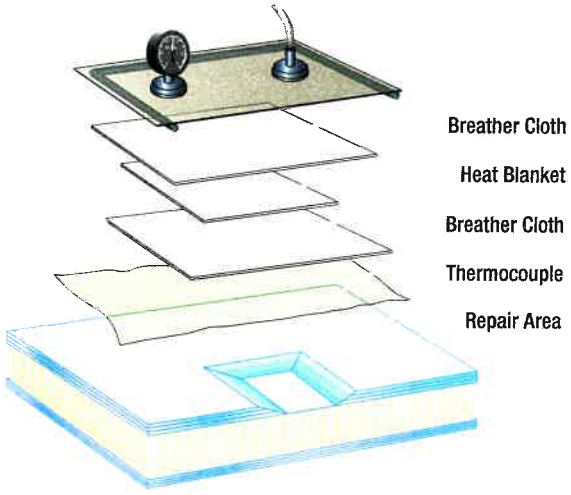


Figure 3-61. Vacuum bag method for drying parts.

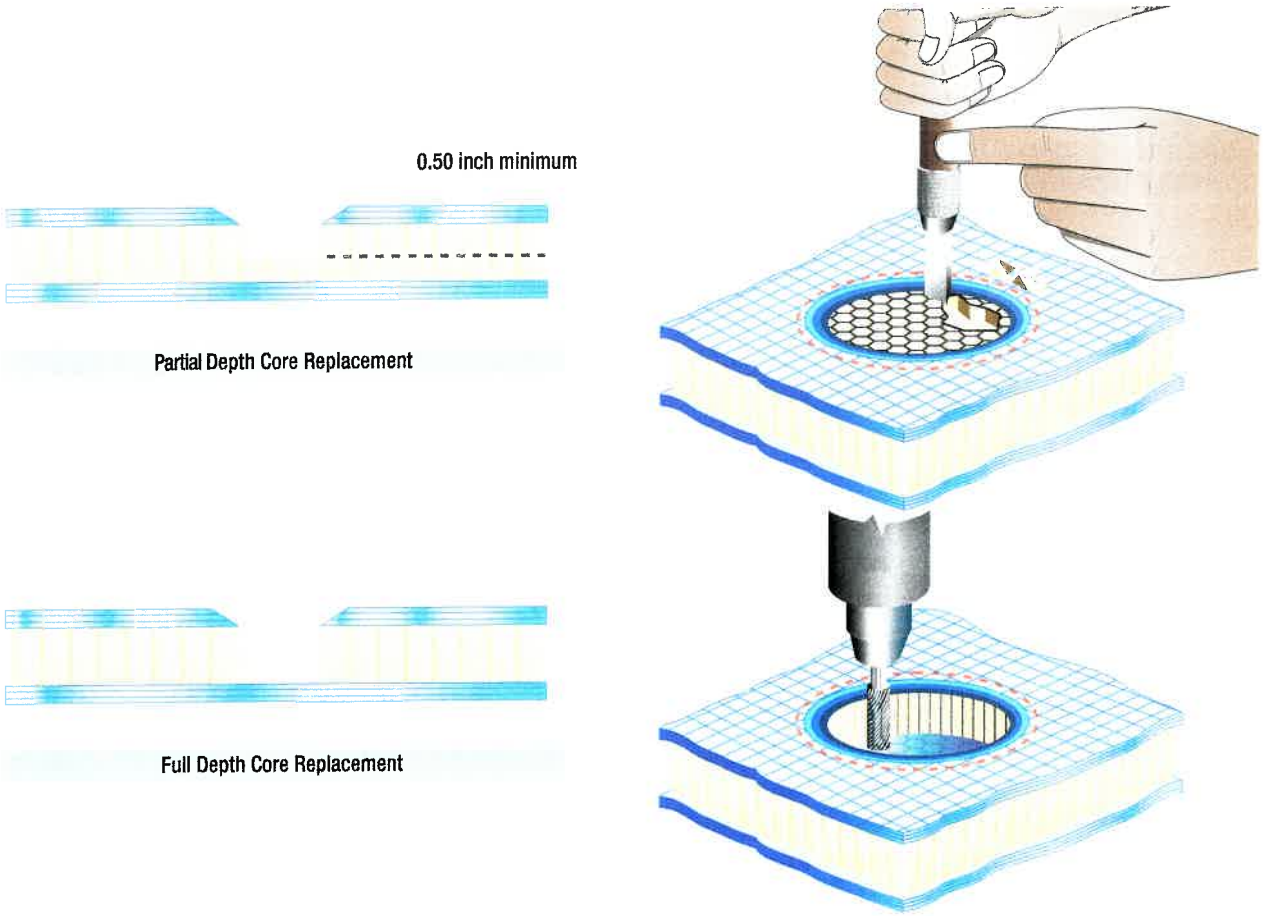


Figure 3-62. Core damage removal.

not damage the undamaged plies, core, or surrounding material. If the core is damaged as well, remove the core by trimming the same outline as the skin. (***Figure 3-62***)

Step 4: Prepare the Damaged Area

Use a flexible disk sander or a rotating pad sander to taper sand a uniform taper around the cleaned up damage. Some manufacturer's give a taper ratio, such as 1:40, and others prescribe a taper distance like a 1 inch overlap for each existing ply of the face sheet. Remove the exterior finish, including conductive coating for an area that is at least 1 inch larger than the border of the taper. Remove all sanding dust with dry compressed air and a vacuum cleaner. Use a clean cloth moistened with approved solvent to clean the damaged area. (***Figure 3-63***)



Figure 3-63. Taper sanding of repair area.

Step 5: Installation of Honeycomb Core (Wet Layup)

Use a knife to cut the replacement core. The core plug must be of the same type, class, and grade of the original core. The direction of the core cells should line up with the honey comb of the surrounding material. The plug must be trimmed to the right length and be solvent washed with an approved cleaner.

For a wet layup repair, cut two plies of woven fabric that fit on the inside surface of the undamaged skin. Impregnate the fabric plies with a resin and place in the hole. Use potting compound around the core and place it in the hole. For a prepreg repair, cut a piece of film adhesive that fits the hole and use a foaming adhesive around the plug. The plug should touch the sides of the hole. Line up the cells of the plug with the original

material. Vacuum bag the repair area and use an oven, autoclave, or heat blanket to cure the core replacement. The wet layup repair can be cured at a room temperature up to 65°C. The prepreg repair must be cured at 120°C or 200°C. Usually, the core replacement is cured with a separate curing cycle and not co-cured with the patch. The plug must be sanded flush with the surrounding area after the cure. (***Figure 3-64***)

Step 6: Prepare and Install the Repair Plies

Consult the repair manual for the correct repair material and the number of plies required for the repair. Typically, one more ply than the original number of plies is installed. Cut the plies to the correct size and ply orientation. The repair plies must be installed with the same orientation as that of the original plies being repaired. Impregnate the plies with resin for the wet layup repair, or remove the backing material from the prepreg material. The plies are usually placed using the smallest ply first taper layup sequence. (***Figure 3-65***)

Step 7: Vacuum Bag the Repair

Once the ply materials are in place, vacuum bagging is used to remove air and to pressurize the repair for curing. Refer to ***Figure 3-66*** for bagging instructions.

Step 8: Curing the Repair

The repair is cured at the required cure cycle. Wet layup repairs can be cured at room temperature. An elevated temperature up to 65°C can be used to speed up the cure. The prepreg repair needs to be cured at an elevated cure cycle. (***Figure 3-67***)

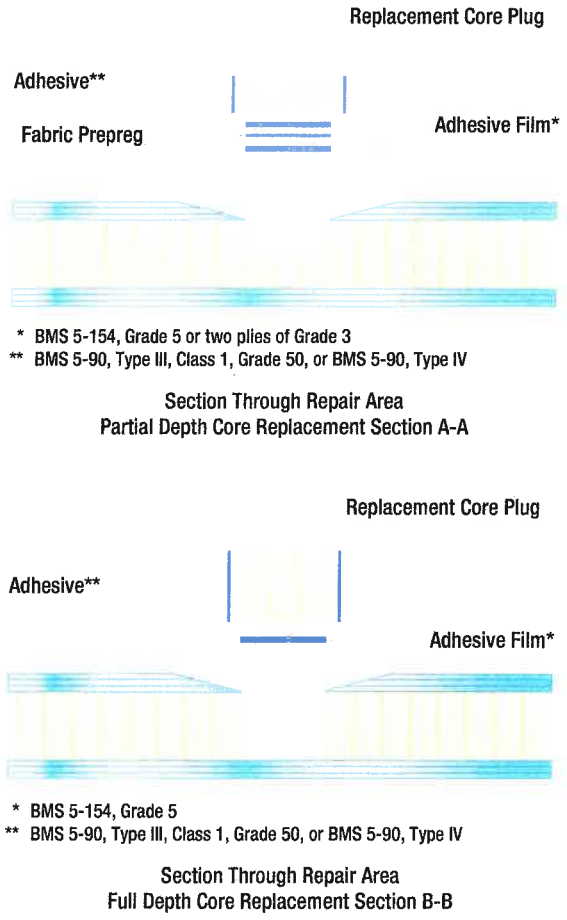


Figure 3-64. Core replacement.

Parts that can be removed from the aircraft could be cured in a hot room, oven, or autoclave. A heating blanket is used for on-aircraft repairs. Remove the bagging materials after curing and inspect the repair. The repair should be free from pits, blisters, resin rich and resin

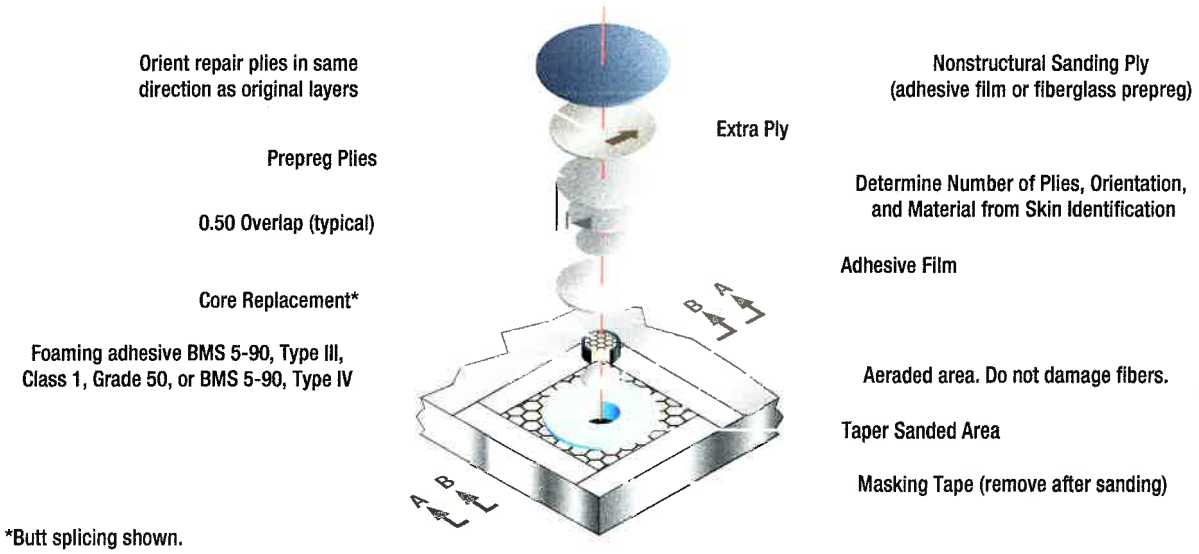


Figure 3-65. Repair ply installation.

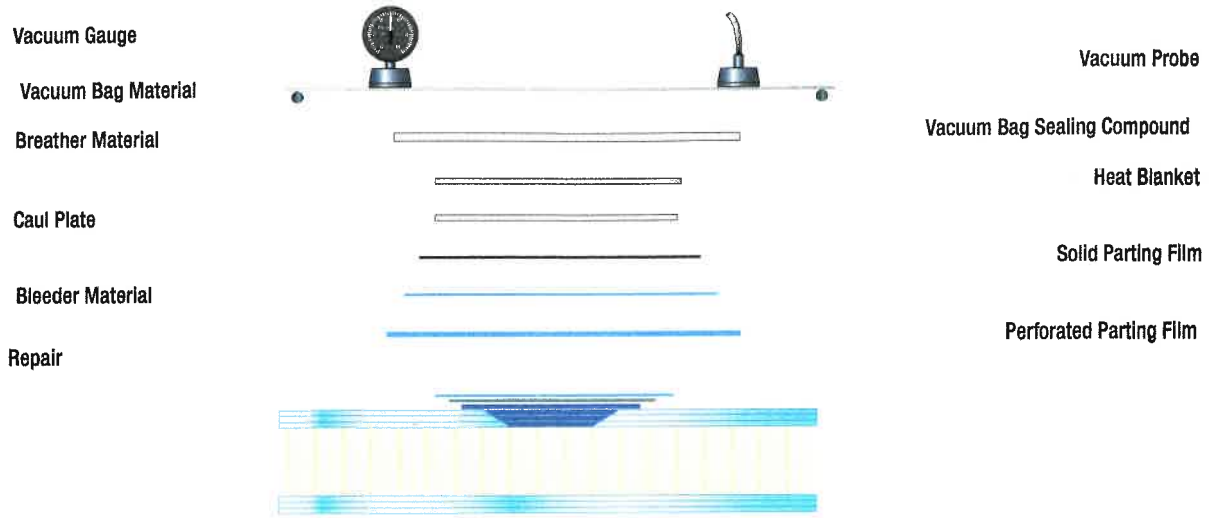
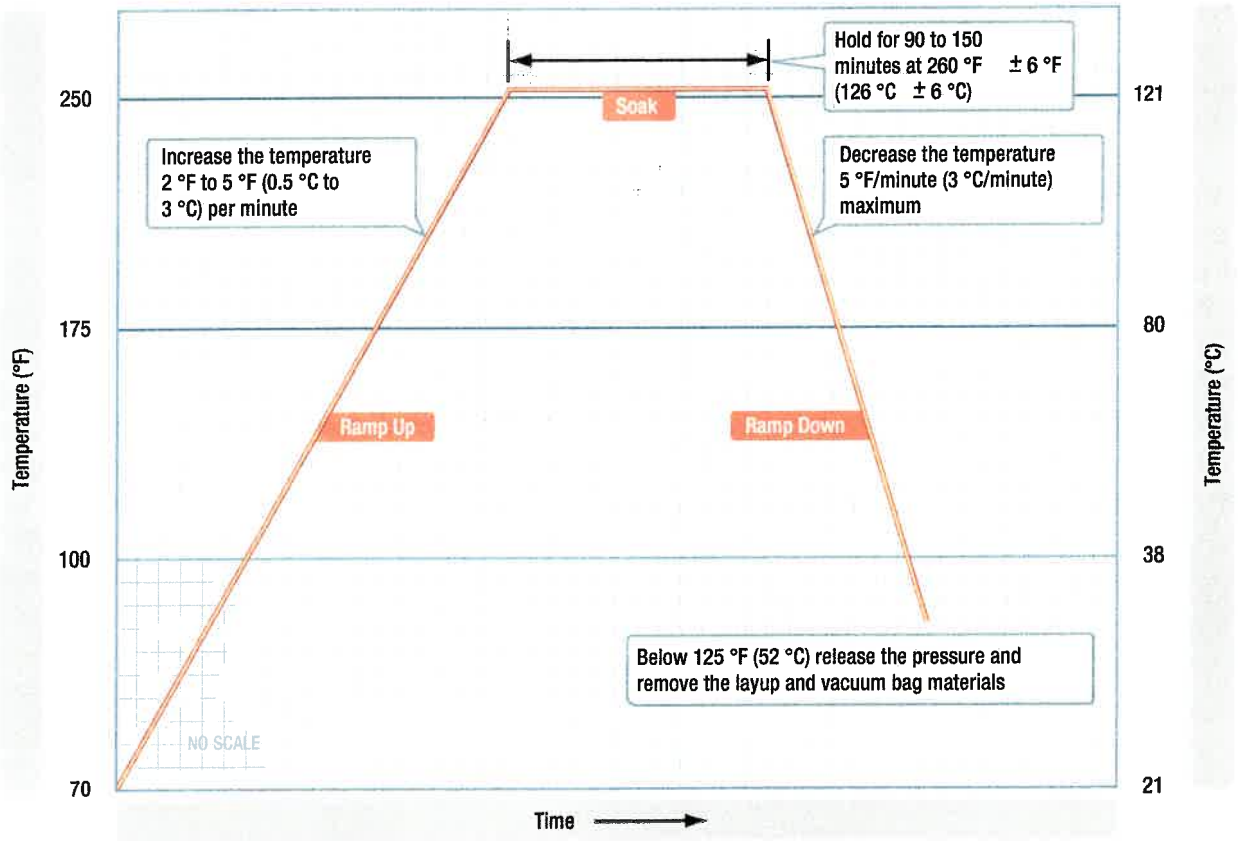


Figure 3-66. Vacuum processing.



Note: Keep a minimum vacuum of 22 inches of mercury during the cure cycle.

Figure 3-67. Curing the repair.

starved areas. Lightly sand the repair patch to produce a smooth finish without damaging the fibers. Apply top finish and conductive coating (lighting protection).

Step 9: Post Repair Inspection

Use visual, tap, and/or ultrasonic inspection to inspect the repair. Remove the repair patch if defects are found.

(***Figure 3-68***) Perform a balance check if a repair to a flight control surface was made, and ensure that the repaired flight control is within limits of the structural repair manual. Failure to do so could result in flight control flutter, and safety of flight could be affected.

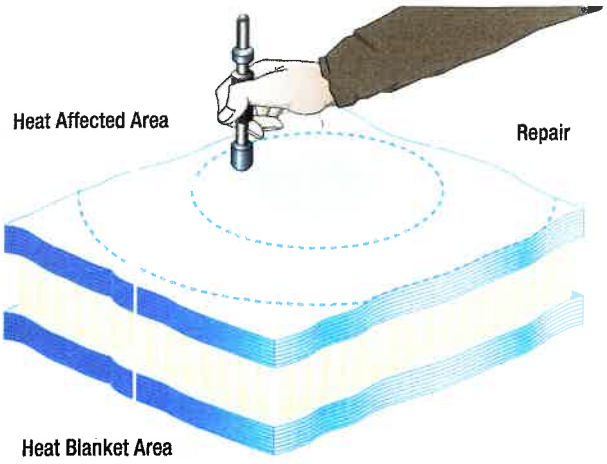


Figure 3-68. Post repair inspection.

BONDED FLUSH PATCH REPAIRS

New generation aircraft have fuselage and wing structures made from solid laminates that are externally stiffened with co-cured or co-bonded stringers. These solid laminates have many more plies than the face sheets of honeycomb sandwich structures. The flush repair techniques for solid laminate structures are similar for fiberglass, Kevlar®, and graphite with minor differences. A flush repair can be stepped or, more commonly, scarved (tapered). The scarf angles are usually small to ease the load into the joint and to prevent the adhesive from escaping. This translates into thickness-to-length ratios of 1:10 to 1:70. Because inspection of bonded repairs is difficult, bonded repairs, as contrasted with bolted repairs, require a higher commitment to quality control, better trained personnel, and cleanliness.

The scarf joint is more efficient from the viewpoint of load transfer as it reduces load eccentricity by closely aligning the neutral axis of the parent and the patch. However, this configuration has many drawbacks in making the repair. First, to maintain a small taper angle, a large quantity of sound material must be removed. Second, the replacement plies must be very accurately laid up and placed in the repair joint. Third, curing of replacement plies can result in significantly reduced strength if not cured in the autoclave. Fourth, the adhesive can run to the bottom of the joint, creating a nonuniform bond line. This can be alleviated by approximating the scarf with a series of small steps.

For these reasons, unless the part is lightly loaded, this type of repair is usually performed at a repair facility where the part can be inserted into the autoclave, which can result in part strength as strong as the original part.

There are several different repair methods for solid laminates. The patch can be precured and then secondarily bonded to the parent material. This procedure most closely approximates the bolted repair. (***Figure 3-69***)

The patch can be made from prepreg and then co-cured at the same time as the adhesive. The patch can also be made using a wet layup repair. The curing cycle can also vary in length of time, cure temperature, and cure pressure, increasing the number of possible repair combinations. Scarf repairs of composite laminates are performed in the sequence of steps described below.

Step 1: Inspection and Mapping of Damage

The size and depth of damage to be repaired must be accurately surveyed using appropriate Non-destructive Evaluation (NDE) techniques. A variety of NDE techniques can be used to inspect for damage in composite structures. The simplest technique is visual inspection, in which whitening due to delamination and/or resin cracking can be used to indicate the damage area in semitransparent composites, such as glass polyester and glass vinyl ester laminates. Visual inspection is not an accurate technique because not all damage is detectable to the eye, particularly damage hidden by paint, damage located deep below the surface, and damage in nontransparent composites, such as carbon and aramid laminates. A popular technique is tap testing, in which a lightweight object, such as a coin or hammer, is used to locate damage. The main benefits of tap testing are that it is simple and it can be used to rapidly inspect large areas. Tap testing can usually be used to detect delamination damage close to the surface, but becomes increasingly less reliable the deeper the delamination is located below the surface. Tap testing is not useful for detecting other types of damage, such as resin cracks and broken fibers.

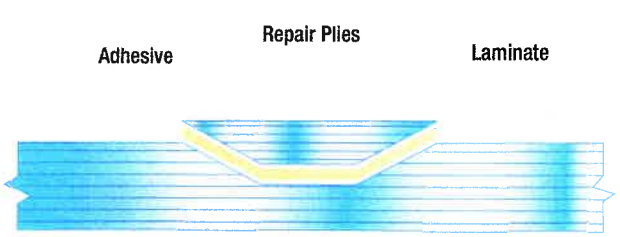


Figure 3-69. A precured patch can be secondarily bound to the partent material.

More advanced NDE techniques for inspecting composites are impedance testing, x-ray radiography, thermography, and ultrasonics. Of these techniques, ultrasonics is arguably the most accurate and practical and is often used for surveying damage. Ultrasonics can be used to detect small delaminations located deep below the surface, unlike visual inspection and tap testing.

Step 2: Removal of Damaged Material

Once the scope of the damaged area to be repaired has been determined, the damaged laminate must be removed. The edges of the sound laminate are then tapered back to a shallow angle. The taper slope ratio, also known as the scarf angle, should be less than 12 to 1 (<5°) to minimize the shear strains along the bond line after the repair patch is applied. The shallow angle also compensates for some errors in workmanship and other shop variables that might diminish patch adhesion. (***Figure 3-70***)

Step 3: Surface Preparation

The laminate close to the scarf zone should be lightly abraded with sandpaper, followed by the removal of dust and contaminates. It is recommended that, if the scarf zone has been exposed to the environment for any considerable period of time, it should be cleaned with a solvent to remove contamination.

Step 4: Molding

A rigid backing plate having the original profile of the composite structure is needed to ensure the repair has the same geometry as the surrounding structure.

Step 5: Laminating

Laminated repairs are usually done using the smallest ply first taper sequence. While this repair is acceptable, it produces relatively weak, resin rich areas at each ply edge at the repair interface. The largest ply first laminate sequence, where the first layer of reinforcing fabric completely covers the work area, produces a stronger interface joint. Follow the manufacturer's structural repair manual instructions.

Selection of the reinforcing material is critical to ensuring the repair has acceptable mechanical performance. The reinforcing fabric or tape should be identical to the reinforcement material used in the original composite. Also, the fiber orientation of the reinforcing layers within the repair laminate should match those of the original part laminate, so that the mechanical properties of the repair are as close to original as possible.

Step 6: Finishing

After the patch has cured, a conducting mesh and finish coat should be applied if needed.

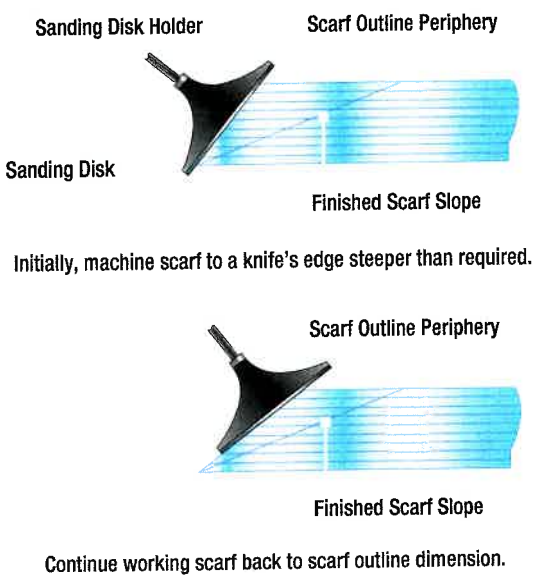


Figure 3-70. Scarf patch of solid laminates.

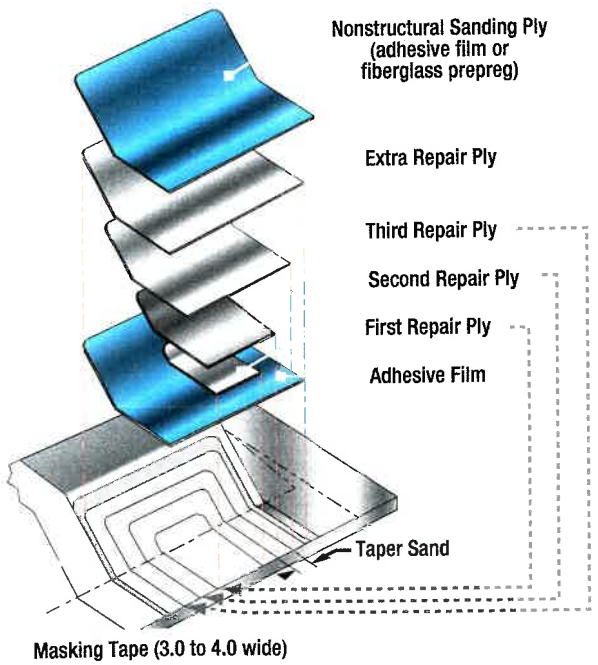


Figure 3-71. Trailing edge repair.

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TRAILING EDGE AND TRANSITION AREA PATCH REPAIRS

Trailing edges of control panels are highly vulnerable to damage. The aft 4 inches are especially subject to ground collision and handling, as well as to lightning strike. Repairs in this region can be difficult because both the skins and the trailing edge reinforcement may be involved. The repairs to a honeycomb core on a damaged edge or panel are similar to the repair of a sandwich honeycomb structure discussed in the Damage Requiring Core Replacement and Repair to One or Both Faceplate Repair sections. Investigate the damage, remove damaged plies and core, dry the part, install new core, layup the repair plies, curing and post inspection. A typical trailing edge repair is shown in ***Figure 3-71***.

RESIN INJECTION REPAIRS

Resin injection repairs are used on lightly loaded structures for small damages to a solid laminate due to delamination. Two holes are drilled on the outside of the delamination area and a low-viscosity resin is injected in one hole until it flows out the other hole. Resin injection repairs are sometimes used on sandwich honeycomb structure to repair a facesheet disbond. Disadvantages of the resin injection method are that the fibers are cut as a result of drilling holes, it is difficult to remove moisture from the damaged area, and it is difficult to achieve complete infusion of resin. (***Figure 3-72***)

COMPOSITE PATCH BONDED TO ALUMINUM STRUCTURE

Composite materials can be used to structurally repair, restore, or enhance metal components. Bonded composite doublers have the ability to slow or stop fatigue cracks, replace lost structural area due to corrosion grind outs, and structurally enhance areas with small and negative margins.

Boron and graphite epoxy materials have been used as composite patches to restore damaged metallic wing skins, fuselage sections, floor beams, and bulkheads. The stiff bonded composite materials constrain the cracked area, reduce gross stress in the metal, and provide an alternate load path around the crack. As a structural enhancement or blendout filler, the high modulus fibers offer negligible aerodynamic resistance and tailorable properties.

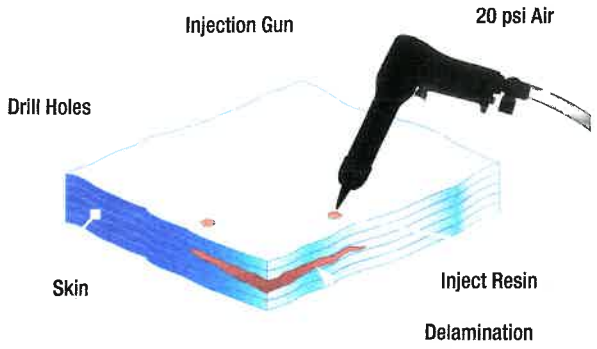


Figure 3-72. Resin injection repair.

Surface preparation is very important to achieve the adhesive strength. Grit blast silane and phosphoric acid anodizing are used to prepare aluminum skin. Film adhesives using a 121°C cure are used to bond the doublers to the metallic structure. Critical areas of the installation process include good thermal cure control, maintaining water free bond surfaces, and chemically and physically prepared bond surfaces.

Secondarily bonded precured doublers and insitu cured doublers have been used on a variety of structural geometries ranging from fuselage frames to door cutouts to blade stiffeners. Vacuum bags are used to apply the bonding and curing pressure between the doubler and metallic surface.

FIBERGLASS MOLDED MAT REPAIRS

Fiberglass molded mats consists of short fibers, with strength much less than composite products that use continuous fibers. Fiberglass molded mats are only used for nonstructural applications. The fiberglass molded mat is typically used in combination with fiberglass fabric. The molded mats are impregnated with resin just like a wet layup for fiberglass fabric. The advantage of the molded mat is the lower cost and the ease of use.

RADOME REPAIRS

Aircraft radomes, being an electronic window for the radar, are often made of nonconducting honeycomb sandwich structure with only three or four plies of fiberglass. The skins are so thin to not block the radar signals. The thin structure, combined with the location in front of the aircraft, makes the radome vulnerable to hail damage, bird strikes, and lightning strikes. Low impact damage could lead to disbonds and delamination. Often, water is found in the radome structure due to impact damage or erosion. Moisture

collects in the core material and begins a freeze/thaw cycle each time the airplane is flown. This eventually breaks down the honeycomb material causing a soft spot on the radome itself.

Damage to a radome needs to be repaired quickly to avoid further damage and radar signal obstructions. Trapped water or moisture can produce a shadow on the radar image and severely degrade the performance of the radar. To detect water ingression in radomes, the available NDE techniques include radiography, thermography, and a radome moisture meter that measures RF power loss caused by the presence of water. Repairs to radomes are similar to other honeycomb structures, but the technician needs to realize that repairs could affect the radar performance. A special mold is necessary to repair severely damaged radomes. (***Figure 3-73***)

Transmissivity testing after radome repair ensures that the radar signal is transmitted properly through the radome. Radomes have lightning protections strips bonded to the outside to dissipate the energy of a lighting strike. It is important that these protection strips are in good condition to avoid damage to the radome structure. Typical failures of lightning protection strips are high resistance caused by shorts in the strips or attaching hardware and disbonding of the strips from the radome surface. (***Figures 3-74***)

EXTERNAL BONDED PATCH REPAIRS

Repairs to damaged composite structures can be made with an external patch. The external patch repair could be made with prepreg, a wet layup, or a precured patch. External patches are usually stepped to reduce the stress concentration at the edge of the patch. The disadvantages of the external patch are the eccentricity of the loading that causes peel stresses and the protrusion of the patch in the air stream. The advantage of the external patch is that it is easier to accomplish than a flush scarf type repair.



Figure 3-73. Radome repair tool.

EXTERNAL BONDED REPAIR WITH PREPREG PLIES

The repair methods for carbon, fiberglass, and Kevlar® are similar. Fiberglass is sometimes used to repair Kevlar® material. The main steps in repairing damage with an external patch are investigating and mapping the damage, removal of the damage, layup of the repair plies, vacuum bagging, curing, and finish coating.

Step 1: Investigating and Mapping the Damage
Use the tap test or ultrasonic test to map out the damage.

Step 2: Damage Removal
Trim out the damage to a smooth round or oval shape. Use scotch or sand paper to rough up the parent surface at least 1 inch larger than the patch size. Clean the surface with an approved solvent and cheese cloth.

Step 3: Layup of the Repair Plies
Use the structural repair manual to determine the number, size, and orientation of the repair plies. The repair ply material and orientation must be the same as the orientation of the parent structure. The repair can be stepped to reduce peel stresses at the edges.

Step 4: Vacuum Bagging
A film adhesive is placed over the damaged area and the repair layup is placed on top of the repair. The vacuum bagging materials are placed on top of the repair (see Prepreg Layup and Controlled Bleed Out) and a vacuum is applied.



Figure 3-74. Lightning protection strips on a radome.

Step 5: Curing the Repair

The prepreg patch can be cured with a heater blanket that is placed inside the vacuum bag, oven, or autoclave when the part can be removed from the aircraft. Most prepregs and film adhesives cure at either 120°C or 176°C. Consult the structural repair manual for the correct cure cycle.

Step 6: Applying Top Coat

Remove the vacuum bag from the repair after the cure and inspect the repair, remove the patch if the repair is not satisfactory. Lightly sand the repair and apply a protective top coating.

EXTERNAL REPAIR USING WET LAYUP AND DOUBLE VACUUM DEBULK METHOD (DVD)

Generally, the properties of a wet layup repair are not as good as a repair with prepreg material; but by using a DVD method, the properties of the wet layup process can be improved. The DVD process is a technique to remove entrapped air that causes porosity in wet layup laminates. The DVD process is often used to make patches for solid laminate structures for complex contoured surfaces. The wet layup patch is prepared in a DVD tool and then secondary bonded to the aircraft structure. (***Figure 3-75***) The laminating process is similar to a standard wet layup process. The difference is how the patch is cured.

Double Vacuum Debulk Principle

The double vacuum bag process is used to fabricate wet layup or prepreg repair laminates. Place the impregnated fabric within the debulking assembly, shown in ***Figure 3-75***.

To begin the debulking process, evacuate the air within the inner flexible vacuum bag. Then, seal the rigid outer box onto the inner vacuum bag, and evacuate the volume of air between the rigid outer box and inner vacuum bag. Since the outer box is rigid, the second evacuation prevents atmospheric pressure from pressing down on the inner vacuum bag over the patch. This subsequently prevents air bubbles from being pinched off within the laminate and facilitates air removal by the inner vacuum. Next, heat the laminate to a predetermined debulking temperature in order to reduce the resin viscosity and further improve the removal of air and volatiles from the laminate. Apply the heat through a heat blanket that is controlled with thermocouples placed directly on the heat blanket.

Once the debulking cycle is complete, compact the laminate to consolidate the plies by venting the vacuum source attached to the outer rigid box, allowing atmospheric pressure to reenter the box and provide positive pressure against the inner vacuum bag. Upon completion of the compaction cycle, remove the laminate from the assembly and prepare for cure. DVD tools can be purchased commercially but can also be fabricated locally from wood two by fours and sheets of plywood, as illustrated in ***Figure 3-75***.

PATCH INSTALLATION ON THE AIRCRAFT

After the patch comes out of the DVD tool, it is still possible to form it to the contour of the aircraft, but the time is typically limited to 10 minutes. Place a film adhesive, or paste adhesive, on the aircraft skin and place the patch on the aircraft. Use a vacuum bag and heater blanket to cure the adhesive. (***Figures 3-76 and 3-77***)

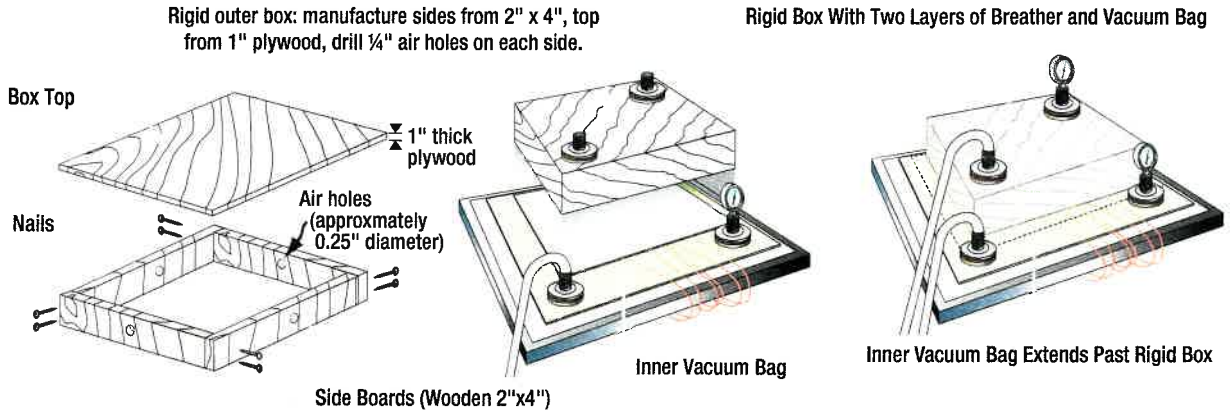


Figure 3-75. DVD tool made from wood 2x4's and plywood.

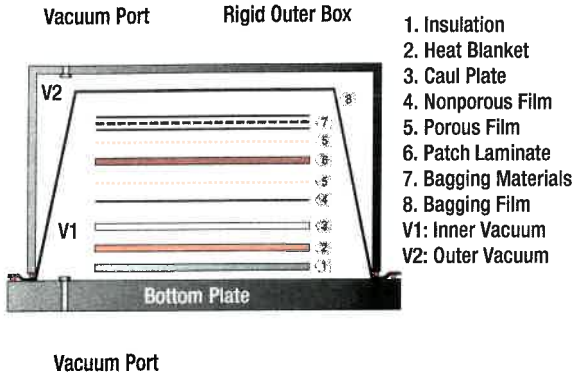


Figure 3-76. Double vacuum debulk schematic.

EXTERNAL REPAIR USING PRECURED LAMINATE PATCHES

Precured patches are not very flexible and cannot be used on highly curved or compound curved surfaces. The repair steps are similar as in External Bonded Repair With Prepreg Plies, except step 3 and 4 that follow.

Step 3: A Precured Patch

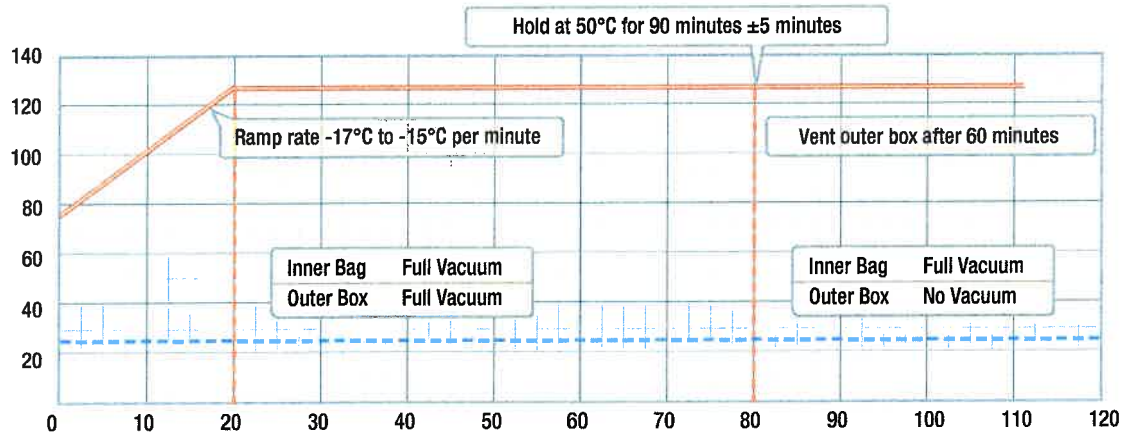
Consult the structural repair manual for correct size, ply thickness, and orientation. You can laminate and cure the precured patch in the repair shop and secondary bond to the parent structure, or obtain standard precured patches. (***Figure 3-78***)

Step 4: For a Precured Patch

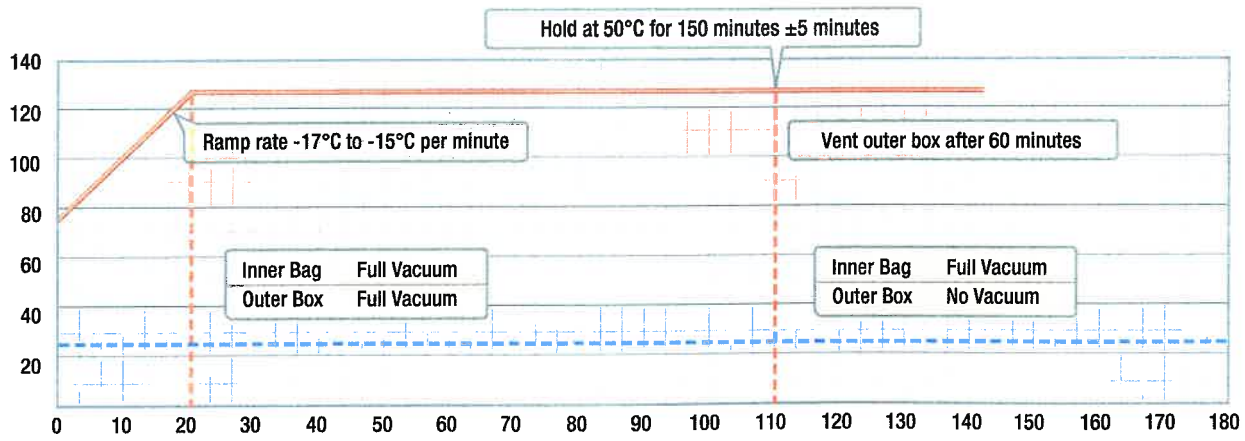
Apply film adhesive or paste adhesive to the damaged area and place the precured patch on top. Vacuum bag the repair and cure at the correct temperature for the film adhesive or paste adhesive. Most film adhesive cure at either 120°C or 200°C. Some paste adhesives cure at room temperature although an elevated temperature could be used to speed the curing process.

BONDED VERSUS BOLTED REPAIRS

Bonded repair concepts have found applicability in both types of manufacturing assembly methods. They have the advantage of not introducing stress concentrations



SMP-0029M1-9 Double Vacuum Debulk Cycle—Laminate Thickness <16 plies



SMP-0029M1-10 Double Vacuum Debulk Cycle—Laminate Thickness <16 plies

Figure 3-77. DVD cure cycle.

by drilling fastener holes for patch installation and can be stronger than original part material. The disadvantage of bonded repairs is that most repair materials require special storage, handling, and curing procedures.

Bolted repairs are quicker and easier to fabricate than bonded repairs. They are normally used on composite skins thicker than 0.125 inch to ensure sufficient fastener bearing area is available for load transfer. They are prohibited in honeycomb sandwich assemblies due to the potential for moisture intrusion from the fastener holes and the resulting core degradation. Bolted repairs are heavier than comparable bonded repairs, limiting their use on weight sensitive flight control surfaces.

Honeycomb sandwich parts often have thin face sheets and are most effectively repaired by using a bonded scarf type repair. A bonded external step patch can be used as an alternative. Bolted repairs are not effective for thin laminates because of the low bearing stress of the composite laminate. Thicker solid laminates used on larger aircraft can be up to an inch thick in highly loaded areas and these types of laminates cannot be effectively repaired using a bonded scarf type repair. (***Figure 3-79***)

Bonded versus Bolted Repair	Bolted	Bonded
Lightly loaded structures – laminate thickness less than 0.1"		X
Highly loaded structures – laminate thickness between 0.125" – 0.5"	X	X
Highly loaded structures – laminate thickness larger than 0.5"	X	
High peeling stresses	X	
Honeycomb structure		X
Dry surfaces	X	X
Wet and/or contaminated surfaces	X	
Disassembly required	X	
Restore unnotched strength		X

Figure 3-79. Bolted versus bonded repair.

Bolted Repairs

Aircraft designed in the 1970s used composite sandwich honeycomb structure for lightly loaded secondary structure, but new large aircraft use thick solid laminates for primary structure instead of sandwich honeycomb.

These thick solid laminate structures are quite different from the traditional sandwich honeycomb structures used for flight controls, landing gear doors, flaps, and spoilers of today's aircraft. They present a challenge to repair and are difficult to repair with a bonded repair method. Bolted repair methods have been developed to repair thicker solid laminates.



Figure 3-78. Precured patches.

Bolted repairs are not desirable for honeycomb sandwich structure due to the limited bearing strength of the thin face sheets and weakened honeycomb structure from drilling holes. The advantage of a bolted repair is that you need to select only patch material and fasteners, and the repair method is similar to a sheet metal repair. There is no need for curing the repair and storing the prepreg repair material and film adhesives in a freezer. Patches may be made from aluminum, titanium, steel, or precured composite material. Composite patches are often made from carbon fiber with an epoxy resin or fiberglass with an epoxy resin.

You can repair a carbon fiber structure with an aluminum patch, but you must place a layer of fiberglass cloth between the carbon part and the aluminum patch to prevent galvanic corrosion. Titanium and precured composite patches are preferred for repair of highly loaded components. Precured carbon/epoxy patches have the same strength and stiffness as the parent material as they are usually cured similarly.

Titanium or stainless steel fasteners are used for bolted repairs of a carbon fiber structure. Aluminum fasteners corrode if used with carbon fiber. Rivets cannot be used because the installation of rivets using a rivet gun introduces damage to the hole and surrounding structure and rivets expand during installation, which is undesirable for composite structures because it could cause delamination of the composite material.

Bolted Repair Procedures

Step 1: Inspection of the Damage

The tap test is not effective to detect delamination in thick laminates unless the damage is close to the surface. An ultrasonic inspection is necessary to determine the damage area. Consult the structural repair manual to find an applicable NDI procedure.

Step 2: Removal of the Damage

The damaged area needs to be trimmed to a round or rectangular hole with large smooth radii to prevent stress concentrations. Remove the damage with a sander, router, or similar tool.

Step 3: Patch Preparation

Determine the size of the patch based on repair information found in the structural repair manual. Cut, form, and shape the patch before attaching the patch to the damaged structure. It is easier to make the patch a little bigger than calculated and trim to size after drilling all fastener holes. In some cases, the repair patches are stocked preshaped and predrilled. If cutting is to be performed, standard shop procedures should be used that are suitable for the patch material.

Titanium is hard to work and requires a large powerful slip roller to curve the material. Metal patches require filing to prevent crack initiation around the cut edges. When drilling pilot holes in the composite, the holes for repair fasteners must be a minimum of four diameters from existing fasteners and have a minimum edge distance of three fastener diameters. This is different from the standard practice for aluminum of allowing a two diameter distance. Specific pilot hole sizes and drill types to be used should follow specific structural repair manual instructions. (***Figure 3-80***)

Step 4: Hole Pattern Lay Out

To locate the patch on the damaged area, draw two perpendicular centerlines on the parent structure and on the patch material that define the principal load or geometric directions. Then, lay out hole pattern on the patch and drill pilot holes in the patch material. Align the two perpendicular centerlines of the patch with the lines on the parent structure and transfer the pilot holes to the parent material. Use clecos to keep the patch in place. Mark the edges so that it can be returned to the same location easily.

Step 5: Drilling and Reaming Holes in Patch and Parent Structure

Composite skins should be backed up to prevent splitting. Enlarge the pilot holes in the patch and parent materials with a drill 1/64 under size and then ream all holes to the correct size. A tolerance of +0.002 5/–0.000 inch is usually recommended for aircraft parts. For composites, this means interference fasteners are not used.

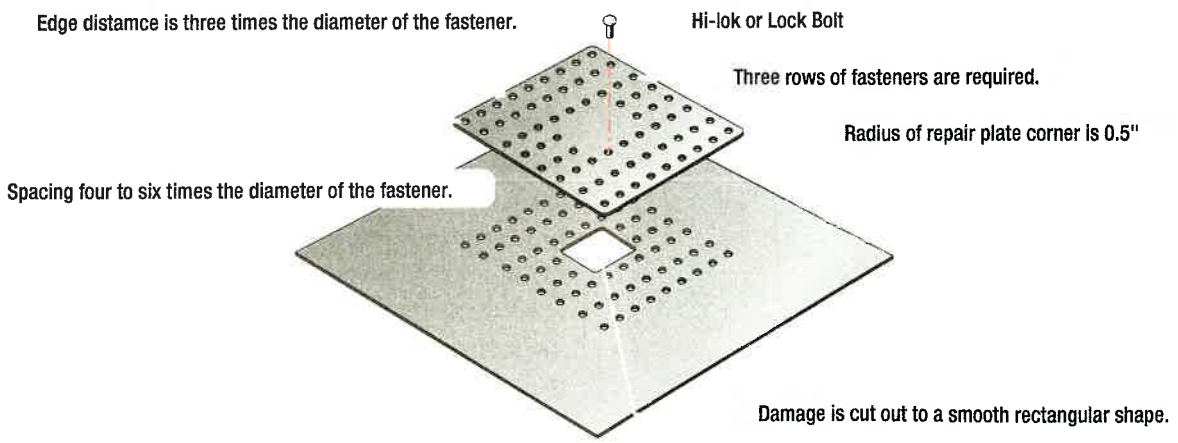


Figure 3-80. Repair layout for bolted repair of composite structure.

Step 6: Fastener Installation
Once fastener holes are drilled full size and reamed, permanent fasteners are installed. Before installation, measure the fastener grip length for each fastener using a grip length gauge. As different fasteners are required for different repairs, consult the SRM for permissible fastener type and installation procedure. However, install all fasteners wet with sealant and with proper torque for screws and bolts.

Step 7: Sealing of Fasteners and Patch
Sealants are applied to bolted repairs for prevention of water/moisture intrusion, chemical damage, galvanic corrosion, and fuel leaks. They also provide contour smoothness. The sealant must be applied to a clean surface. Masking tape is usually placed around the periphery of the patch, parallel with the patch edges and leaving a small gap between the edge of the patch and the masking tape. Sealing compound is applied into this gap.

Step 8: Finish Coat and Lightening Protection Mesh
The repair needs to be sanded, primed, and painted with an approved paint system. A lightning protection mesh needs to be applied if composite patches are used in an area that is prone to lightening strikes.

FASTENERS USED WITH COMPOSITE LAMINATES

Many companies make specialty fasteners for composite structures and several types of fasteners are commonly used: threaded fasteners, lock bolts, blind bolts, blind rivets, and specialty fasteners for soft structures, such as honeycomb panels. The main differences between fasteners for metal and composite structures are the materials and the footprint diameter of nuts and collars.

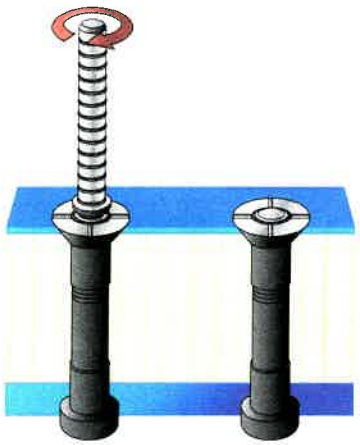


Figure 3-81. ASP fastener system.

Corrosion Precautions

Neither fiberglass nor Kevlar® fiber reinforced composites cause corrosion problems when used with most fastener materials. Composites reinforced with carbon fibers, however, are quite cathodic when used with materials, such as aluminum or cadmium, the latter of which is a common plating used on fasteners for corrosion protection.

Fastener Materials

Titanium alloy Ti-6Al-4V is the most common alloy for fasteners used with carbon fiber reinforced composite structures. Austenitic stainless steels, superalloys (e.g., A286), multiphase alloys (e.g., MP35N or MP159), and nickel alloys (e.g., alloy 718) also appear to be very compatible with carbon fiber composites.

FASTENER SYSTEMS

Adjustable Sustain Preload Fastening System (ASP)

The Adjustable Sustain Preload (ASP) fastening system provides a simplified method of fastening composite, soft core, metallic or other materials, which are sensitive to fastener clamp-up or installation force conditions. Clamping force can be infinitely adjustable within maximum recommended torque limits and no further load is applied during installation of the lock collar. The fastener is available in two types. The ASP® has full shank and the 2ASP® has a pilot type shank. (***Figures 3-81 and 3-82***)

Hi-Lok® and Huck-Spin® Lockbolt Fasteners

Most composite primary structures for the aircraft industry are fastened with Hi-Loks® (Hi-Shear Corp.) or Huck-Spin® lockbolts for permanent installations. The Hi-Lok® is a threaded fastener that incorporates a hex key in the threaded end to react to the torque applied to the collar during installation. The collar includes a frangible portion that separates at a predetermined torque value. (***Figure 3-83***)

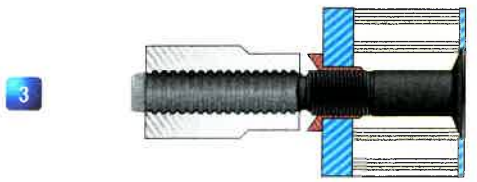
The lockbolt incorporates a collar that is swaged into annular grooves. It comes in two types: pull and stump. The pull type is the most common, where a frangible pintail is used to react the axial load during the swaging of the collar. When the swaging load reaches a predetermined limit, the pintail breaks away at the breakneck groove. The installation of the Hi-Lok® and the pull-type Huck-Spin® lockbolt can be performed by



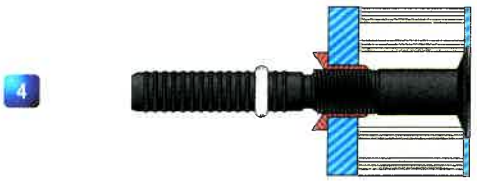
Pin Component Installed Clearance Fit



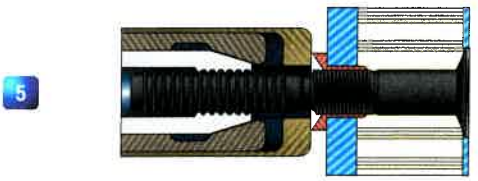
Sleeve Component Threaded On Pin



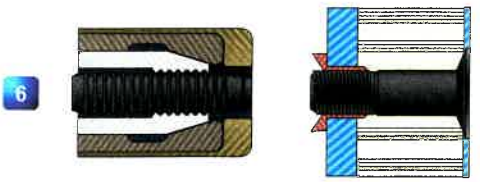
Torque Controlled Tool Tightens Sleeve



Lock Collar Placed On Pin



Lock Collar Swaged On Pin Splines



Pintail Breaks Off

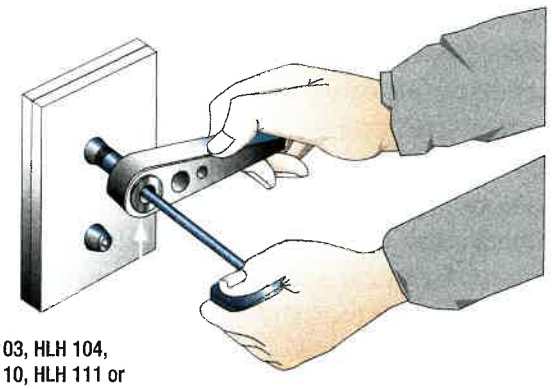
Figure 3-82. ASP fastener system installation sequence.

one technician from one side of the structure. The stump-type lockbolt, on the other hand, requires support on the head side of the fastener to react the swage operation. This method is usually reserved for automated assembly of detail structure in which access is not a problem. The specific differences in these fasteners for composite structure in contrast to metal structure are small.

For the Hi-Lok®, material compatibility is the only issue; aluminum collars are not recommended. Standard collars of A286, 303 stainless steel, and titanium alloy are normally used. The Huck-Spin® lockbolt requires a hat-shaped collar that incorporates a flange to spread the high bearing loads during installation. The lockbolt pin designed for use in composite structure has six annular grooves as opposed to five for metal structure. (***Figures 3-84 and 3-85***)

Eddie-Bolt® Fasteners

Eddie-Bolt® fasteners (Alcoa) are similar in design to Hi-Loks® and are a natural choice for carbon fiber composite structures. The Eddie-Bolt® pin is designed with flutes in the threaded portion, which allow a positive lock to be made during installation using a specially designed mating nut or collar. The mating nut has three lobes that serve as driving ribs. During installation, at a predetermined preload, the lobes compress the nut material into the flutes of the pin and form the locking feature. The advantage for composite structure is that titanium alloy nuts can be used for compatibility and weight saving without the fear of galling. The nuts spin on freely, and the locking feature is established at the end of the installation cycle. (***Figure 3-86***)



HLH 103, HLH 104,
HLH 110, HLH 111 or
HLH 500 installation tool

Figure 3-83. Hi-Lok® installation.

Large 130 Flush Shear Head

Lock Grooves



CP titanium Flanged Collar

Pull Grooves

Figure 3-84. Huck-Spin® lockbolt.

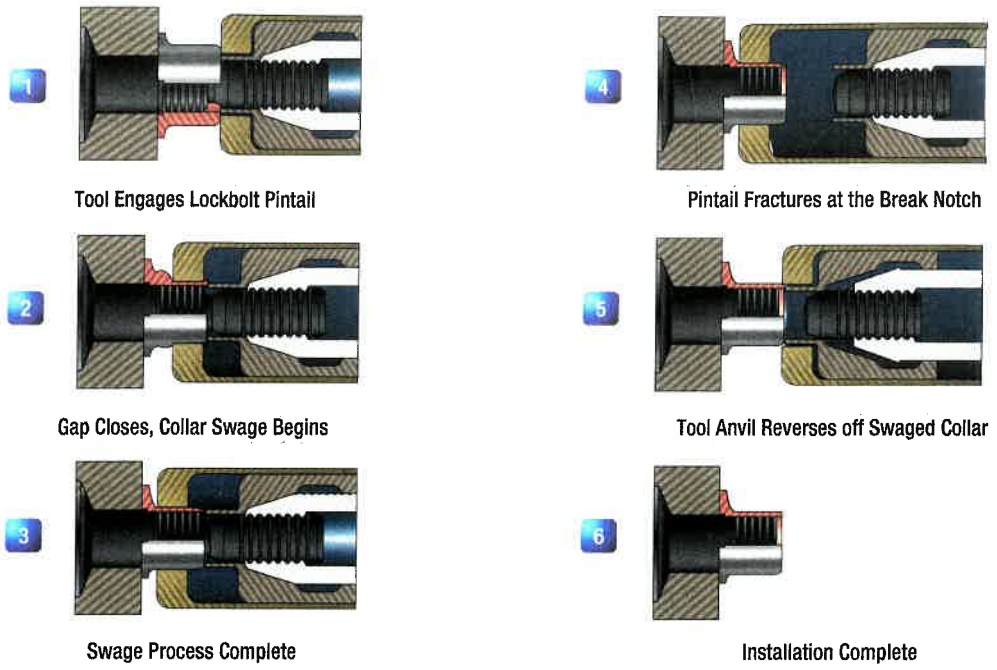


Figure 3-85. Huck-Spin® installation sequence.



Figure 3-86. Eddie Bolts®.

Cherry's E-Z Buck® (CSR90433) Hollow Rivet

The Cherry Hollow End E-Z Buck® rivet is made from titanium/columbium alloy and has a shear strength of 40 KSI. The E-Z Buck® rivet is designed to be used in a double flush application for fuel tanks. The main advantage of this type of rivet is that it takes less than half the force of a solid rivet of the same material. The rivets are installed with automated riveting equipment or a rivet squeezer. Special optional dies ensure that the squeezer is always centered during installation, avoiding damage to the structure. (***Figure 3-87***)

Blind Fasteners

Composite structures do not require as many fasteners as metal aircraft because stiffeners and doublers are co-cured with the skins, eliminating many fasteners. The size of panels on aircraft has increased in composite structures, which causes backside inaccessibility. Therefore, blind fasteners or screws and nutplates must be used in these areas. Many manufacturer's make blind fasteners for composite structures; a few are discussed below.

Blind Bolts

The Cherry Maxibolt® is available in titanium for compatibility with composite structures. The shear strength of the Maxibolt® is 95 KSI. It can be installed from one side with a G-83 or equivalent pneumatic-hydraulic installation tool, and is available in 100° flush head, 130° flush head and protruding head styles. (***Figure 3-88***)

The Alcoa UAB™ blind bolt system is designed for composite structures and is available in titanium and stainless steel. The UAB™ blind bolt system is available in 100° flush head, 130° flush head, and protruding head styles. The Accu-Lok™ Blind Fastening System is designed specifically for use in composite structures in which access is limited to one side of the structure. It combines high joint preload with a large diameter footprint on the blind side. The large footprint enables distribution of the joint preload over a larger area, virtually eliminating the possibility of delaminating the composite structure. The shear strength of the Accu-Lok™ is 95 KSI, and it is available in 100° flush

head, 130° flush head, and protruding head styles. A similar fastener designed by Monogram is called the Radial-Lok®. (***Figure 3-89***)

Fiberlite

The fiberlite fastening system uses composite materials for a wide range of aerospace hardware. The strength of fiberlite fasteners is equivalent to aluminum at two-thirds the weight. The composite fastener provides good material compatibility with carbon fiber and fiberglass.

Screws and Nutplates in Composite Structures

The use of screws and nutplates in place of Hi-Loks® or blind fasteners is recommended if a panel must be removed periodically for maintenance. Nutplates used in composite structures usually require three holes: two for attachment of the nutplate and one for the removable screw, although rivetless nut plates and adhesive bonded nutplates are available that do not require drilling and countersinking two extra holes.

MACHINING PROCESSES AND EQUIPMENT

Drilling

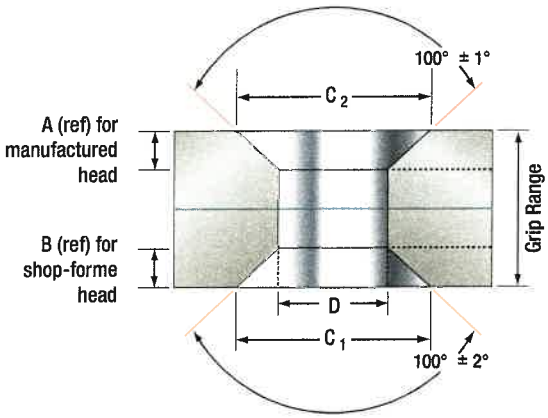
Hole drilling in composite materials is different from drilling holes in metal aircraft structures. Different types of drill bits, higher speeds, and lower feeds are required to drill precision holes. Structures made from carbon fiber and epoxy resin are very hard and abrasive, requiring special flat flute drills or similar four flute drills. Aramid fiber (Kevlar®) epoxy composites are not as hard as carbon but are difficult to drill unless special cutters are used because the fibers tend to fray or shred unless they are cut clean while embedded in the epoxy.

Rivet Diameter	A REF	CSR 90433			
		B REF	C1 DIA	C2 DIA	D DIA
1/8 (–4)	0.028	0.028	0.195 0.189	0.195 0.189	0.132 0.129
5/32 (–5)	0.037	0.037	0.247 0.242	0.247 0.242	0.162 0.159
3/16 (–6)	0.046	0.046	0.302 0.297	0.302 0.297	0.195 0.191
7/32 (–7)	0.046	0.046	0.328 0.323	0.328 0.323	0.227 0.224

Hollow End E-Z Buck® Nominal Diameter	Upset Load (Lb) + 200 Lb
1/8" (–4)	2 500
5/32" (–5)	2 700
3/16" (–6)	3 000
7/32" (–7)	3 750

Cherry Flaring Snap Die Part Numbers

Rivet Diameter	3/16" Diameter Mount	1/4" Diameter Mount
1/8"	839B1- 4	839B10- 4
5/32"	839B1- 5	839B10- 5
3/16"	839B1- 6	839B10- 6
7/32"	839B1- 7	839B10- 7



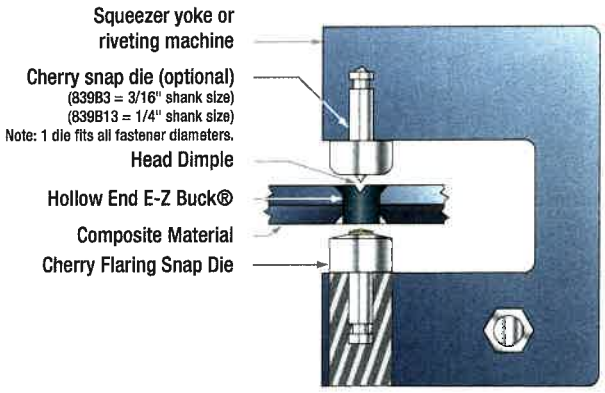




Figure 3-87. Cherry's E-Z buck hollow rivet.

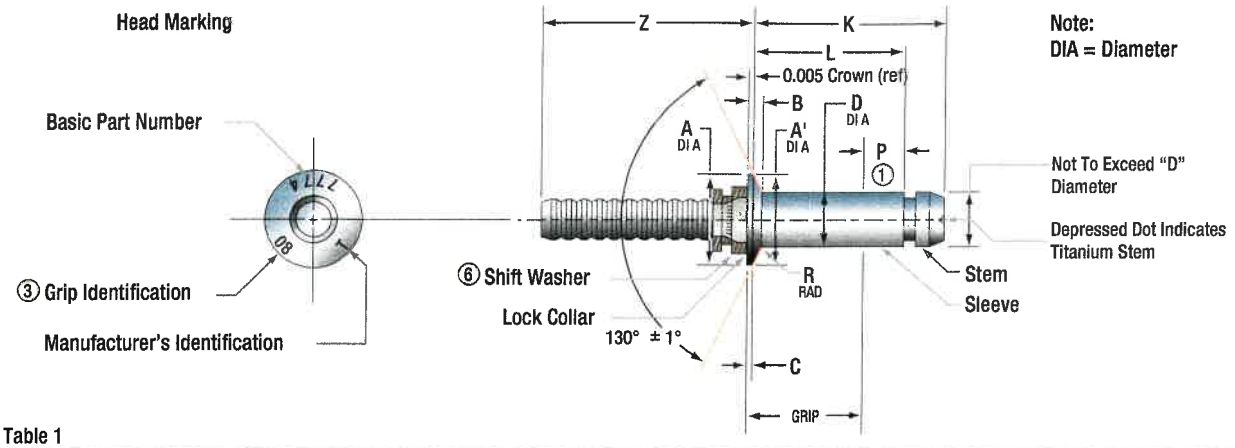


Table 1

Dia. Dash No.	D ± 0.001 ⑥	A Max	A' Min	B Max	P Max	R Max	Z Min	Hole Limits	Installed Strength (Lb) ④	
									Single Shear Minimum ⑤	Tensile Minimum
–05	0.163	0.333	0.296	0.039	0.215	0.025	0.844	0.164/0.167	1980	900
–06	0.198	0.386	0.342	0.043	0.250	0.025	0.875	0.199/0.202	2925	1400
–06	0.259	0.507	0.463	0.057	0.305	0.030	1.000	0.260/0.263	5005	2100

Table 2

Grip Dash No.	Grip Limits				–05 Diameter		Grip Limits				–06 Diameter		–08 Diameter	
	Overlap Min	1/16 Range ④		Overlap Max			Overlap Min	1/16 Range ④		Overlap Max				
		Min	Max		L Ref	K Max		Min	Max		L Ref	K Max	L Ref	K Max
–02	—	0.094	0.157	0.173	0.336	0.476	—	0.120	0.157	0.173	0.355	0.521	—	—
–03	0.146	0.154	0.220	0.236	0.398	0.536	⑨	0.156	0.220	0.236	0.417	0.584	0.479	0.645
–04	0.209	0.219	0.282	0.298	0.460	0.602	0.203	0.219	0.282	0.298	0.480	0.647	0.541	0.708
–05	0.271	0.281	0.345	0.361	0.523	0.664	0.265	0.281	0.345	0.361	0.542	0.709	0.604	0.770
–06	0.334	0.344	0.407	0.423	0.585	0.727	0.328	0.344	0.407	0.423	0.605	0.772	0.666	0.833
–07	0.396	0.406	0.470	0.486	0.648	0.789	0.390	0.406	0.470	0.486	0.667	0.834	0.729	0.895
–08	0.459	0.469	0.532	0.548	0.710	0.852	0.453	0.469	0.532	0.548	0.730	0.897	0.791	0.958
–09	0.521	0.531	0.595	0.611	0.773	0.914	0.515	0.531	0.595	0.611	0.792	0.959	0.854	1.020
–10	0.584	0.594	0.657	0.673	0.835	0.977	0.578	0.594	0.657	0.673	0.855	1.022	0.916	1.083
–11	0.646	0.656	0.720	0.736	0.898	1.039	0.640	0.656	0.720	0.736	0.917	1.084	0.979	1.145
–12	0.709	0.719	0.782	0.798	0.960	1.102	0.703	0.719	0.782	0.798	0.980	1.147	1.041	1.208

Figure 3-88. Cherry's titanium Maxibolt.

Special drill bits with close pin points and fish tail points have been developed that slice the fibers prior to pulling them out of the drilled hole. If the Kevlar®/epoxy part is sandwiched between two metal parts, standard twist drills can be used.

Equipment

Air driven tools are used for drilling holes in composite materials. Drill motors with free speed of up to 20 000 rpm are used. A general rule for drilling composites is to use high speed and a low feed rate (pressure). Drilling equipment with a power feed control produces better hole quality than drill motors without power feed control. Drill guides are recommended, especially for thicker laminates. Do not use standard twist drill bits for drilling composite structures. Standard high speed steel is unacceptable, because it dulls immediately, generates excessive heat, and causes ply delamination, fiber tear-out, and unacceptable hole quality.

Drill bits used for carbon fiber and fiberglass are made from diamond coated material or solid carbide because the fibers are so hard that standard high-speed steel (HSS) drill bits do not last long. Typically, twist drills are used, but brad point drills are also available. The Kevlar® fibers are not as hard as carbon, and standard HSS drill bits can be used. The hole quality can be poor if standard drill bits are used and the preferred drill style is the sickle-shaped Klenk drill. This drill first pulls on the fibers and then shears them, which results in a better

quality hole. Larger holes can be cut with diamond coated hole saws or fly cutters, but only use fly cutters in a drill press, and not in a drill motor.
(***Figures 3-90, 3-91, and 3-92***)

Processes and Precautions
Composite materials are drilled with drill motors operating between 2 000 and 20 000 rpm and a low feed rate. Drill motors with a hydraulic dash pod or other type of feed control are preferred because they restrict the surging of the drill as it exits the composite materials. This reduces breakout damage and delaminations. Parts made from tape products are especially susceptible to breakout damage; parts made from fabric material have experienced less damage. The composite structure needs to be backed with a metal plate or sheet to avoid breakout. Holes in composite structures are often predrilled with a small pilot hole, enlarged with a diamond coated or carbide drill bit and reamed with a carbide reamer to final hole size.

Back counterboring is a condition that can occur when carbon/epoxy parts mate metal substructure parts. The back edge of the hole in the carbon/epoxy part can be eroded or radiused by metal chips being pulled through the composite. The condition is more prevalent when there are gaps between the parts or when the metal debris is stringy rather than small chips. Back counterboring can be minimized or eliminated by changing feeds and speeds, cutter geometry, better part clamp-up adding a final ream pass, using a peck drill, or combination of these.

When drilling combinations of composite parts with metal parts, the metal parts may govern the drilling speed. For example, even though titanium is compatible with carbon/epoxy material from a corrosion perspective, lower drilling speeds are required in order to ensure no metallurgical damage occurs to the titanium. Titanium is drilled with low speed and high feed. Drill bits suitable for titanium might not be suitable for carbon or fiberglass.

Drill bits that are used for drilling titanium are often made from cobalt-vanadium; drill bits used for carbon fiber are made from carbide or are diamond coated to increase drill life and to produce an accurate hole. Small diameter High Speed Steel (HSS) drill bits, such as No. 40 drill, which are used to manually drill pilot holes, are typically used because carbide drills are relatively brittle and are easily broken. The relatively low cost of these small HSS drill bits offsets the limited life expectancy. High-speed steel drill bits may last for only one hole.

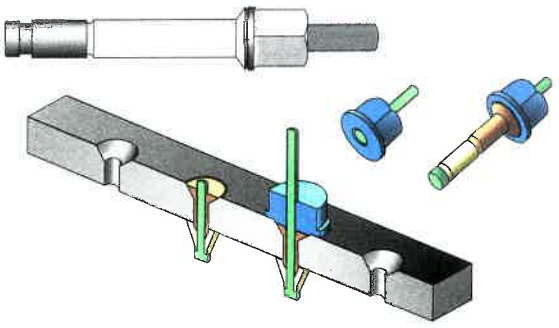


Figure 3-89. Accu-Lok™ installation.



Figure 3-90. Klenk-type drill for drilling Kevlar®.



Figure 3-91. Drilling and cutting tools for composite materials.

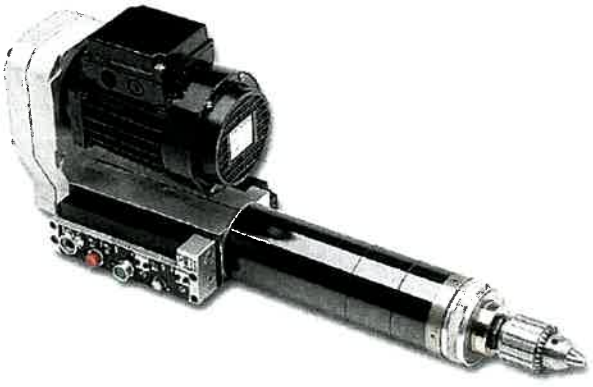


Figure 3-92. Autofeed drill.

The most common problem with carbide cutters used in hand drill operations is handling damage (chipped edges) to the cutters. A sharp drill with a slow constant feed can produce a 0.1mm tolerance hole through carbon/epoxy plus thin aluminum, especially if a drill guide is used. With hard tooling, tighter tolerances can be maintained. When the structure under the carbon/epoxy is titanium, drills can pull titanium chips through the carbon/epoxy and enlarge the hole. In this case, a final ream operation may be required to hold tight hole tolerances. Carbide reamers are needed for holes through carbon/epoxy composite structure. In addition, the exit end of the hole needs good support to prevent splintering and delaminations when the reamer removes more than about 0.13mm on the diameter. The support can be the substructure or a board held firmly against the back surface. Typical reaming speeds are about one half of the drilling speed.

Cutting fluids are not normally used or recommended for drilling thin (less than 6.3mm thick) carbon/epoxy structure. It is good practice to use a vacuum while drilling in composite materials to avoid that carbon dust freely floats around the work area.

Countersinking

Countersinking a composite structure is required when flush head fasteners are to be installed in the assembly. For metallic structures, a 100° included angle shear or tension head fastener has been the typical approach. In composite structures, two types of fastener are commonly used: a 100° included angle tension head fastener or a 130° included angle head fastener. The advantage of the 130° head is that the fastener head can have about the same diameter as a tension head 100° fastener with the head depth of a shear type head 100° fastener. For seating flush fasteners in composite parts, it is recommended that the countersink cutters be designed to produce a controlled radius between the hole and the countersink to accommodate the head to shank fillet radius on the fasteners. In addition, a chamfer operation or a washer may be required to provide proper clearance for protruding head fastener head to shank radii. Whichever head style is used, a matching countersink/chamfer must be prepared in the composite structure.

Carbide cutters are used for producing a countersink in carbon/epoxy structure. These countersink cutters usually have straight flutes similar to those used on metals. For Kevlar® fiber/epoxy composites, S-shaped positive rake cutting flutes are used. If straight fluted countersink cutters are used, a special thick tape can be applied to the surface to allow for a clean cutting of the Kevlar® fibers, but this is not as effective as the S-shaped fluted cutters. Use of a piloted countersink cutter is recommended because it ensures better concentricity between the hole and the countersink and decreases the possibility of gaps under the fasteners due to misalignment or delaminations of the part.

Use a microstop countersink gauge to produce consistent countersink wells. Do not countersink through more than 70 percent of the skin depth because a deeper countersink well reduces material strength. When a piloted countersink cutter is used, the pilot must be periodically checked for wear, as wear can cause reduction of concentricity between the hole and countersink. This is especially true for countersink cutters with only one cutting edge. For piloted countersink cutters, position the pilot in the hole and bring the cutter to full rpm before beginning to feed the cutter into the hole and preparing the countersink. If the cutter is in contact with the composite before triggering the drill motor, you may get splintering.

Cutting Processes

Cutters that work well for metals would either have a short life or produce a poorly cut edge if used for composite materials. The cutters that are used for composites vary with the composite material that is being cut. The general rule for cutting composites is high speed and slow feed:

- Carbon fiber reinforced plastics: Carbon fiber is very hard and quickly wears out high speed steel cutters. For most trimming and cutting tasks, diamond grit cutters are best. Aluminum oxide or silicon carbide sandpaper or cloth is used for sanding. Silicon carbide lasts longer then aluminum oxide. Router bits can also be made from solid carbide or diamond coated.
- Glass fiber reinforced plastics: Glass fibers, like carbon, are very hard and quickly wear out high speed steel cutters. Fiberglass is drilled with the same type and material drill bits as carbon fiber.
- Aramid (Kevlar®) fiber reinforced plastics: Aramid fiber is not as hard as carbon and glass fiber, and cutters made from high-speed steel can be used. To prevent loose fibers at the edge of aramid

composites, hold the part and then cut with a shearing action. Aramid composites need to be supported with a plastic backup plate. The aramid and backup plate are cut through at the same time. Aramid fibers are best cut by being held in tension and then sheared. There are specially shaped cutters that pull on the fibers and then shear them. When using scissors to cut aramid fabric or prepreg, they must have a shearing edge on one blade and a serrated or grooved surface on the other. These serrations hold the material from slipping. Sharp blades should always be used as they minimize fiber damage. Always clean the scissor serrations immediately after use so the uncured resin does not ruin the scissors. Always use safety glasses and other protective equipment when using tools and equipment.

Cutting Equipment

The bandsaw is the equipment that is most often used in a repair shop for cutting composite materials. A toothless carbide or diamond coated saw blade is recommended. A typical saw blade with teeth does not last long if carbon fiber or fiberglass is cut. (***Figure 3-93***)

Air driven hand tools, such as routers, saber saws, die grinders, and cut off wheels can be used to trim composite parts. Carbide or diamond coated cutting tools produce a better finish and they last much longer. Specialized shops have ultrasonic, waterjet, and laser cutters. These types of equipment are Numerical Controlled (NC) and produce superior edge and hole quality. A waterjet cutter cannot be used for honeycomb structure because it introduces water in the part. Do not cut anything else on equipment that is used for composites because other materials can contaminate the composite material.



Figure 3-93. Diamond coated saw blade.

Prepreg materials can be cut with a CNC Gerber table. The use of this equipment speeds up the cutting process and optimizes the use of the material. Design software is available that calculates how to cut plies for complex shapes. (***Figures 3-94***)

REPAIR SAFETY

Advanced composite materials including prepreg, resin systems, cleaning solvents, and adhesives could be hazardous, and it is important that you use personal protection equipment. It is important to read and understand the Material Safety Data Sheets (MSDS) and handle all chemicals, resins, and fibers correctly. The MSDS lists the hazardous chemicals in the material system, and it outlines the hazards. The material could be a respiratory irritant or carcinogenic, or another kind of dangerous substance.

Eye Protection

Always protect eyes from chemicals and flying objects. Wear safety glasses at all times and, when mixing or pouring acids, wear a face shield. Never wear corrective contact lenses in a shop, even with safety glasses. Some of the chemical solvents can melt the lenses and damage eyes. Dust can also get under the lenses, causing damage.

Respiratory Protection

Do not breathe carbon fiber dust and always ensure that there is a good flow of air where the work is performed. Always use equipment to assist in breathing when working in a confined space. Use a vacuum near the source of the dust to remove the dust from the air. When sanding or applying paint, you need a dust mask



Figure 3-94. Gerber cutting table.

or a respirator. A properly fitted dust mask provides the protection needed. For application of paints, a sealed respirator with the correct filters or a fresh air supply respirator is required.

Downdraft Tables

A downdraft table is an efficient and economical device for protecting workers from harmful dust caused by sanding and grinding operations. The tables are also useful housekeeping tools because the majority of particulate material generated by machining operations is immediately collected for disposal. Downdraft tables should be sized and maintained to have an average face velocity between 100 and 150 cubic feet per minute. The downdraft table draws contaminants like dust and fibers away from the operator's material. Downdraft tables should be monitored and filters changed on a regular basis to provide maximum protection and particulate collection.

Skin Protection

During composite repair work, protect your skin from hazardous materials. Chemicals could remain on hands that burn sensitive skin. Always wear gloves and clothing that offer protection against toxic materials. Use only approved gloves that protect skin and do not contaminate the composite material. Always wash hands prior to using the toilet or eating. Damaged composite components should be handled with care. Single fibers can easily penetrate the skin, splinter off, and become lodged in the skin.

Fire Protection

Most solvents are flammable. Close all solvent containers and store in a fireproof cabinet when not in use. Make sure that solvents are kept away from areas where static electricity can occur. Static electricity can occur during sanding operations or when bagging material is unrolled. It is preferable to use air driven tools. If electric tools are used, ensure that they are the enclosed type. Do not mix too much resin. The resin could overheat and start smoking caused by the exothermic process. Ensure that a fire extinguisher is always nearby.

6.3.2 - WOODEN STRUCTURES

Wood was among the first materials used to construct aircraft. Most of the airplanes built during World War I (WWI) were constructed of wood frames with fabric coverings. Wood was the material of choice for aircraft construction into the 1930s. Part of the reason was the slow development of strong, lightweight, metal aircraft structures and the lack of suitable corrosion resistant materials for all metal aircraft.

In the late 1930s, the British airplane company DeHavilland designed and developed a bomber named the Mosquito. Well into the late 1940s, DeHavilland produced more than 7 700 airplanes made of spruce, birch plywood, and balsa wood. (***Figure 3-95***)

During the early part of World War II (WWII), the U.S. government put out a contract to build three flying boats. Hughes Aircraft ultimately won the contract with the mandate to use only materials not critical to the war, such as aluminum and steel. Hughes designed the aircraft to be constructed out of wood.

After many delays and loss of government funding, Howard Hughes continued construction, using his own money and completing one aircraft. On November 2, 1947, during taxi tests in the harbor at Long Beach, California, Hughes piloted the Spruce Goose for over a mile at an altitude of 70 feet, proving it could fly.

This was the largest seaplane and the largest wooden aircraft ever constructed. Its empty weight was 300 000 pounds with a maximum takeoff weight of 400 000 pounds. The entire airframe, surface structures, and flaps were composed of laminated wood with fabric covered primary control surfaces. It was powered by eight Pratt & Whitney R-4360 radial engines, each producing 3 000 horsepower. (***Figure 3-96***)

As the aircraft design and manufacturing evolved, the development of lightweight metals and the demand for increased production moved the industry away from aircraft constructed entirely of wood. Some general aviation aircraft were produced with wood spars and wings, but today only a limited number of wood aircraft are produced. Most of those are built by their owners for education or recreation and not for production.

Quite a number of airplanes in which wood was used as the primary structural material still exist and are operating, including certificated aircraft that were constructed during the 1930s and later. With the proper maintenance and repair procedures, these older aircraft can be maintained in an airworthy condition and kept operational for many years.

WOOD AIRCRAFT CONSTRUCTION AND REPAIRS

The information presented on wood aircraft and repair is general in nature and should not be regarded as a substitute for specific instructions contained in the aircraft manufacturer's maintenance and repair manuals. Methods of construction vary greatly with different types of aircraft, as do the various repair and maintenance procedures required to keep them airworthy. Be sure to follow all manufacturer's instructions.

The ability to inspect wood structures and recognize defects (dry rot, compression failures, etc.) can be learned through experience and instruction from knowledgeable certificated mechanics and appropriately qualified technical instructors.



Figure 3-95. British DeHavilland Mosquito bomber.



Figure 3-96. Hughes Flying Boat, H-4 Hercules named the Spruce Goose.

CHARACTERISTICS, PROPERTIES AND TYPES OF WOOD AND GLUE IN AEROPLANES

SUITABLE WOOD

Several forms of wood are commonly used in aircraft.

- Solid wood or the adjective "solid" used with such nouns as "beam" or "spar" refers to a member consisting of one piece of wood.
- Laminated wood is an assembly of two or more layers of wood that have been glued together with the grain of all layers or laminations approximately parallel.
- Plywood is an assembled product of wood and glue that is usually made of an odd number of thin plies, or veneers, with the grain of each layer placed 90° with the adjacent ply or plies.
- High-density material includes compreg, impreg, or similar commercially made products, heat stabilized wood, or any of the hardwood plywoods commonly used as bearing or reinforcement plates.

The various species of wood listed in ***Figure 3-97*** are acceptable for structural purposes when used for the repair of aircraft. Spruce is the preferred choice and the standard by which the other wood is measured.

Figure 3-97 provides a comparison of other wood that may be suitable for aircraft repair. It lists the strength and characteristics of the wood in comparison to spruce. The one item common to all the species is that the slope of the grain cannot be steeper than 1:15.

All solid wood and plywood used for the construction and repair of aircraft should be of the highest quality and grade. For certificated aircraft, the wood should have traceability to a source that can provide certification to a Military Specification (MIL-SPEC). The term "aircraft quality" or "aircraft grade" is referred to and specified in some repair documents, but that grade wood cannot be purchased from a local lumber company. To purchase the material, contact one of the specialty aircraft supply companies and request a certification document with the order. The MIL-SPEC for solid spruce is MIL-S-6073 and for plywood it is MIL-P-6070B.

When possible, fabricated wood components should be purchased from the aircraft manufacturer, or someone who may have a Parts Manufacturer Approval (PMA) to produce replacement parts for the aircraft. With either of these sources supplying the wood components, the mechanic can be assured of installing approved material. At the completion of the repair, as always, it is the responsibility of the person returning the aircraft to service to determine the quality of the replacement wood and the airworthiness of the subsequent repair.

To help determine the suitability of the wood, inspect it for defects that would make it unsuitable material to repair or construct an aircraft. The type, location, and amount or size of the defects grade the wood for possible use. All woods used for structural repair of aircraft are classified as softwood. Softwood is typically used for construction and is graded based on strength, load carrying ability, and safety. Hardwoods, on the other hand, are typically appearance woods and are graded based on the number and size of clear cuttings from the tree.

Defects Permitted

The following defects are permitted in the wood species used for aircraft repair that are identified in ***Figure 3-97***:

1. Cross grain—Spiral grain, diagonal grain, or a combination of the two is acceptable if the grain does not diverge from the longitudinal axis of the material more than specified in column 3. A check of all four faces is necessary to determine the amount of divergence. Direction of free flowing ink frequently assists in determining grain direction.
2. Wavy, curly, and interlocked grain-Acceptable, if local irregularities do not exceed limitations specified for spiral and diagonal grain.
3. Hard knots—Sound, hard knots up to ⅜ inch in diameter are acceptable if: (1) they are not projecting portions of I-beams, along the edges of rectangular or beveled unrouted beams, or along the edges of flanges of box beams (except in portions of low stress); (2) they do not cause grain divergence at the edges of the board or in the flanges of a beam more than specified in ***Figure 3-97*** column 3; and (3) they are in the center third of the beam and not closer than 20 inches to another knot or other defect (pertains to ⅜ inch knots; smaller knots may be proportionately closer). Knots greater than ¼ inch must be used with caution.
4. Pin knot clusters—small clusters are acceptable if they produce only a small effect on grain direction.

Species of Wood	Strength Properties (as compared to spruce)	Maximum Permissible Grain Deviation (slope of grain)	Remarks
1	2	3	4
Spruce (Picea) Sitka (P. sitchensis) Red (P. rubra) White (P. glauca)	100%	1.15	Excellent for all uses. Considered standard for this table.
Douglas fir (Pseudotsuga taxifolia)	Exceeds spruce	1.15	May be used as substitute for spruce in same sizes or in slightly reduced sizes if reductions are substantiated. Difficult to work with hand tools. Some tendency to split and splinter during fabrication and much greater care in manufacture is necessary. Large solid pieces should be avoided due to inspection difficulties. Satisfactory for gluing.
Noble fir (Abies procera, also known as Abies nobilis)	Slightly exceeds spruce except 8% deficient in shear	1.15	Satisfactory characteristics of workability, warping, and splitting. May be used as direct substitute for spruce in same sizes if shear does not become critical. Hardness somewhat less than spruce. Satisfactory for gluing.
Western hemlock (Tsuga heterophylla)	Slightly exceeds spruce	1.15	Less uniform in texture than spruce. May be used as direct substitute for spruce. Upland growth superior to lowland growth. Satisfactory for gluing.
Northern white pine, also known as Eastern white pine (Pinus strobus)	Properties between 85% and 96% those of spruce	1.15	Excellent working qualities and uniform in properties, but somewhat low in hardness and shock-resistance. Cannot be used as substitute for spruce without increase in sizes to compensate for lesser strength. Satisfactory for gluing.
Port Orford white cedar (Chamaecyparis lawsoniana)	Exceeds spruce	1.15	May be used as substitute for spruce in same sizes or in slightly reduced sizes if reductions are substantiated. Easy to work with hand tools. Gluing is difficult, but satisfactory joints can be obtained if suitable precautions are taken.
Yellow poplar (Liriodendron tulipifera)	Slightly less than spruce except in compression (crushing) and shear	1.15	Excellent working qualities. Should not be used as a direct substitute for spruce without carefully accounting for slightly reduced strength properties. Somewhat low in shock-resistance. Satisfactory for gluing.

Figure 3-97. Selection and properties of wood for aircraft repairs.

5. Pitch pockets—Acceptable in center portion of a beam if they are at least 14 inches apart when they lie in the same growth ring and do not exceed 1½ inches in length by ⅛ inch width by ⅛ inch depth, and if they are not along the projecting portions of I-beams, along the edges of rectangular or beveled unrouted beams, or along the edges of the flanges of box beams.
6. Mineral streaks—acceptable if careful inspection fails to reveal any decay.

Defects Not Permitted

The following defects are not permitted in wood used for aircraft repair. If a defect is listed as unacceptable, please refer to the previous section.

Defects permitted, for acceptable conditions:

1. Cross grain—*unacceptable.*
2. Wavy, curly, and interlocked grain—*unacceptable.*
3. Hard knots—*unacceptable.*
4. Pin knot clusters—*unacceptable, if they produce large effect on grain direction.*
5. Spike knots—knots running completely through the depth of a beam perpendicular to the annual rings and appear most frequently in quarter sawed lumber. Reject wood containing this defect.
6. Pitch pockets—*unacceptable.*
7. Mineral streaks—*unacceptable, if accompanied by decay.*
8. Checks, shakes, and splits—checks are longitudinal cracks extending, in general, across the annual

rings. Shakes are longitudinal cracks usually between two annual rings. Splits are longitudinal cracks caused by artificially induced stress. Reject wood containing these defects.

9. Compression—very detrimental to strength and is difficult to recognize readily, compression wood is characterized by high specific gravity, has the appearance of an excessive growth of summer wood, and in most species shows little contrast in color between spring wood and summer wood. If in doubt, reject the material or subject samples to toughness machine test to establish the quality of the wood. Reject all material containing compression wood.
10. Compression failures—caused from overstress in compression due to natural forces during the growth of the tree, felling trees on rough or irregular ground, or rough handling of logs or lumber. Compression failures are characterized by a buckling of the fibers that appears as streaks substantially at right angles to the grain on the surface of the piece, and vary from pronounced failures to very fine hairlines that require close inspection to detect. Reject wood containing obvious failures. If in doubt, reject the wood or make a further inspection in the form of microscopic examination or toughness test, the latter being more reliable.
11. Tension—forming on the upper side of branches and leaning trunks of softwood trees, tension wood is caused by the natural overstressing of trying to pull the branches and leaning trunk upright. It is typically harder, denser, and may be darker in color than normal wood, and is a serious defect, having higher than usual longitudinal shrinkage that may break down due to uneven shrinkage. When in doubt, reject the wood.
12. Decay—rot, dote, red heart, purple heart, etc., must not appear on any piece. Examine all stains and discoloration carefully to determine whether or not they are harmless or in a stage of preliminary or advanced decay.

GLUES (ADHESIVES)

Because adhesives play a critical role in the bonding of aircraft structure, the mechanic must employ only those types of adhesives that meet all of the performance requirements necessary for use in certificated aircraft. The product must be used strictly in accordance with the aircraft and adhesive manufacturer's instructions. All instructions must be followed exactly, including the mixing ratios, the ambient and surface temperatures, the open and closed assembly times, the gap filling ability, or glue line thickness, the spread of the adhesive, whether one or two surfaces, and the amount of clamping pressure and time required for full cure of the adhesive.

Refer to the aircraft maintenance or repair manual for specific instructions on acceptable adhesive selection for use on that type aircraft. Adhesives meeting the requirements of a MILSPEC, Aerospace Material Specification (AMS), or Technical Standard Order (TSO) for wooden aircraft structures are satisfactory, providing they are found to be compatible with existing structural materials in the aircraft and fabrication methods to be used in the repair.

New adhesives have been developed in recent years, and some of the older ones are still in use. Some of the more common adhesives that have been used in aircraft construction and repair include casein glue, plastic resin glue, resorcinol glue, and epoxy adhesives.

Casein glue should be considered obsolete for all aircraft repairs. The adhesive deteriorates when exposed to moisture and temperature variations that are part of the normal operating environment of any aircraft.

NOTE: Some modern adhesives are incompatible with casein adhesive. If a joint that has previously been bonded with casein is to be reglued using another type adhesive, all traces of the casein must be scraped off before a new adhesive is applied. If any casein adhesive is left, residual alkalinity may cause the new adhesive to fail to cure properly.

Plastic resin glue, also known as a urea-formaldehyde adhesive, came on the market in the middle to late 1930s. Tests and practical applications have shown that exposure to moist conditions, and particularly to a warm humid environment, under swell shrink stress, leads to deterioration and eventual failure of the bond. For these reasons, plastic resin glue should be considered obsolete for all aircraft repairs.

Resorcinol glue, or resorcinol-formaldehyde glue, is a two component synthetic adhesive consisting of resin and a catalyst. It was first introduced in 1943 and

almost immediately found wide application in the wood boatbuilding and wood aircraft industry in which the combination of high durability and moderate temperature curing was extremely important. It has better wet weather and Ultraviolet (UV) resistance than other adhesives. This glue meets all strength and durability requirements if the fit of the joint and proper clamping pressure results in a very thin and uniform bond line.

The manufacturer's product data sheets must be followed regarding mixing, usable temperature range, and the open and close assembly times. It is very important that this type of glue is used at the recommended temperatures because the full strength of the joint cannot be relied on if assembly and curing temperatures are below 21°C. With that in mind, higher temperatures shorten the working life because of a faster cure rate, and open and closed assembly times must be shortened.

Epoxy adhesive is a two-part synthetic resin product that depends less on joint quality and clamping pressure. However, many epoxies have not exhibited joint durability in the presence of moisture and elevated temperatures and are not recommended for structural aircraft bonding unless they meet the acceptable standards.

Definition of Terms Used in the Glue Process

- Close contact adhesive—a non-gap-filling adhesive (e.g., resorcinol-formaldehyde glue) suitable for use only in those joints where the surfaces to be joined can be brought into close contact by means of adequate pressure, to allow a glue line of no more than 0.005 inch gap.
- Gap-filling adhesive—an adhesive suitable for use in those joints in which the surfaces to be joined may not be close or in continuous contact (e.g., epoxy adhesives) due either to the impracticability of applying adequate pressure or to the slight inaccuracies of fabricating the joint.
- Glue line—resultant layer of adhesive joining any two adjacent wood layers in the assembly.
- Single spread—spread of adhesive to one surface only.
- Double spread—spread of adhesive to both surfaces and equally divided between the two surfaces to be joined.
- Open assembly time—period of time between the application of the adhesive and the assembly of the joint components.
- Closed assembly time—time elapsing between the assembly of the joints and the application of pressure.
- Pressing or clamping time—time during which the components are pressed tightly together under recommended pressure until the adhesive cures (may vary from 10 to 150 pounds per square inch (psi) for softwoods, depending on the viscosity of the glue).
- Caul—a clamping device, usually two rigid wooden bars, to keep an assembly of flat panel boards aligned during glue-up. It is assembled with long bolts and placed on either side of the boards, one on top and another below, and parallel with the pipe/bar clamps. A caul is usually finished and waxed before each use to keep glue from adhering to it.
- Adhesive pot life—time elapsed from the mixing of the adhesive components until the mixture must be discarded, because it no longer performs to its specifications. The manufacturer's product data sheet may define this as working time or useful life; once expired, the adhesive must not be used. It lists the specific temperature and quantity at which the sample amount can be worked. Pot life is a product of time and temperature. The cooler the mix is kept, within the recommended temperature range, the longer it is usable.

Preparation of Wood for Gluing

Satisfactory glue joints in aircraft should develop the full strength of the wood under all conditions of stress. To produce this result, the conditions involved in the gluing operation must be carefully controlled to obtain a continuous, thin, uniform film of solid glue in the joint with adequate adhesion to both surfaces of the wood. These conditions required:

1. Proper and equal moisture content of wood to be joined (8 to 12 percent).
2. Properly prepared wood surfaces that are machined or planed, and not sanded or sawed.
3. Selection of the proper adhesive for the intended task, which is properly prepared and of good quality.
4. The application of good gluing techniques, including fitment, recommended assembly times, and adequate equal pressure applied to the joint.
5. Performing the gluing operation under the recommended temperature conditions.

The surfaces to be joined must be clean, dry, and free from grease, oil, wax, paint, etc. Keep large prepared surfaces covered with a plastic sheet or masking paper prior to the bonding operation. It is advisable to clean all surfaces with a vacuum cleaner just prior to adhesive application.

Smooth even surfaces produced on planers and joiners with sharp knives and correct feed adjustments are the best surfaces for gluing solid wood. The use of sawn surfaces for gluing has been discouraged for aircraft component assembly because of the difficulty in producing a surface free of crushed fibers. Glue joints made on surfaces that are covered with crushed fibers do not develop the normal full strength of the wood.

Some of the surface changes in plywood, such as glazing and bleed-through, that occur in manufacture and may interfere with the adhesion of glue in secondary gluing are easily recognized. A light sanding of the surface with 220-grit sandpaper in the direction of the grain restores the surface fibers to their original condition, removes the gloss, and improves the adhesion of the glue. In contrast to these recognized surface conditions, wax deposits from cauls used during hot pressing produce unfavorable gluing surfaces that are not easily detected.

Wetting tests are a useful means of detecting the presence of wax. A finely sprayed mist or drops of water on the surface of wax coated plywood bead and do not wet the wood. This test may also give an indication of the presence of other materials or conditions that would degrade a glue joint. Only a proper evaluation of the adhesion properties, using gluing tests, determines the gluing characteristics of the plywood surfaces.

Preparing Glues for Use

The manufacturer's directions should be followed for the preparation of any glue or adhesive. Unless otherwise specified by the glue manufacturer, clear, cool water should be used with glues that require mixing with water. The recommended proportions of glue, catalyst, and water or other solvent should be determined by the weight of each component. Mixing can be either by hand or machine. Whatever method is used, the glue should be thoroughly mixed and free of air bubbles, foam, and lumps of insoluble material.

Applying the Glue/Adhesive

To make a satisfactorily bonded joint, it is generally desirable to apply adhesive to both surfaces and join in a thin even layer. The adhesive can be applied with a brush, glue spreader, or a grooved rubber roller. Follow the adhesive manufacturer's application instructions for satisfactory results. Be careful to ensure the surfaces make good contact and the joint is positioned correctly before applying the adhesive. Keep the open assembly time as short as possible and do not exceed the recommended times indicated in the product data sheet.

Pressure on the Joint

To ensure the maximum strength of the bonded surfaces, apply even force to the joint. Non-uniform gluing pressure commonly results in weak areas and strong areas in the same joint. The results of applied pressure are illustrated in ***Figure 3-98***.

Use pressure to squeeze the glue out into a thin continuous film between the wood layers, to force air from the joint, to bring the wood surfaces into intimate contact with the glue, and to hold them in this position during the setting of the glue. Pressure may be applied by means of clamps, elastic straps, weight, vacuum bags, or other mechanical devices. Other methods used to apply pressure to joints in aircraft gluing operations range from the use of brads, nails, and screws to the use of electric and hydraulic power presses.

The amount of pressure required to produce strong joints in aircraft assembly operations may vary from 10 to 150 psi for softwoods and as high as 200 psi for hardwoods.

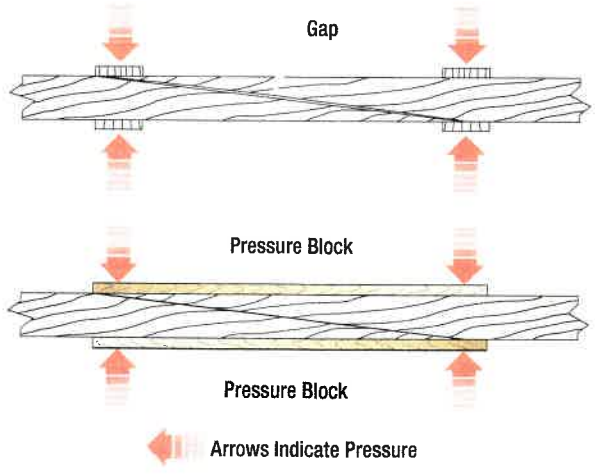


Figure 3-98. Even distribution of gluing pressure creates a strong, gap-free joint.

Glue	Gluing Pressure	Type of Assembly	Maximum Assembly Time
Resorcinol resins	100–250 psi	Closed	Up to 50 minutes
	100–250 psi	Open	Up to 12 minutes
	Less than 100 psi	Closed	Up to 40 minutes
	Less than 100 psi	Open	Up to 10 minutes

Figure 3-99. Even distribution of gluing pressure creates a strong, gap-free joint.

Insufficient pressure to poorly machined or fitted wood joints usually results in a thick glue line, indicating a weak joint, and should be carefully avoided.

High clamping pressure is neither essential nor desirable, provided good contact between the surfaces being joined is obtained. When pressure is applied, a small quantity of glue should be squeezed from the joint. This excess should be removed before it sets. It is important that full pressure be maintained on the joint for the entire cure time of the adhesive because the adhesive does not chemically relink and bond if it is disturbed before it is fully cured.

The full curing time of the adhesive is dependent on the ambient temperature; therefore, it is very important to follow the manufacturer's product data sheets for all phases of the gluing operation from the shelf life to the moisture content of the wood to the proper mixing of the adhesive to the application, and especially to the temperature. The successful assembly and fabrication depends on the workmanship and quality of the joints and following the glue manufacturer's instructions.

All gluing operations should be performed above 21°C for proper performance of the adhesive. Higher temperatures shorten the assembly times, as does coating the pieces of wood with glue and exposing openly to the air. This open assembly promotes a more rapid thickening of the glue than pieces being mated together as soon as the spreading of the glue is completed.

Figure 3-99 provides an example of resorcinol resin glue and the allowable assembly times and gluing pressure when in the open and closed assembly condition. All examples are for an ambient temperature of 24°C.

Figure 3-100 provides examples of strong and weak glue joints resulting from different gluing conditions. A is a well glued joint with a high percentage of wood failure made under proper conditions; B is a glue starved joint resulting from the application of excessive pressure with thin glues; C is a dried glue joint resulting from an excessively long assembly time and/or insufficient pressure.

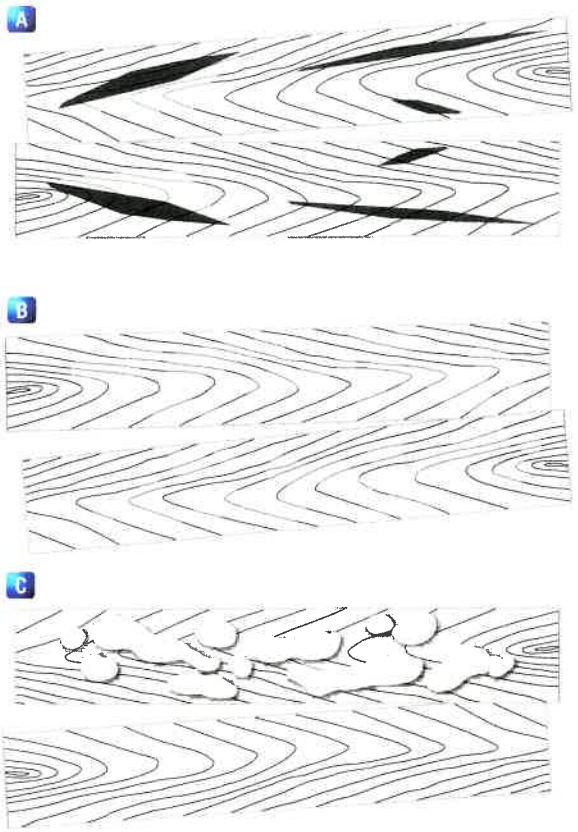


Figure 3-100. Strong and weak glue joints.

Testing Glued Joints

Satisfactory glue joints in aircraft should develop the full strength of the wood under all conditions of stress. Tests should be made by the mechanic prior to gluing a joint of a major repair, such as a wing spar. Whenever possible, perform tests using pieces cut from the actual wood used for the repair under the same mechanical and environmental conditions that the repair will undergo.

Perform a sample test using two pieces of scrap wood from the intended repair, each cut approximately 1" × 2" × 4". The pieces should be joined by overlapping each approximately 2 inches.

The type of glue, pressure, and curing time should be the same as used for the actual repair. After full cure, place the test sample in a bench vise and break the joint by exerting pressure on the overlapping member. The fractured glue faces should show a high percentage of at least 75 percent of the wood fibers evenly distributed over the fractured glue surface. (***Figure 3-101***)

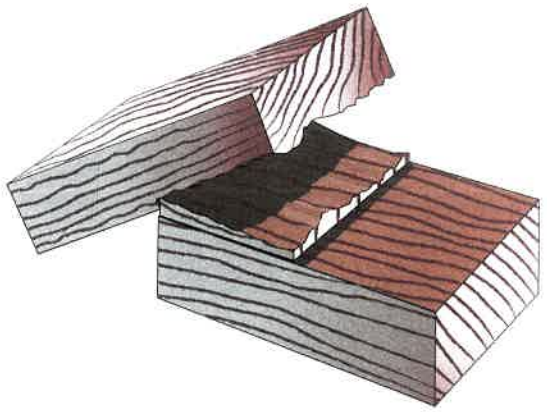


Figure 3-101. An example of good glue joint.

PRESERVATION AND MAINTENANCE OF WOOD FRAME STRUCTURES

WOOD PRESERVATION

All measures which are taken to ensure a long life of wood falls under the definition of wood preservation or timber treatment One method is through the use of wood preservatives such as the common brand shown in ***Figure 3-102***. Wood preservatives must possess the following qualities:

- They must be toxic to fungi, pests, and marine organisms.
- They must be free from objectionable properties in use and handling.
- They must possess satisfactory properties permanence under the conditions for which they are recommended.
- They should not have corrosive properties.



Figure 3-102. A quality wood preserver protects against fungal decay, mold, and insects.

CARE AND MAINTENANCE OF WOOD FRAME STRUCTURES

Wood structures must be kept clean by wiping occasionally with a soft and slightly dampened cloth, and then wiped dry with a clean soft cloth. A good quality furniture spray (such as Endust) is then rubbed into the structure with a soft cloth.

DEFECTS AND INSPECTION IN WOOD STRUCTURES

To properly inspect an aircraft constructed or comprised of wood components, the aircraft must be dry. It should be placed in a dry, well ventilated hanger with all inspection covers, access panels, and removable fairings opened and removed. This allows interior sections and compartments to thoroughly dry. Wet, or even damp, wood causes swelling and makes it difficult to make a proper determination of the condition of the glue joints.

If there is any doubt that the wood is dry, a moisture meter should be utilized to verify the percentage of moisture in the structure. Non-destructive meters are available that check moisture without making holes in the surface. The ideal range is 8–12 percent, with any reading over 20 percent providing an environment for the growth of fungus in the wood.

EXTERNAL AND INTERNAL INSPECTION

The inspection should begin with an examination of the external surface of the aircraft. This provides a general assessment of the overall condition of the wood and structure. The wings, fuselage, and empennage should be inspected for undulation, warping, or any other disparity from the original shape. Where the wings, fuselage, or empennage structure and skins form stressed structures, no departure from the original contour or shape is permissible. (***Figure 3-103***)

Where light structures using single plywood covering are concerned, some slight sectional undulation or bulging between panels may be permissible if the wood and glue are sound. However, where such conditions exist, a careful check must be made of the attachment

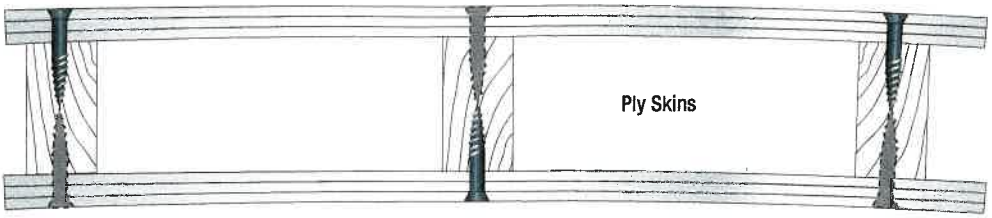


Figure 3-103. Cross sectional view of a stressed skin structure.

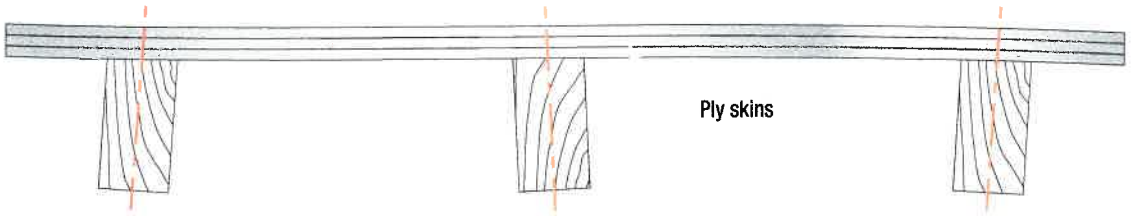


Figure 3-104. A distorted single plywood structure.

of the plywood to its supporting structure. A typical example of a distorted single plywood structure is illustrated in ***Figure 3-104***.

The contours and alignment of leading and trailing edges are of particular importance. A careful check should be made for any deviation from the original shape. Any distortion of these light plywood and spruce structures is indicative of deterioration, and a detailed internal inspection has to be made for security of these parts to the main wing structure. If deterioration is found in these components, the main wing structure may also be affected.

Splits in the fabric covering on plywood surfaces must be investigated to ascertain whether the plywood skin beneath is serviceable. In all cases, remove the fabric and inspect the plywood, since it is common for a split in the plywood skin to initiate a similar defect in the protective fabric covering.

Although a preliminary inspection of the external structure can be useful in assessing the general condition of the aircraft, note that wood and glue deterioration can often take place inside a structure without any external indications. Where moisture can enter a structure, it seeks the lowest point, where it stagnates and promotes rapid deterioration. A musty or moldy odor apparent as you remove the access panels during the initial inspection is a good indication of moisture, fungal growth, and possible decay. Glue failure and wood deterioration are often closely related, and the inspection of glued joints must include an examination of the adjacent wood structure. NOTE: Water need not be present for glue deterioration to take place.

The inspection of a complete aircraft for glue or wood deterioration requires scrutiny of parts of the structure that may be known, or suspected, trouble spots. In many instances, these areas are boxed in or otherwise inaccessible. Considerable dismantling may be required. It may be necessary to cut access holes in some of the structures to facilitate the inspection. Do such work only in accordance with approved drawings or instructions in the maintenance manual for the aircraft concerned. If drawings and manuals are not available, engineering review may be required before cutting access holes.

GLUED JOINT INSPECTION

The inspection of glued joints in wooden aircraft structures presents considerable difficulties. Even where access to the joint exists, it is still difficult to positively assess the integrity of the joint. Keep this in mind when inspecting any glue joint. Some common factors in premature glue deterioration include:

- Chemical reactions of the glue caused by aging or moisture, extreme temperatures, or a combination of these factors, and
- Mechanical forces caused mainly by wood shrinkage, and
- Development of fungal growths.

An aircraft painted in darker colors experiences higher skin temperatures and heat buildup within its structure. Perform a more detailed inspection on a wooden aircraft structure immediately beneath the upper surfaces for signs of deteriorating adhesives.

Aircraft that are exposed to large cyclic changes of temperature and humidity are especially prone to wood shrinkage that may lead to glue joint deterioration. The amount of movement of a wooden member due to these changes varies with the size of each member, the rate of growth of the tree from which it was cut, and the way the wood was converted in relation to the grain.

This means that two major structural members joined to each other by glue are not likely to have identical characteristics. Over a period of time, differential loads are transmitted across the glue joint because the two members do not react identically. This imposes stresses in the glue joint that can normally be accommodated when the aircraft is new and for some years afterwards. However, glue tends to deteriorate with age, and stresses at the glued joints may cause failure of the joints. This is a fact even when the aircraft is maintained under ideal conditions. The various cuts of lumber from a tree have tendency to shrink and warp in the direction(s) indicated in the yellow area around each cut in ***Figure 3-105***.

When checking a glue line, all protective coatings of paint should be removed by careful scraping. It is important to ensure that the wood is not damaged during the scraping operation. Scraping should cease immediately when the wood is revealed in its natural state and the glue line is clearly discernible. At this point, it is important that the surrounding wood is dry; otherwise, you will get a false indication of the integrity of the glue line due to swelling of the wood and subsequent closing of the joint.

Inspect the glue line using a magnifying glass. Where the glue line tends to part, or where the presence of glue cannot be detected or is suspect, probe the glue line with a thin feeler gauge. If any penetration is observed, the joint is defective. The structure usually dictates the feeler gauge thickness, but use the thinnest feeler gauge whenever possible. The illustration indicates the points a feeler gauge should probe. (***Figure 3-106***)

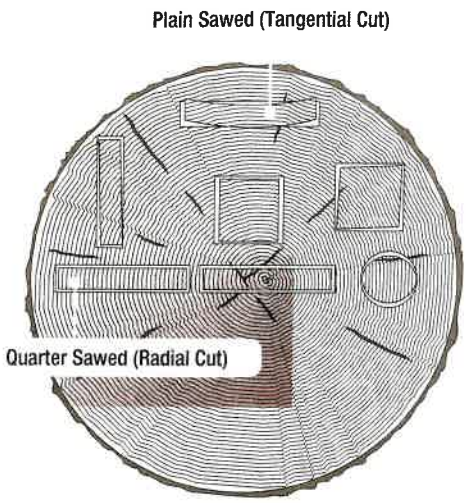


Figure 3-105. Effects of shrinkage on the various shapes during drying from the green condition.

Pressure exerted on a joint either by the surrounding structure or by metal attachment devices, such as bolts or screws, can cause a false appearance of the glue condition. The joint must be relieved of this pressure before the glue line inspection is performed.

A glued joint may fail in service as a result of an accident or because of excessive mechanical loads having been imposed upon it. Glued joints are generally designed to take shear loads. If a joint is expected to take tension loads, it is secured by a number of bolts or screws in the area of tension loading. In all cases of glued joint failure, whatever the direction of loading, there should be a fine layer of wood fibers adhering to the glue. The presence of fibers usually indicates that the joint itself is not at fault.

Examination of the glue under magnification that does not reveal any wood fibers, but shows an imprint of the wood grain, indicates that the cause of the failure was the predrying of the glue before applying pressure during the manufacture of the joint. If the glue exhibits an irregular appearance with star-shaped patterns, this is an indication that precuring of the glue occurred before pressure was applied, or that pressure had been incorrectly applied or maintained on the joint. If there is no evidence of wood fiber adhesion, there may also be glue deterioration.

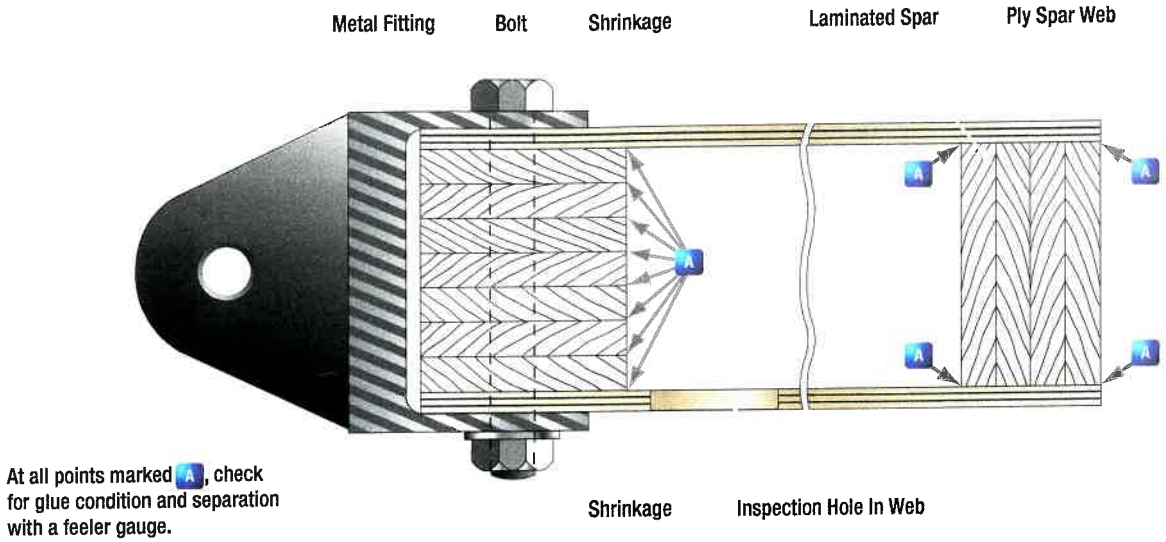


Figure 3-106. Inspection points for laminated glue joints.

WOOD CONDITION

Wood decay and dry rot are usually easy to detect. Decay may be evident as either a discoloration or a softening of the wood. Dry rot is a term loosely applied to many types of decay, but especially to a condition that, in an advanced stage, permits the wood to be crushed to a dry powder. The term is actually a misnomer for any decay, since all fungi require considerable moisture for growth.

Dark discolorations of the wood or gray stains running along the grain are indicative of water penetration. If such discoloration cannot be removed by light scraping, replace the part. Disregard local staining of the wood by dye from a synthetic adhesive hardener.

In some instances where water penetration is suspected, a few screws removed from the area in question reveal, by their degree of corrosion, the condition of the surrounding joint. (***Figure 3-107***)

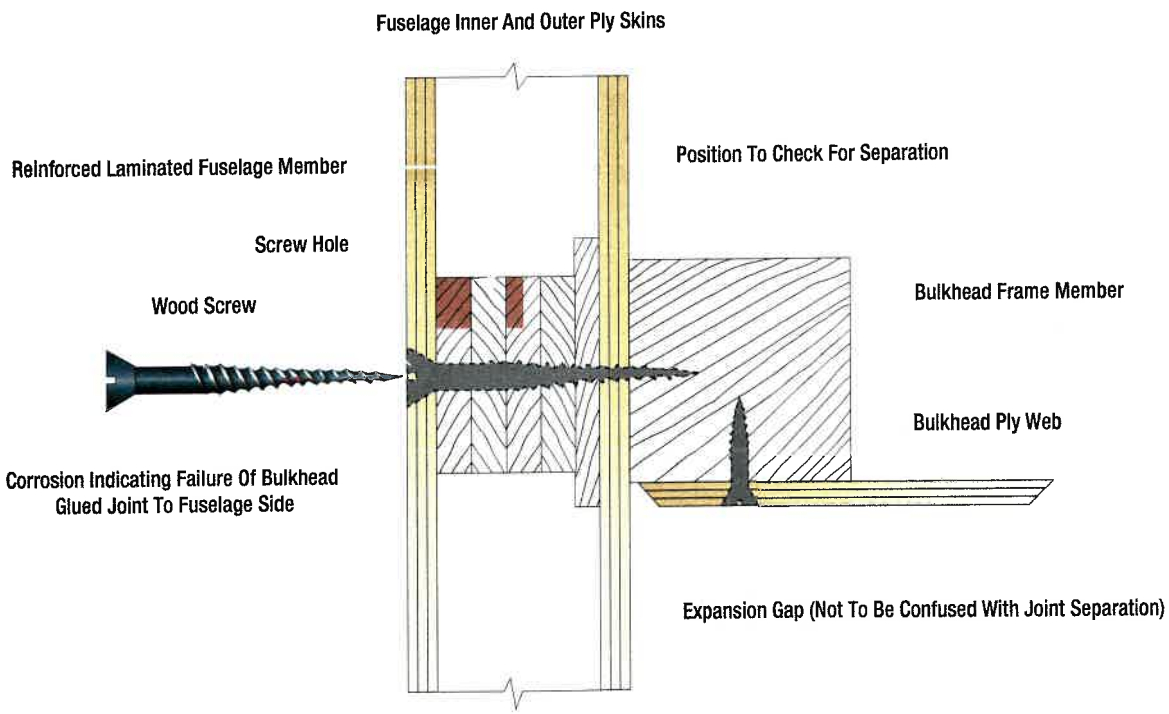


Figure 3-107. Checking a glued joint for water penetration.

Another method of detecting water penetration is to remove the bolts holding the fittings at spar root end joints, aileron hinge brackets, etc. Corrosion on the surface of such bolts and wood discoloration provide a useful indication of water penetration.

Plain brass screws are normally used for reinforcing glued wooden members. For hardwoods, such as mahogany or ash, steel screws may be used. Unless specified by the aircraft manufacturer, replace removed screws with new screws of identical length, but one gauge larger in diameter.

Inspection experience with a particular type of aircraft provides insight to the specific areas most prone to water penetration and moisture entrapment. Wooden aircraft are more prone to the damaging effects of water, especially without the protection of covered storage. Control system openings, fastener holes, cracks or breaks in the finish, and the interfaces of metal fittings and the wood structure are points that require additional attention during an inspection. Additionally, windshield and window frames, the area under the bottom of entrance and cargo doors, and the lower sections of the wing and fuselage are locations that require detailed inspections for water damage and corrosion on all aircraft.

The condition of the fabric covering on plywood surfaces provides an indication of the condition of the wood underneath. If there is any evidence of poor adhesion, cracks in the fabric, or swelling of the wood, remove the fabric to allow further inspection. The exposed surface shows water penetration by the existence of dark gray streaks along the grain and dark discoloration at ply joints or screw holes.

Cracks in wood spars are often hidden under metal fittings or metal rib flanges and leading edge skins. Any time a reinforcement plate exists that is not feathered out on its ends, a stress riser exists at the ends of the plate. A failure of the primary structure can be expected at this point. (***Figure 3-108***)

As part of the inspection, examine the structure for other defects of a mechanical nature, including any location where bolts secure fittings that take load-carrying members, or where the bolts are subject to landing or shear loads. Remove the bolts and examine

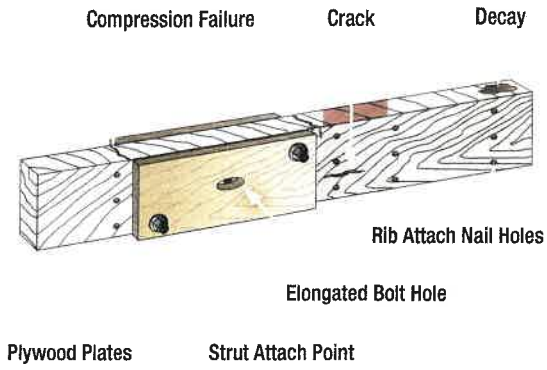


Figure 3-108. Areas likely to incur structural damage.

the holes for elongation or surface crushing of the wood fibers. It is important to ensure the bolts are a good fit in the holes. Check for evidence of bruises or crushing of the structural member, which can be caused by over torquing of the bolts.

Check all metal fittings that are attached to a wood structure for looseness, corrosion, cracks, or bending. Areas of particular concern are strut attach fittings, spar butt fittings, aileron and flap hinges, jury strut fittings, compression struts, control cable pulley brackets, and landing gear fittings. All exposed end grain wood, particularly the spar butts, should be inspected for cracking or checking.

Inspect structural members for compression failures, which is indicated by rupture across the wood fibers. This is a serious defect that can be difficult to detect. If a compression failure is suspected, a flashlight beam shown along the member, and running parallel to the grain, will assist in revealing it. The surface will appear to have minute ridges or lines running across the grain. Particular attention is necessary when inspecting any wooden member that has been subjected to abnormal bending or compression loads during a hard landing. If undetected, compression failures of the spar may result in structural failure of the wing during flight. (***Figure 3-109***)

When a member has been subjected to an excessive bending load, the failure appears on the surface that has been compressed. The surface subject to tension normally shows no defects. In the case of a member taking an excessive direct compression load, the failure is apparent on all surfaces.

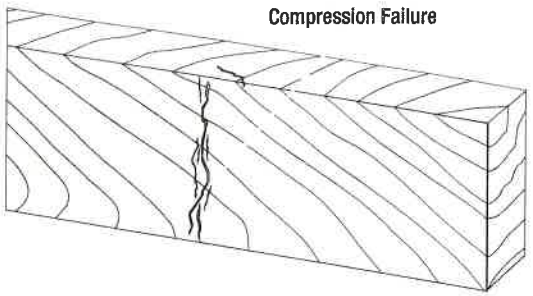


Figure 3-109. Pronounced compression failure in wood beam.

Front and rear spars should be checked for longitudinal cracks at the ends of the plywood reinforcement plates where the lift struts attach. (***Figure 3-108***) Check the ribs on either side of the strut attach points for cracks where the cap strips pass over and under the spars, and for missing or loose rib-to-spar attach nails. All spars, those in the wing(s) and empennage, should be inspected on the face and top surface for compression cracks. A borescope can be utilized by accessing existing inspection holes.

Various mechanical methods can be employed to enhance the visual inspection of wood structures. Tapping the subject area with a light plastic hammer or screwdriver handle should produce a sharp solid sound. If the suspected area sounds hollow and dull, further inspection is warranted. Use a sharp metal awl or thin bladed screwdriver to probe the area. The wood structure should be solid and firm. If the area is soft and mushy, the wood is rotted and disassembly and repair of the structure is necessary.

REPAIR OF WOOD AIRCRAFT STRUCTURES

The standard for any repair is that it should return the aircraft or component to its original condition in strength, function, and aerodynamic shape. It should also be accomplished in accordance with the manufacturer's specifications and/or instructions, or other approved data.

The purpose of repairing all wood structural components is to obtain a structure as strong as the original. Major damage probably requires replacement of the entire damaged assembly, but minor damage can be repaired by removing or cutting away the damaged members and replacing them with new sections. This replacement may be accomplished by gluing, glue and nails, or glue and screw reinforced splicing.

WING RIB REPAIRS

Ribs that have sustained damage may be repaired or replaced, depending upon the type of damage and location in the aircraft. If new parts are available from the aircraft manufacturer or the holder of a parts manufacturer's approval for the part, it is advisable to replace the part rather than to repair it.

If you make a repair to a rib, do the work in such a manner and using materials of such quality that the completed repair is at least equal to the original part in aerodynamic function, structural strength, deterioration, and other qualities affecting airworthiness, such as fit and finish. Follow manufacturer's repair manual instructions when performing the repair.

When necessary, a rib can be fabricated and installed using the same materials and dimensions from a manufacturer approved drawing or by reference to an original rib. However, if you fabricated it from an existing rib, you must provide evidence to verify that the dimensions are accurate and the materials are correct for the replacement part.

You can repair a cap strip of a wood rib using a scarf splice. The repair is reinforced on the side opposite the wing covering by a spruce block that extends beyond the scarf joint not less than three times the thickness of the strips being repaired. Reinforce the entire splice, including the spruce reinforcing block, on each side with a plywood side plate.

The scarf length bevel is 10 times dimension A (thickness of the rib cap strip) with the spruce reinforcement block being 16 times dimension A (the scarf length plus extension on either end of the scarf). The plywood splice plates should be of the same material and thickness as the original plates used to fabricate the rib. The spruce block should have a 5:1 bevel on each end. (***Figure 3-110***)

These specific rib repairs describing the use of one scarf splice implies that either the entire forward or aft portion of the cap strip beyond the damage can be replaced to complete the repair and replace the damaged section. Otherwise, replacement of the damaged section may require a splice repair at both ends of the replaced section of the cap strip using the indicated dimensions for cutting and reinforcing of each splice.

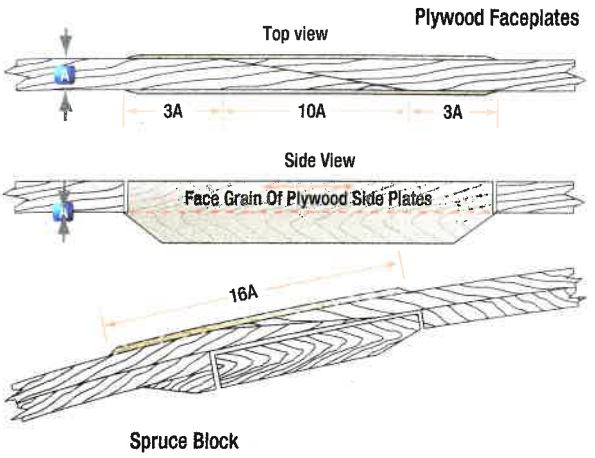


Figure 3-110. A rib cap strip repair.

When a cap strip is to be repaired at a point where there is a joint between it and cross members of the rib, make the repair by reinforcing the scarf joint with plywood gussets, as shown in ***Figure 3-111***.

If a cap strip must be repaired where it crosses a spar, reinforce the joint with a continuous gusset extending over the spar, as shown in ***Figure 3-112***. The scarf joints referred to in the rib repairs are the most satisfactory method of fabricating an end joint between two solid wood members. When the scarf splice is used to repair a solid wood component, the mechanic must be aware of the direction and slope of the grain. To ensure the full strength of the joint, the scarf cut is made in the general direction of the grain on both connecting ends of the wood and then correctly oriented to each other when glued. (***Figure 3-113***)

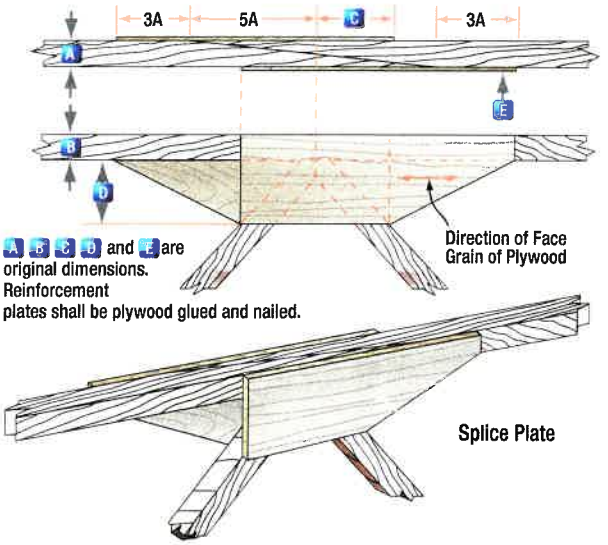


Figure 3-111. Cap strip repair at cross member.

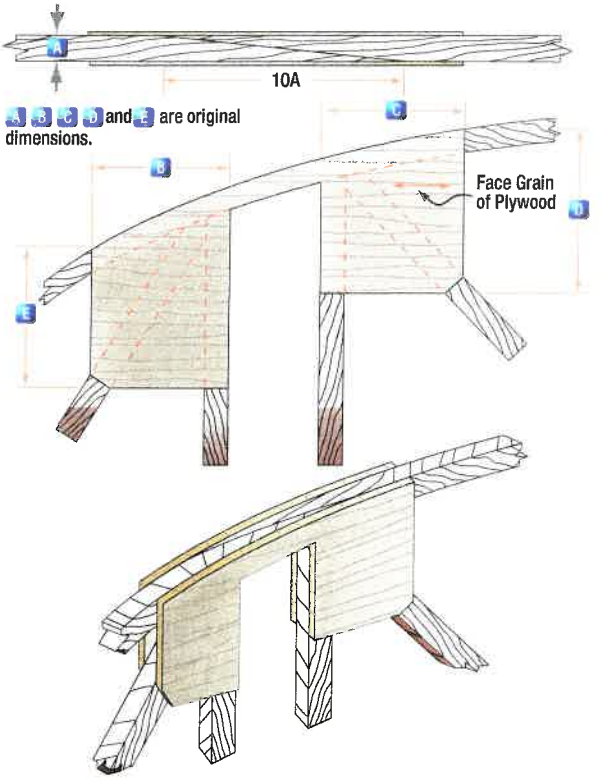


Figure 3-112. Cap strip repair at a spar.

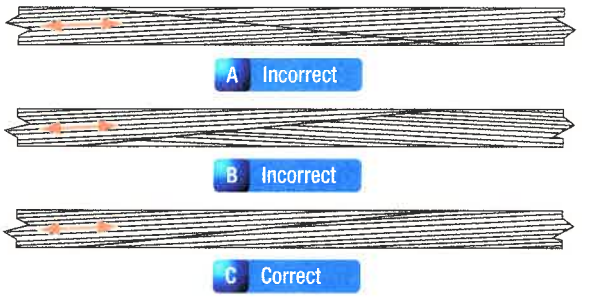


Figure 3-113. Relationship of scarf slope to grain slope.

The trailing edge of a rib can be replaced and repaired by removing the damaged portion of the cap strip and inserting a softwood block of white pine, spruce, or basswood. The entire repair is then reinforced with plywood gussets and nailed and glued, as shown in ***Figure 3-114***. Compression ribs are of many different designs, and the proper method of repairing any part of this type of rib is specified by the manufacturer. All repairs should be performed using recommended or approved practices, materials and adhesives.

Figure 3-115A Illustrates the repair of a compression rib of the I section type (i.e., wide, shallow cap strips, and a center plywood web with a rectangular compression member on each side of the web). The rib damage

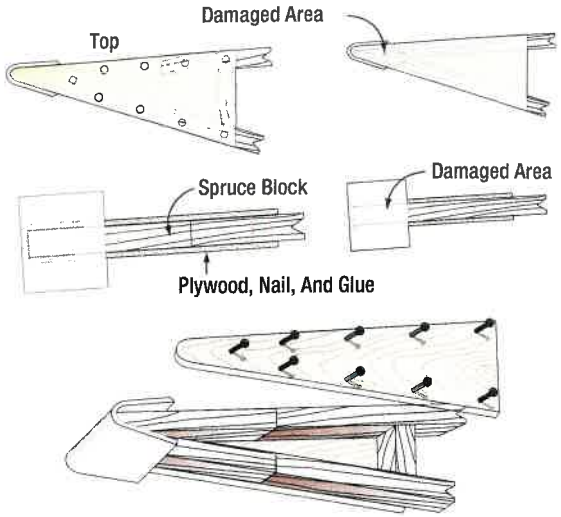


Figure 3-114. Rib trailing edge repair.

suggests that the upper and lower cap strips, the web member, and the compression members are cracked completely through.

To facilitate this repair, cut the compression members as shown in ***Figure 3-115D*** and repair as recommended using replacement sections to the rear spar. Cut the damaged cap strips and repair as shown in ***Figure 3-115***, replacing the aft section of the cap strips. Plywood side plates are then bonded on each side diagonally to reinforce the damaged web as shown in ***Figure 3-115A***.

Figure 3-115B illustrates a compression rib of the type that is a standard rib with rectangle compression members added to one side and a plywood web to the

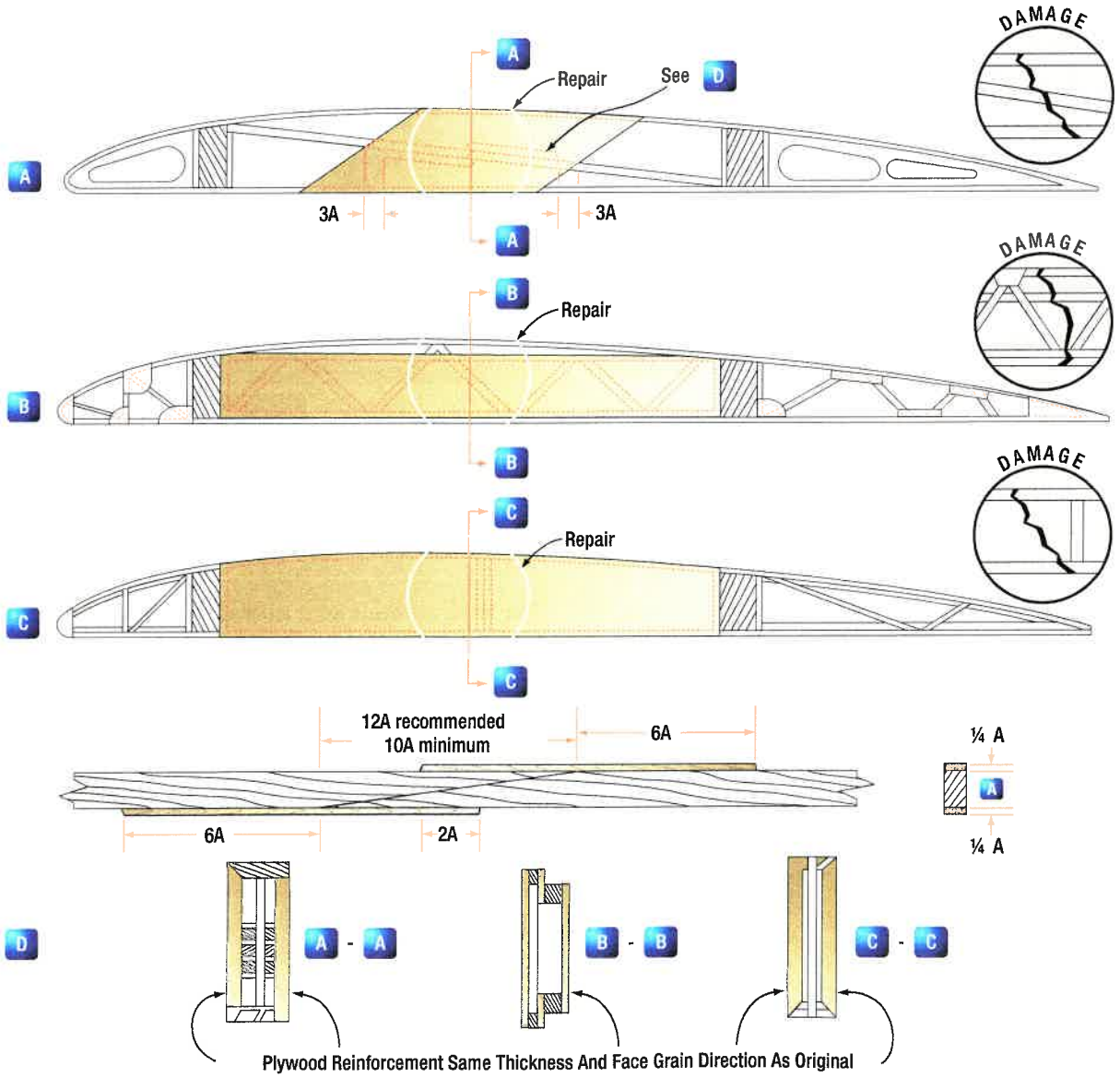


Figure 3-115. Typical compression rib repair.

other side. The method used in this repair is essentially the same as in ***Figure 3-115A***, except that the plywood reinforcement plate, shown in ***Figure 3-115B***, is continued the full distance between the spars.

Figure 3-115C illustrates a compression rib of the I type with a rectangular vertical member on each side of the web. The method of repair is essentially the same as in ***Figure 3-115A***, except the plywood reinforcement plates on each side, shown in ***Figure 3-115C***, are continued the full distance between the spars.

WING SPAR REPAIRS

Wood wing spars are fabricated in various designs using solid wood, plywood, or a combination of the two. (***Figure 3-116***)

When a spar is damaged, the method of repair must conform to the manufacturer's instructions and recommendations. If instructions are not available for a specific type of repair, it is highly recommended that you request appropriate engineering assistance to evaluate and provide guidance for the intended repair.

Shown in ***Figure 3-117*** is a recommended method to repair either a solid or laminated rectangle spar. The slope of the scarf in any stressed part, such as a spar, should not be steeper than 15 to 1.

Unless otherwise specified by the aircraft manufacturer, a damaged spar may be spliced at almost any point except at wing attachment fittings, landing gear fittings, engine mount fittings, or lift and interplane strut fittings. These fittings may not overlap any part of the splice. The reinforcement plates of the splice should not interfere with the proper attachment or alignment of the fittings. Taper reinforcement plates on the ends at a 5:1 slope (***Figure 3-118***).

The use of a scarf joint to repair a spar or any other component of an aircraft is dependent on the accessibility to the damaged section. It may not be possible to utilize a scarf repair where recommended, so the component may have to be replaced. A scarf must be precisely cut on both adjoining pieces to ensure an even thin glue line; otherwise, the joint may not achieve full strength. The primary difficulty encountered in making this type of joint is obtaining the same bevel on each piece. (***Figure 3-119***)

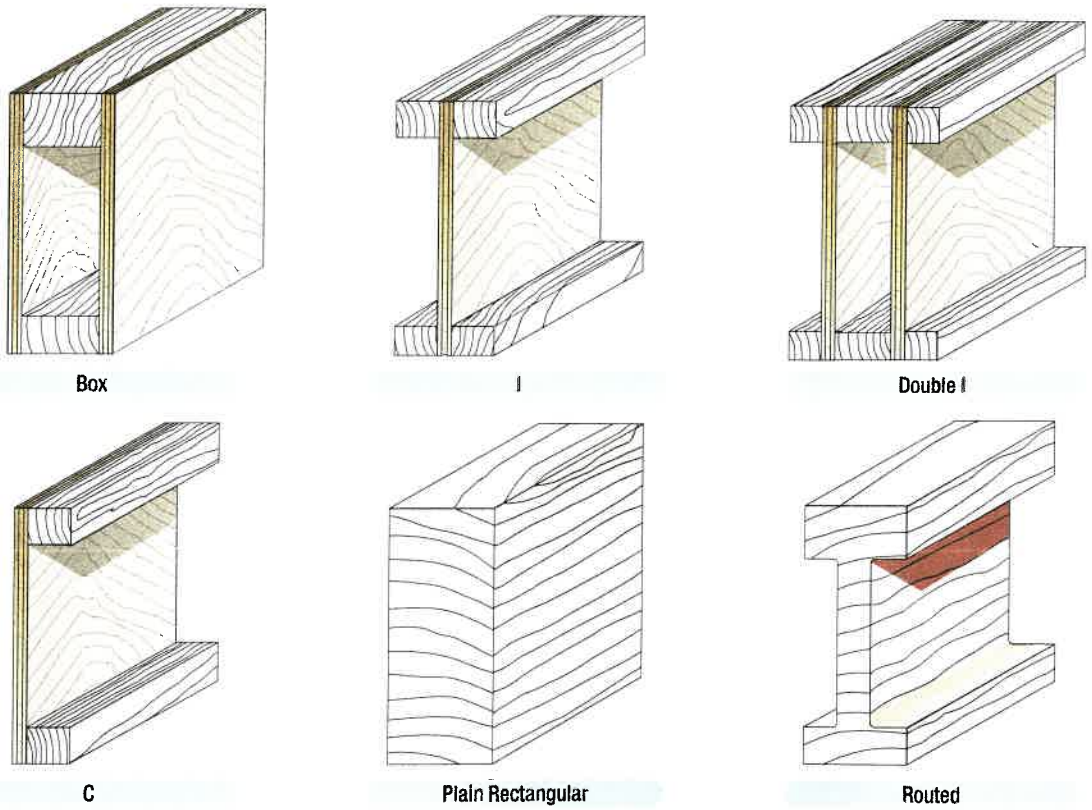


Figure 3-116. Different wood wing spar designs.

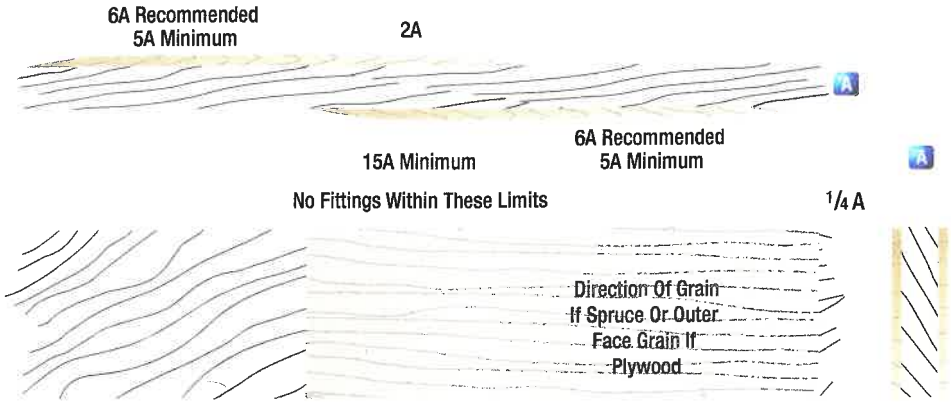


Figure 3-117. Typical splice repair of solid rectangular spar.

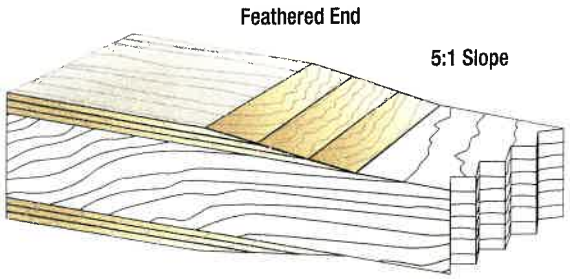


Figure 3-118. Tapered faceplate.

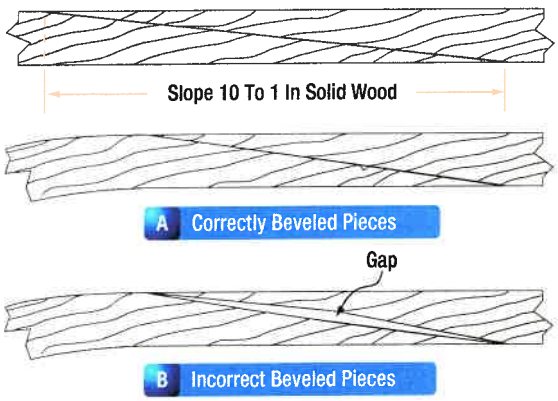


Figure 3-119. Beveled scarf joint.

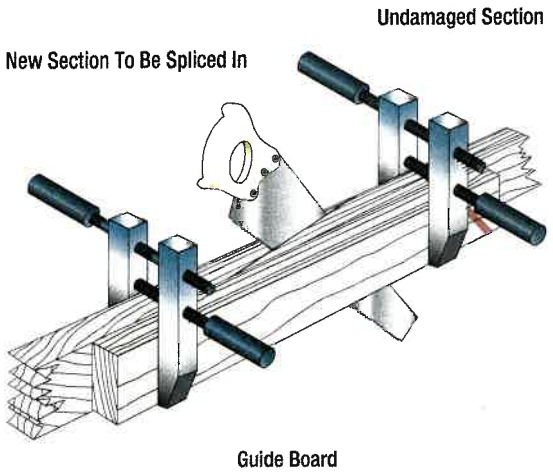


Figure 3-120. Making a scarf joint.

The mating surfaces of the scarf must be smooth. You can machine smooth a saw cut using any of a variety of tools, such as a plane, a joiner, or a router. For most joints, you need a beveled fixture set at the correct slope to complete the cut. ***Figure 3-120*** illustrates one method of producing an accurate scarf joint.

Once the two bevels are cut for the intended splice, clamp the pieces to a flat guide board of similar material. Then, work a sharp, fine tooth saw all the way through the joint. Remove the saw, decrease pressure, and tap one of the pieces on the end to close the gap. Work the saw again through the joint. Continue this procedure until the joint is perfectly parallel with matching surfaces. Then, make a light cut with the grain, using a sharp plane, to smooth both mating surfaces. Another method of cutting a scarf uses a simple scarf-cutting fixture that you can also fabricate for use with a router. Extend the work piece beyond the edge so the finished cut results in a feathered edge across the end of the scarf. (***Figure 3-121***)

There are numerous tools made by individuals, and there are commercial plans for sale with instructions for building scarf cutting tools. Most of them work, but some are better than others. The most important requirement for the tool is that it produces a smooth, repeatable cut at the appropriate angle.

Local damage to the top or bottom edge of a solid spar may be repaired by removing the damaged portion and fabricating a replacement filler block of the same material

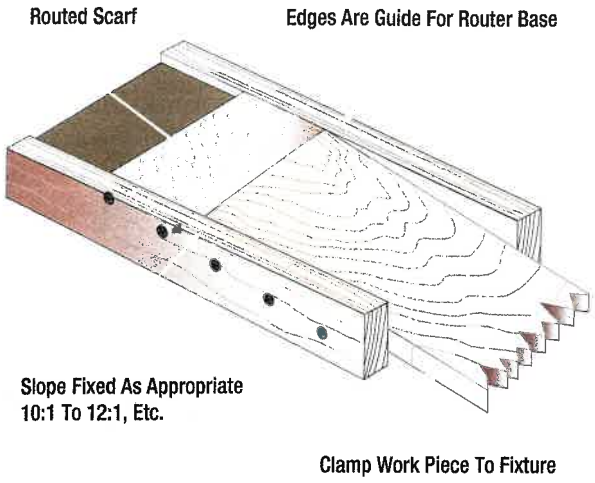


Figure 3-121. Scarf cutting fixture.

as the spar. Full width doublers are fabricated as shown and then all three pieces are glued and clamped to the spar. Nails or screws should not be used in spar repairs. A longitudinal crack in a solid spar may be repaired using doublers made from the proper thickness plywood. Care must be taken to ensure the doublers extend the minimum distance beyond the crack. (***Figure 3-122***)

A typical repair to a built up I spar is illustrated using plywood reinforcement plates with solid wood filler blocks. As with all repairs, the reinforcement plate ends should be feathered out to a 5:1 slope. (***Figure 3-123***)

Repair methods for the other types of spar illustrated at the start of this section all follow the basic steps of repair. The wood used should be of the same type and size as the original spar. Always splice and reinforce plywood webs with the same type of plywood as the original. Do not use solid wood to replace plywood webs because plywood is stronger in shear than solid wood of the same thickness. The splices and scarf cuts must be of the correct slope for the repair with the face grain running in the same direction as the original member. Not more than two splices should be made in any one spar.

When a satisfactory repair to a spar cannot be accomplished, the spar should be replaced. New spars may be obtained from the manufacturer or the holder of a Parts Manufacturer Approval (PMA) for that part. An owner produced spar may be installed provided it is made from a manufacturer-approved drawing. Care should be taken to ensure that any replacement spars accurately match the manufacturer's original design.

BOLT AND BUSHING HOLES

All bolts and bushings used in aircraft structures must fit snugly into the holes. If the bolt or bushing is loose, movement of the structure allows it to enlarge the hole. In the case of elongated bolt holes in a spar or cracks in close proximity to the bolt holes, the repair may require a new section to be spliced in the spar, or replacement of the entire spar.

All holes drilled in a wood structure to receive bolts or bushings should be of such size that inserting the bolt or bushing requires a light tapping with a wood or rawhide mallet. If the hole is so tight that heavy blows are necessary, deformation of the wood may cause splitting or unequal load distribution.

For boring accurate smooth holes, it is recommended that a drill press be utilized where possible. Holes should be drilled with sharp bits using slow steady pressure.

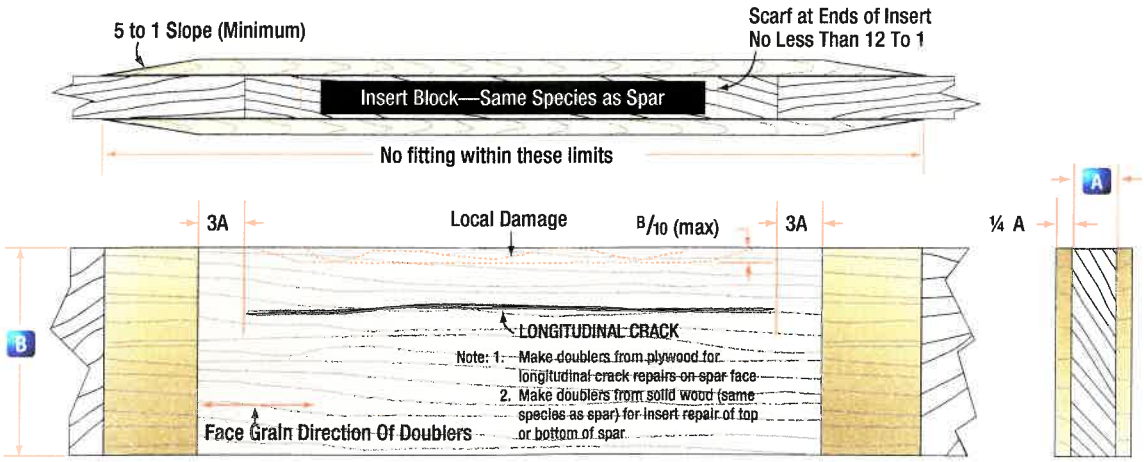


Figure 3-122. A method to repair damage to solid spar.

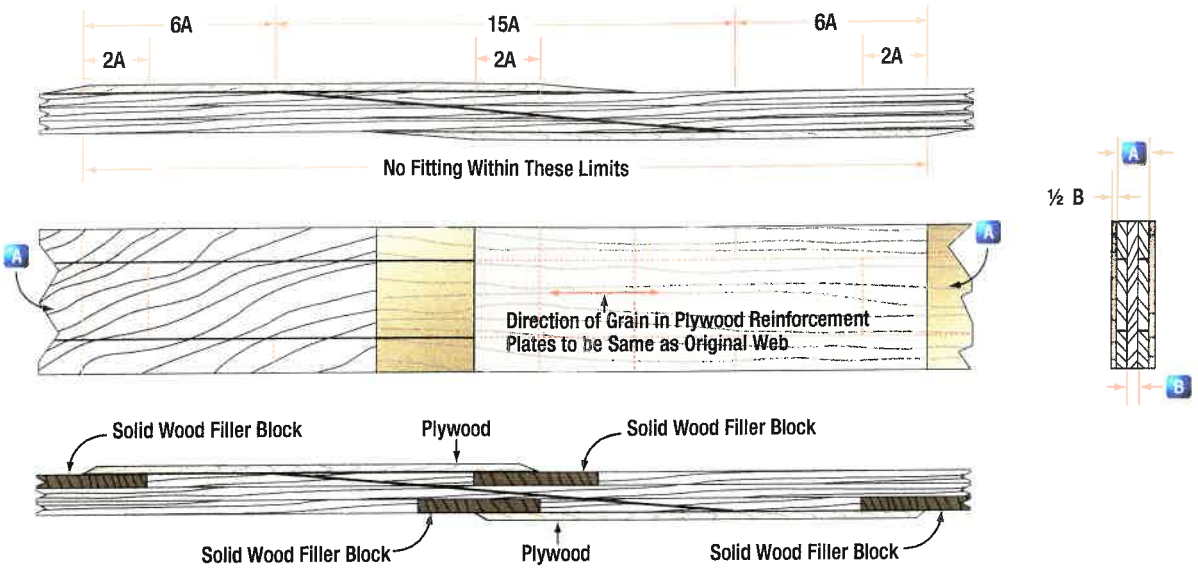


Figure 3-123. Repairs to a built-up I spar.

Standard twist drills can be used in wood when sharpened to a 60° angle. However, a better designed drill was developed for wood boring called a lip and spur or brad point. The center of the drill has a spur with a sharp point and four sharp corners to center and cut rather than walk as a conventional drill sometimes does. It has the outside corner of the cutting edges leading, so that it cuts the periphery of the hole first and maximizes the chance that the wood fibers cut cleanly, leaving a smooth bore.

Forstner bits bore precise, flat bottomed holes in wood, in any orientation with respect to the wood grain. They must be used in a drill press because more force is needed for their cutting action. Also, they are not designed to clear chips from the hole and must be pulled out periodically to do this. A straight, accurate bore-through hole can be completed by drilling through the work piece and into a piece of wood backing the work piece.

All holes bored for bolts that are to hold fittings in place should match the hole diameter in the fitting. Bushings made of steel, aluminum, or plastic are sometimes used to prevent crushing the wood when bolts are tightened. Holes drilled in the wood structure should be sealed after being drilled. This can be accomplished by application of varnish or other acceptable sealer into the open hole. The sealer must be allowed to dry or cure thoroughly prior to the bolts or bushings being installed.

PLYWOOD SKIN REPAIRS

Plywood skin can be repaired using a number of different methods depending on the size of the hole and its location on the aircraft. Manufacturer's instructions, when available, should be the first source of a repair scheme.

Fabric Patch

A fabric patch is the simplest method to repair a small hole in plywood. This repair is used on holes not exceeding 1 inch in diameter after being trimmed to a smooth outline. The edges of the trimmed hole should first be sealed, preferably with a two part epoxy varnish. This varnish requires a long cure time, but it provides the best seal on bare wood.

The fabric used for the patch should be of an approved material using the cement recommended by the manufacturer of the fabric system. The fabric patch should be cut with pinking shears and overlap the plywood skin by at least 1 inch. A fabric patch should not be used to repair holes in the leading edge of a wing, in the frontal area of the fuselage, or nearer than 1 inch to any frame member.

Splayed Patch

A splayed patch is a flush patch. The term splayed denotes that the edges of the patch are tapered, with the slope cut at a 5:1 ratio to the thickness of the skin. This may be used for small holes where the largest dimension of the hole to be repaired is not more than 15 times the skin thickness and the skin is not more than 1/10 inch thick. This calculates to nothing larger than a 1½ inch

trimmed hole in very thin plywood. Using the sample 1/10 inch thick plywood and a maximum trimmed hole size of 1½ inches, and cutting a 5:1 scarf, results in a 2½ inches round section to be patched. The patch should be fabricated with a 5:1 scarf, from the same type and thickness plywood as the surface being repaired.

Glue is applied to the beveled edges and the patch is set with the grain parallel to the surface being repaired. A pressure plate of thicker plywood cut to the exact size of the patch is centered over the patch covered with waxed paper. A suitable weight is used for pressure until the glue has set. The repair is then sanded and finished to match the original surface. (***Figure 3-124***)

Surface Patch

Plywood skins not over ⅛ inch thick that are damaged between or along framing members may be repaired with a surface or overlay patch. Surface patches located aft of the 10 percent chord line, or which wrap around the leading edge and terminate aft of the 10 percent chord line, are permissible. You can use surface patches to patch trimmed holes up to a 50 inch perimeter, and may cover an area as large as one frame or rib space.

Trim the damaged area to a rectangle or triangular shape with rounded corners. The radius of the corners must be at least 5 times the skin thickness. Doublers made of plywood at least ¼ inch thick are reinforcements placed under the edge of the hole inside the skin. Nail and glue the doublers in place. Extend the doublers from one framing member to another and strengthen at the ends by saddle gussets attached to the framing members. (***Figure 3-125***)

The surface patch is sized to extend beyond the cutout as indicated. All edges of the patch are beveled, but the leading edge of the patch should be beveled at an angle at least 4:1 of the skin thickness. The face-grain direction of the patch must be in the same direction of the original skin. Where possible, weights are used to apply pressure to a surface patch until the glue has dried. If the location of the patch precludes the use of weight, small round head wood screws can be used to apply glue pressure to secure the patch. After a surface patch has dried, the screws can be removed and the holes filled. The patch should be covered with fabric that overlaps the original surface by at least 2 inches. The fabric should be from one of the approved fabric covering systems using the procedures recommended by the manufacturer to cement and finish the fabric.

Plug Patch

Two types of plug patch, oval and round, may be used on plywood skins. Because the plug patch is only a skin repair, use it only for damage that does not involve the supporting structure under the skin. Cut the edges of a plug patch at right angles to the surface of the skin. Cut the skin also to a clean round or oval hole with edges at right angles to the surface. Cut the patch to the exact

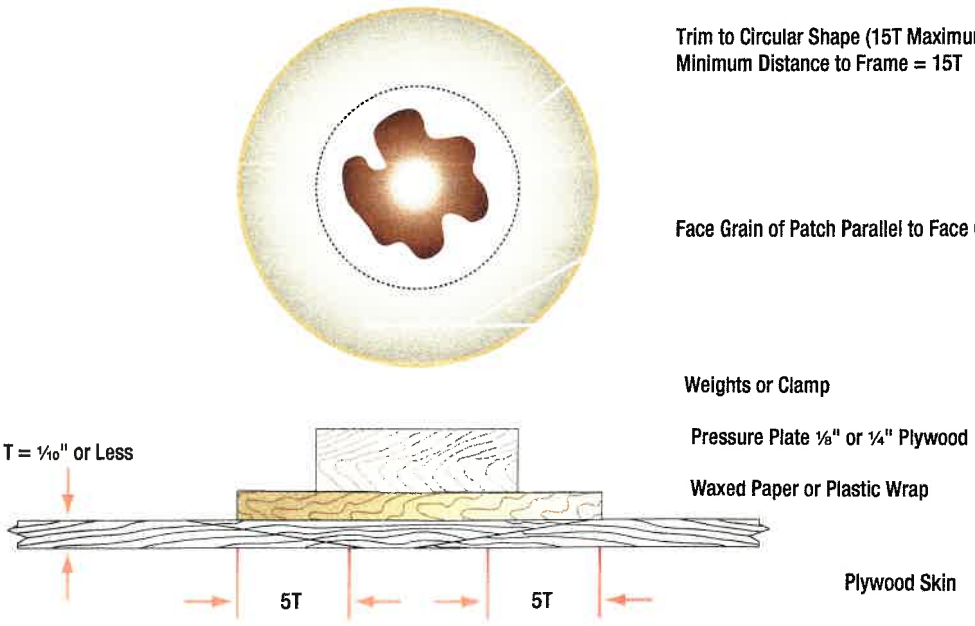


Figure 3-124. Splayed patch.

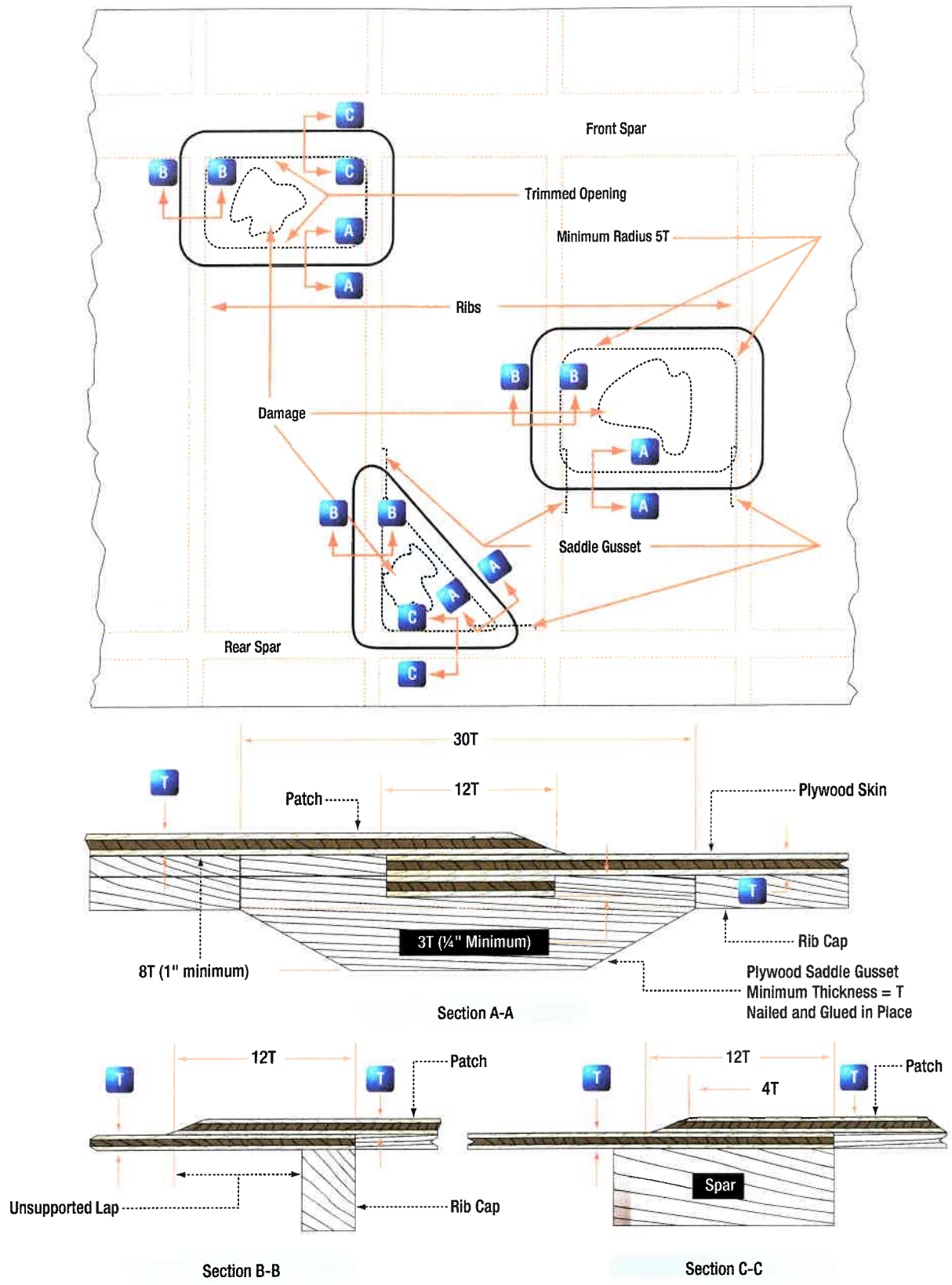
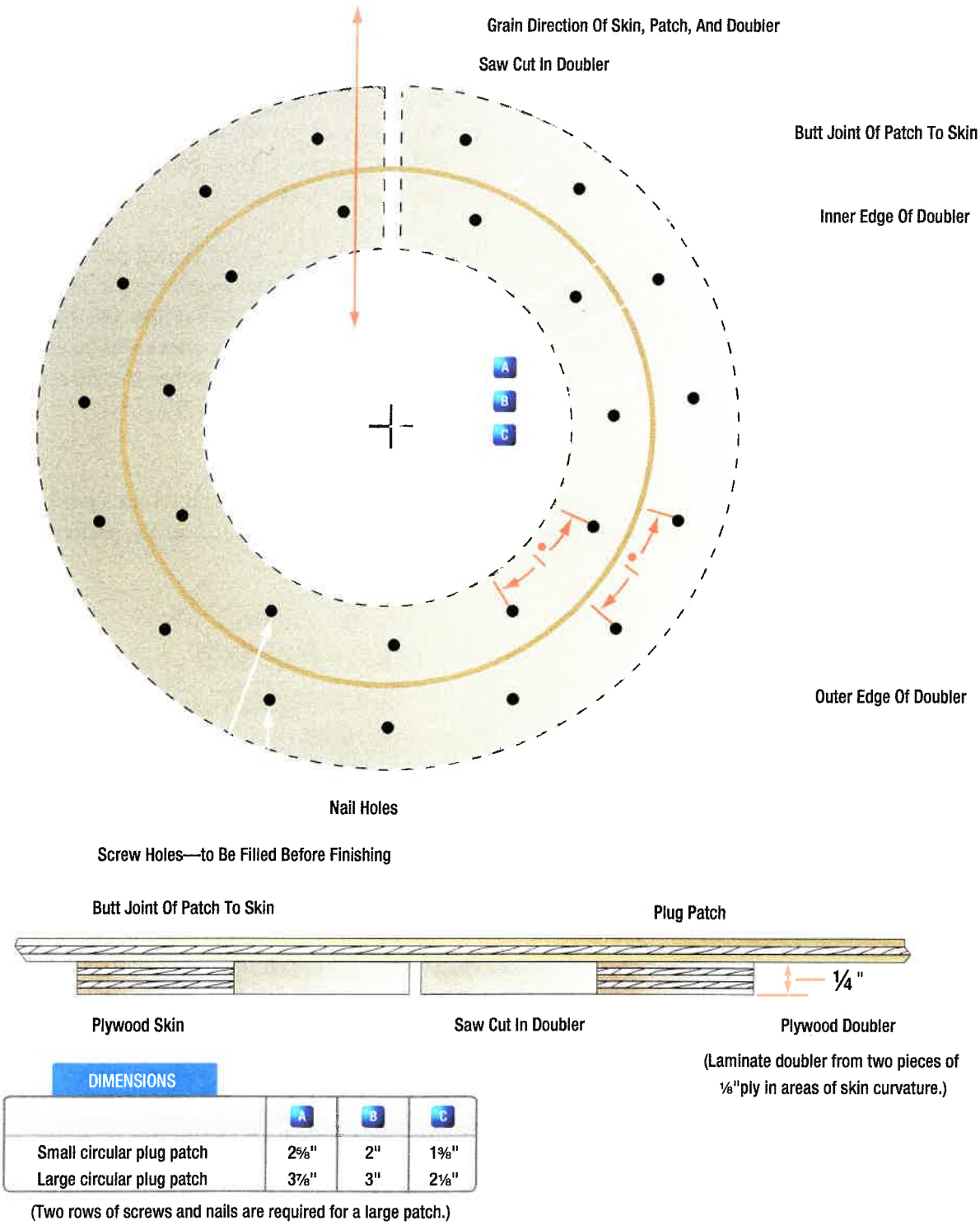


Figure 3-125. Surfaces patches.

size of the hole; when installed, the edge of the patch forms a butt joint with the edge of the hole. You can use a round plug patch where the cutout repair is no larger than 6 inches in diameter. Sample dimensions for holes of 4 inches and 6 inches in diameter appear in ***Figure 3-126***.

The following steps provide a method for making a round plug patch:

1. Cut a round patch large enough to cover the intended repair. If applicable for size, use the sample dimensions in ***Figure 3-126***. The patch must be of the same material and thickness as the original skin.



DIMENSIONS

	A	B	C
Small circular plug patch	2⅝"	2"	1⅜"
Large circular plug patch	3⅞"	3"	2⅛"

(Two rows of screws and nails are required for a large patch.)

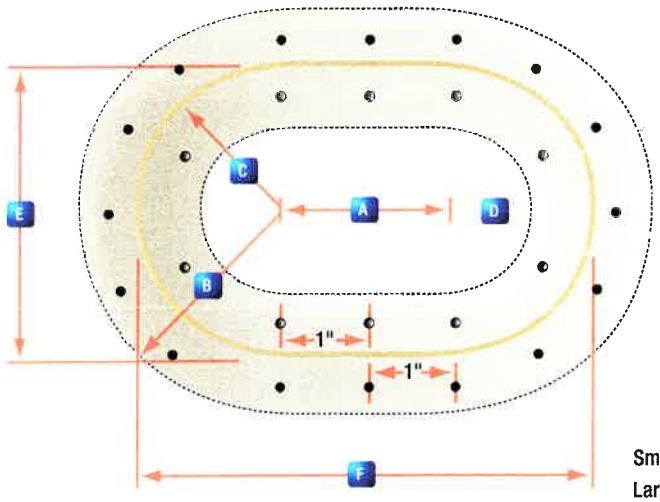
Figure 3-126. Round plug patch assembly.

2. Place the patch over the damaged spot and mark a circle of the same size as the patch.
3. Cut the skin inside the marked circle so that the plug patch fits snugly into the hole around the entire perimeter.
4. Cut a doubler of soft quarter inch plywood, such as poplar. A small patch is cut so that its outside radius is ⅝ inch greater than the hole to be patched and the inside radius is ⅝ inch less. For a large patch the dimensions would be increased to ⅞ inch each. If the curvature of the skin surface is greater than a rise of ⅛ inch in 6 inches, the doubler should be preformed to the curvature using hot water or steam. As an alternative, the doubler may be laminated from two pieces of ⅛ inch plywood.
5. Cut the doubler through one side so that it can be inserted through the hole to the back of the skin. Place the patch plug centered on the doubler and mark around its perimeter. Apply a coat of glue outside the line to the outer half of the doubler surface that will bear against the inner surface of the skin.
6. Install the doubler by slipping it through the cutout hole and place it so that the mark is concentric with the hole. Nail it in place with nailing strips, while holding a bucking bar or similar object under the doubler for backup. Place waxed paper between the nailing strips and the skin. Cloth webbing under the nailing strips facilitates removal of the strips and nails after the glue dries.
7. After the glue has set for the installed doubler, and you have removed the nail strips, apply glue to the inner half of the doubler and to the patch plug. Drill holes around the plug's circumference to accept No. 4 round head wood screws. Insert the plug with the grain aligned to the surface wood.
8. Apply the pressure to the patch by means of the wood screws. No other pressure is necessary.
9. After the glue has set, remove the screws and fill the nail and screw holes. Sand and finish to match the original surface.

The steps for making an oval plug patch are identical to those for making the round patch. The maximum dimensions for large oval patches are 7 inches long and 5 inches wide. Oval patches must be cut, so when installed, the face grain matches the direction of the original surface. (***Figure 3-127***)

Scarf Patch

A properly prepared and installed scarf patch is the best repair for damaged plywood and is preferred for most



Outer Edge of Doubler

Butt Joint of Patch to Skin

Inner Edge of Doubler

Nail Holes

Screw Holes—to be Filled Before Finishing

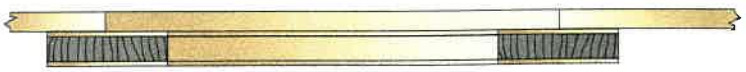
PATCH DIMENSIONS

	A	B	C	D	E	F
Small	1⅛"	2⅜"	1½"	⅞"	3"	4½"
Large	2"	3⅜"	2½"	1⅜"	5"	7"

(Two rows of screws and nails required for large patch.)

Butt Joint of Patch to Skin

Plug Patch (Grain Parallel To Skin)



Plywood Skin

Plywood Doubler (Grain Parallel to Skin)

Figure 3-127. An oval plug patch.

skin repairs. The scarf patch has edges beveled at a 12:1 slope; the splayed patch is beveled at a 5:1 slope. The scarf patch also uses reinforcements under the patch at the glue joints.

Much of the outside surface of a plywood aircraft is curved. If the damaged plywood skin has a radius of curvature not greater than 100 times the skin thickness, you can install a scarf patch. However, it may be necessary to soak or steam the patch, to preform it prior to gluing it in place. Shape backing blocks or other reinforcements to fit the skin curvature. You can make scarf cuts in plywood with various tools, such as a hand plane, spoke shave, a sharp scraper, or sanding block. Sawn or roughly filed surfaces are not recommended because they are normally inaccurate and do not form the best glue joint.

The Back Of The Skin Is Accessible For Repair

When the back of a damaged plywood skin is accessible, such as a fuselage skin, repair it with scarf patches cut and installed with the grain parallel to the surface skin. Details for this type of repair are shown in ***Figure 3-128***.

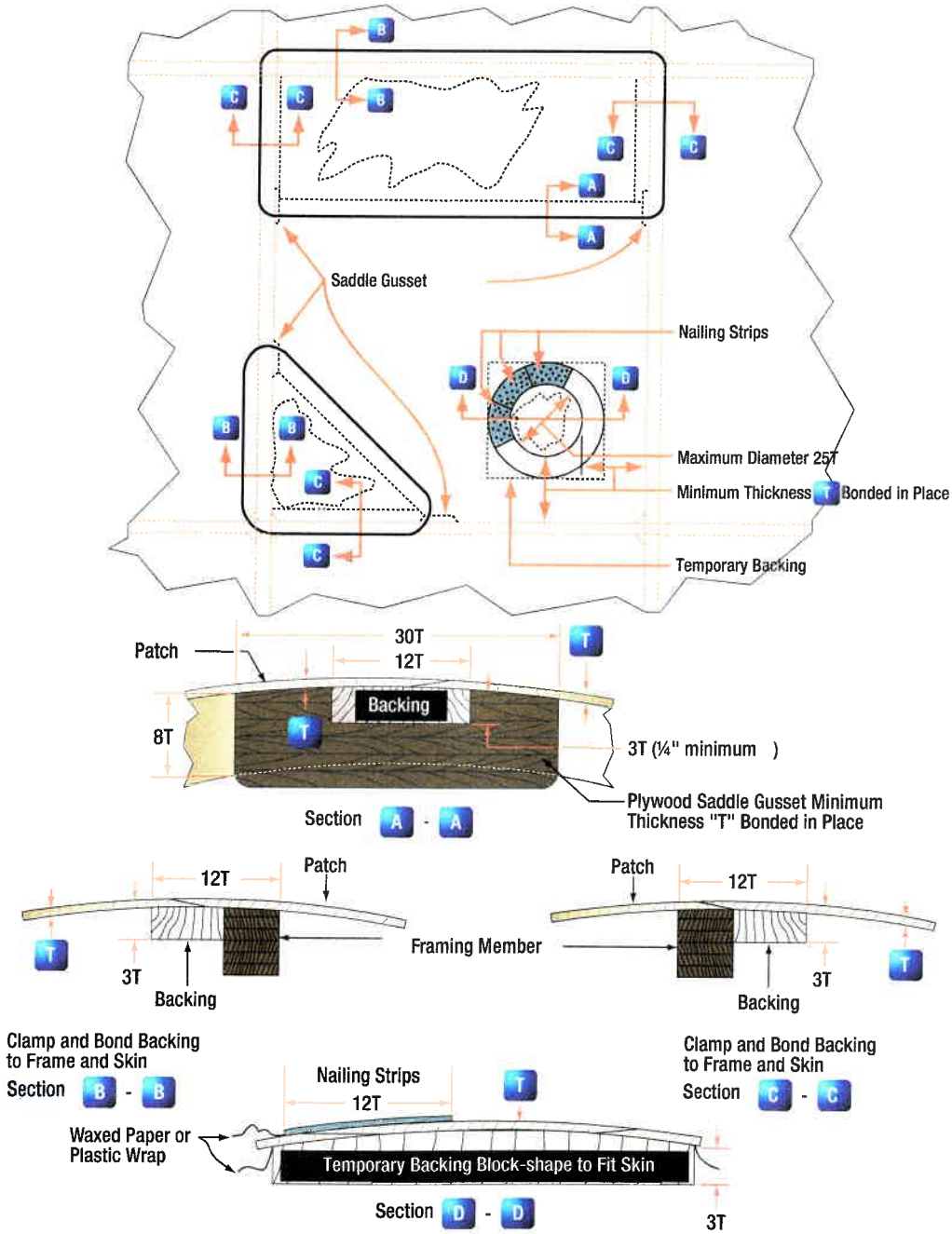


Figure 3-128. Round plug patch assembly.

Figure 3-128, Section A-A, shows methods of support for a scarf between frame members using permanent backing and gussets. When the damage follows or extends to a framing member, support the scarf as shown in section B-B. When the scarf does not quite extend to a frame member, support the patch as shown in section C-C. Damage that does not exceed 25 times the skin thickness (3⅛ inches for ⅛ inch thick skin) after being trimmed to a circular shape can be repaired as shown in section D-D, provided the trimmed opening is not nearer than 15 times the skin thickness to a frame member (1⅞ inches for ⅛ inch thick skin).

A temporary backing block is carefully shaped from solid wood and fitted to the inside surface of the skin. A piece of waxed paper or plastic wrap is placed between the block and the underside of the skin. The scarf patch is installed and temporarily attached to the backing block, being held together in place with nailing strips. When the glue sets, remove the nails and block, leaving a flush surface on both sides of the repaired skin.

The Back Of The Skin Is Not Accessible For Repair

To repair a section of the skin with a scarf patch when access to the back side is not possible, use the following steps to facilitate a repair, as shown in ***Figure 3-129***.

Cut out and remove the damaged section. Carefully mark and cut the scarf around the perimeter of the hole. Working through the cutout, install backing strips along all edges that are not fully backed by a rib or spar. To prevent warping of the skin, fabricate backing strips from soft textured plywood, such as yellow poplar or spruce, rather than a piece of solid wood.

Use nailing strips to hold backing strips in place while the glue sets. Use a bucking bar, where necessary, to provide support for nailing. A saddle gusset of plywood should support the end of the backing strip at all junctions between the backing strips and ribs or spars. If needed, nail and bond the new gusset plate to the rib or spar. It may be necessary to remove and replace an old gusset plate with a new saddle gusset, or nail a new gusset over the original.

Unlike some of the other type patches that are glued and installed as one process, this repair must wait for the glue to set on the backing strips and gussets. At that point, the scarf patch can be cut and fit to match the grain, and glued, using weight for pressure on the patch as appropriate. When dry, fill and finish the repair to match the original surface.

FINISHING WOOD STRUCTURES

Any repair to spars, ribs, skin surfaces, or other structural parts of the airframe involves finishing as the final step of the job. The surface finish is the final line of defense to prevent the destructive effects of moisture entry into the structure. The time and effort spent during the preparatory phase of the refinishing process will be reflected in the appearance and longevity of the finished surface. Adherence to the instructions issued by the finish manufacturer is necessary to obtain the appearance desired and protective characteristics for the product used.

The primary objective of interior finishes is to afford protection of the wood against serious change in moisture content when exposed to damp air or to water that gains entrance to closed spaces by condensation or by penetration of rain, mist, or fog. Coatings, on contact areas between wood and metal protect the metal against corrosion from moisture in the wood. The primary objectives of the exterior finish are to protect the wood against weathering, provide a suitable appearance, and present a smooth surface in flight.

Acceptable Finishes

In the U.S., any varnish conforming to Federal Specification TT-V-109, as amended, or other coating approved by the airframe manufacturer or the FAA is acceptable. Exterior surfaces must be further protected from the effects of abrasion, weather, and sunlight. A number of systems for exterior finishing have STC approval and are manufactured under a Parts Manufacturer Approval (PMA). Low viscosity epoxy adhesive (meeting the requirements of paragraph 1-4 for that purpose) may be used as an internal surface finish when subsequent bonding is necessary.

Precautions

When making repairs, avoid excessive contamination of surfaces with adhesive squeeze out at joints and on all surfaces. Excess adhesive should always be removed before applying finish. Because many paints and adhesives are incompatible, even a slight amount of adhesive underneath the finish may cause premature deterioration of the finish.

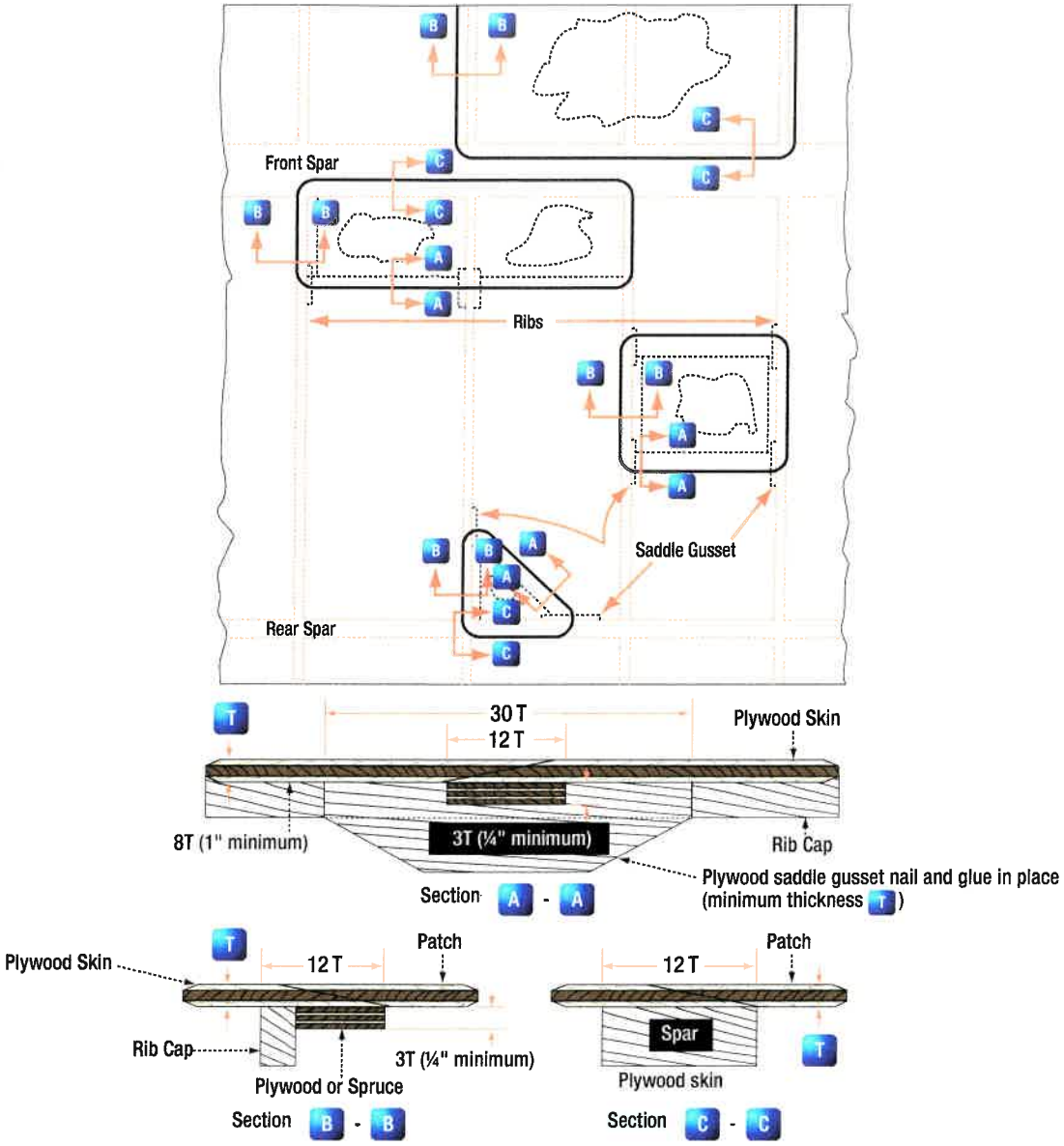


Figure 3-129. Round plug patch assembly.

Soiling substances, such as oil and grease, should be removed as completely as possible. Naphtha may be used to remove surface deposits of oil and grease; however, thinned residue may penetrate into any unprotected wood. In areas where minor amounts of oil or grease have penetrated the wood surface, removal may be accomplished by use of an absorbent type of cleaner such as gunsmith's whiting or a clothing spot lifter. Marks that are made by grease pencils or lumber crayons containing wax are harmful and should be removed, but marks made by ordinary soft graphite pencils and nonoily stamp pad inks may be safely finished over. All dust, dirt, and other solid particles should be removed.

Sawdust, shavings, and chips must be removed from enclosed spaces before they are sealed off by replacement of skin. A vacuum cleaner is useful for such cleaning.

Since most adhesives will not bond satisfactorily to sealers, it is necessary to avoid applying sealer over the areas where adhesive will be applied. Mark off areas to receive adhesive, and allow an additional ¼ inch on each side of the adhesive area to provide for misalignment when mating the parts. It is preferable to leave some unsealed areas rather than risk weakening the joint by accidental overlap of the sealer into the bonded areas. Wherever possible, apply sealer to the margins after the adhesive has cured. As an example, the lower skin of a wing bay would be installed first, leaving access from

above to apply sealer. All low spots (where moisture would collect) are well sealed. The top skin would be installed last, so that the only unsealed margins would be on upper surfaces where moisture is least likely to collect.

An alternative to the previous paragraph is to use an approved epoxy coating and compatible epoxy adhesive. Apply the coating, allowing ¼ inch margins as in the previous paragraph. After the coating has cured, apply epoxy adhesive to joint surfaces, and overlap the adhesive onto the sealer. Close joint and clamp. The epoxy adhesive will bond satisfactorily to the coating and ensure a complete coverage of the wood surfaces. Use only approved and compatible adhesives and coatings for this method.

Finishing Interior Surfaces

Finish repaired ribs, spars, interior of plywood skin, and other internal members, including areas of contact between metal and wood, by applying one thinned coat (for penetration into wood grain) of varnish or other acceptable finish, followed by two full coats. Protect built up box spars and similar closed structures on the interior in the same way. Where better protection is required, as on the surfaces of wheel wells and the bottoms of hulls below the floor boards, an additional coat of aluminized sealer consisting of 12 to 16 ounces of aluminum paste per gallon of sealer, may be applied.

Finishing Exterior Surfaces

Exterior surfaces require more protection than interior areas due to the effects of abrasion, weather, and sunlight. Tests have shown that the interior temperature of wooden aircraft structures can reach 85°C when the aircraft is finished in a dark color and parked outdoors on a hot, still day. Exposure to prolonged high temperature is detrimental to wood, adhesives, and finishes. Wood loses approximately 25 percent of its strength at 52°C. For this reason, the mechanic should consider temperature effects when selecting finish colors or looking for areas of likely deterioration. The lowest temperatures are found when the aircraft is finished in white or very light colors, while darker colors produce higher temperatures. A general trend toward higher temperatures may be seen when exterior colors are yellow, pink, light blue, aluminum, purple, blue, light green, orange, tan, red, green, brown, and black. A lighter shade of a particular color helps to reduce temperatures.

Finishing of End Grain Surfaces

End grain portions of wooden members are much more absorbent than side grain. Because of this extreme vulnerability to moisture entry, it is necessary to take extra precautions to seal end grain.

Apply at least one thinned coat of acceptable sealer to ensure maximum penetration, and then follow with as many full strength coats as necessary to achieve a smooth, glossy coating. Depending on the type wood to be finished, two to four full coats will be required. A final coat of aluminized varnish may be applied to end grain surfaces. If the surfaces are to be finished with dope or lacquer, a dope proof sealer, similar to Federal Specification TT-V-109, or epoxy sealer should be used. Exposed end grain includes such surfaces as spar butts, skin edges, areas around vent holes, inspection holes, fittings, and exposed scarfed or tapered surfaces.

Finishing With Fabric or Tape

A number of systems for exterior finishing have STC approval and are manufactured under a Parts Manufacturer Approval (PMA). Follow the product manufacturer's instructions for the system used. If the finish surrounding the repair is a traditional dope system, seal the wood grain with a suitable solvent resistant one part varnish, commonly described as "dope proof," or a two part epoxy varnish. Follow with two coats of clear dope, and allow sufficient drying time between coats. Apply a third coat of clear dope and lay a piece of pinked edge airplane cloth into the wet film. All air bubbles should be worked out by brushing to ensure maximum adhesion. When dry, apply one brush coat, to ensure proper penetration, and at least one spray coat of clear dope. The dried spray coat may be sanded with fine sandpaper to obtain a smoother finish. Complete the refinishing of the surface by application of a topcoat as required to match the adjacent area.

Sealing of Bolt Holes

Bolt holes in wooden structure provide a vulnerable entry point for moisture. Variations in moisture content around bolt holes can lead to decay or splitting. In addition, excessive moisture at bolt holes promotes corrosion of the bolts. Sealing of the wood surfaces in bolt holes can be accomplished by application of varnish or other acceptable sealer into the open hole. The sealer must be allowed to dry or cure thoroughly prior to bolt installation.

6.3.3 - AIRCRAFT FABRIC COVERINGS

GENERAL HISTORY

Fabric covered aircraft play an important role in the history of aviation. The famous Wright Flyer utilized a fabric covered wood frame in its design, and fabric covering continued to be used by many aircraft designers and builders during the early decades of production aircraft. The use of fabric covering on an aircraft offers one primary advantage: light weight. In contrast, fabric coverings have two disadvantages: flammability and lack of durability.

Finely woven organic fabrics, such as Irish linen and cotton, were the original fabrics used for covering airframes, but their tendency to sag left the aircraft structure exposed to the elements. To counter this problem, builders began coating the fabrics with oils and varnishes. In 1916, a mixture of cellulose dissolved in nitric acid, called nitrate dope, came into use as an aircraft fabric coating. Nitrate dope protected the fabric, adhered to it well, and tautened it over the airframe. It also gave the fabric a smooth, durable finish when dried. The major drawback to nitrate dope was its extreme flammability.

To address the flammability issue, aircraft designers tried a preparation of cellulose dissolved in butyric acid called butyrate dope. This mixture protected the fabric from dirt and moisture, but it did not adhere as well to the fabric as nitrate dope. Eventually, a system combining the two dope coatings was developed. First, the fabric was coated with nitrate dope for its adhesion and protective qualities. Then, subsequent coats of butyrate dope were added. Since the butyrate dope coatings reduced the overall flammability of the fabric covering, this system became the standard fabric treatment system.

The second problem, lack of durability, stems from the eventual deterioration of fabric from exposure to the elements that results in a limited service life. Although the mixture of nitrate dope and butyrate dope kept out dirt and water, solving some of the degradation issue, it did not address deterioration caused by Ultraviolet (UV) radiation from the sun. Ultraviolet radiation passed through the dope and degraded not only the fabric, but also the aircraft structure underneath. Attempts to paint the coated fabric proved unsuccessful, because paint does not adhere well to nitrate dope. Eventually, aluminum solids were added to the butyrate coatings. This mixture reflected the sun's rays, prevented harmful UV rays from penetrating the dope, and protected the fabric, as well as the aircraft structure.

Regardless of treatments, organic fabrics have a limited lifespan; cotton or linen covering on an actively flown aircraft lasts only about 5–10 years. Furthermore, aircraft cotton has not been available for over 25 years. As the aviation industry developed more powerful engines and more aerodynamic aircraft structures, aluminum became the material of choice. Its use in engines, aircraft frames, and coverings revolutionized aviation. As a covering, aluminum protected the aircraft structure from the elements, was durable, and was not flammable.

Although aluminum and composite aircraft dominate modern aviation, advances in fabric coverings continue to be made because gliders, home built, and light sport aircraft, as well as some standard and utility certificated aircraft, are still produced with fabric coverings. (***Figure 3-130***) The nitrate/butyrate dope process works well, but does not mitigate the short lifespan of organic fabrics. It was not until the introduction of polyester fabric as an aircraft covering in the 1950's that



Figure 3-130. Examples of aircraft produced using fabric skin.

the problem of the limited lifespan of fabric covering was solved. The transition to polyester fabric had some problems because the nitrate and butyrate dope coating process is not as suitable for polyester as it is for organic fabrics. Upon initial application of the dopes to polyester, good adhesion and protection occurred; as the dopes dried, they would eventually separate from the fabric. In other words, the fabric outlasted the coating.

Eventually, dope additives were developed that minimized the separation problem. For example, plasticizers keep the dried dope flexible and non-tautening dope formulas eliminate separation of the coatings from the fabric. Properly protected and coated, polyester lasts indefinitely and is stronger than cotton or linen. Today, polyester fabric coverings are the standard and use of cotton and linen on United States certificated aircraft has ceased. In fact, the long staple cotton from which grade-A cotton aircraft fabric is made is no longer produced in this country.

Recovering existing fabric aircraft is an accepted maintenance procedure. Not all aircraft covering systems include the use of dope coating processes. Modern aircraft covering systems that include the use of non-dope fabric treatments show no signs of deterioration even after decades of service. In this chapter, various fabrics and treatment systems are discussed, as well as basic covering techniques.

CHARACTERISTICS, PROPERTIES, AND TYPES OF FABRIC USED IN AIRPLANES

FABRIC TERMS

To facilitate the discussion of fabric coverings for aircraft, the following definitions are presented. ***Figure 3-131*** illustrates some of these items.

- Warp—the direction along the length of fabric.
- Fill or weave—the direction across the width of the fabric.
- Count—the number of threads per inch in warp or filling.
- Ply—the number of yarns making up a thread.
- Bias—a cut, fold, or seam made diagonally to the warp or fill threads.
- Pinked edge—an edge which has been cut by machine or special pinking shears in a continuous series of Vs to prevent raveling.

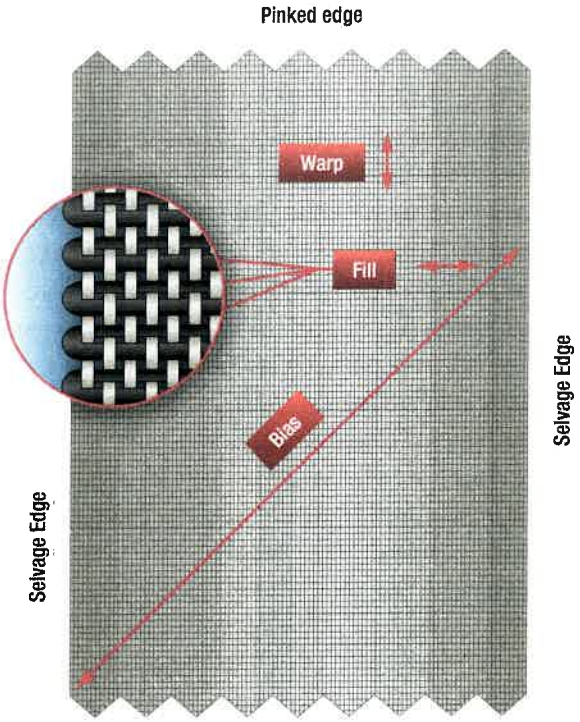


Figure 3-131. Aircraft fabric nomenclature.

- Selvage edge—the edge of cloth, tape, or webbing woven to prevent raveling.
- Greige—condition of polyester fabric upon completion of the production process before being heat shrunk.
- Cross coat—brushing or spraying where the second coat is applied 90° to the direction the first coat was applied. The two coats together make a single cross coat. (***Figure 3-132***)

APPROVED MATERIALS

There are a variety of approved materials used in aircraft fabric covering and repair processes. In order for the items to legally be used, the FAA must approve the fabric, tapes, threads, cords, glues, dopes, sealants, coatings, thinners, additives, fungicides, rejuvenators, and paints for the manufacturer, the holder of an STC, or a field approval.

Fabric

A Technical Standard Order (TSO) is a minimum performance standard issued by the FAA for specified materials, parts, processes, and appliances used on civil aircraft. For example, TSO-15d, Aircraft Fabric, Grade A, prescribes the minimum performance standards that approved aircraft fabric must meet. Fabric that meets or exceeds the TSO can be used as a covering. Fabric approved to replace Grade-A cotton, such as polyester,

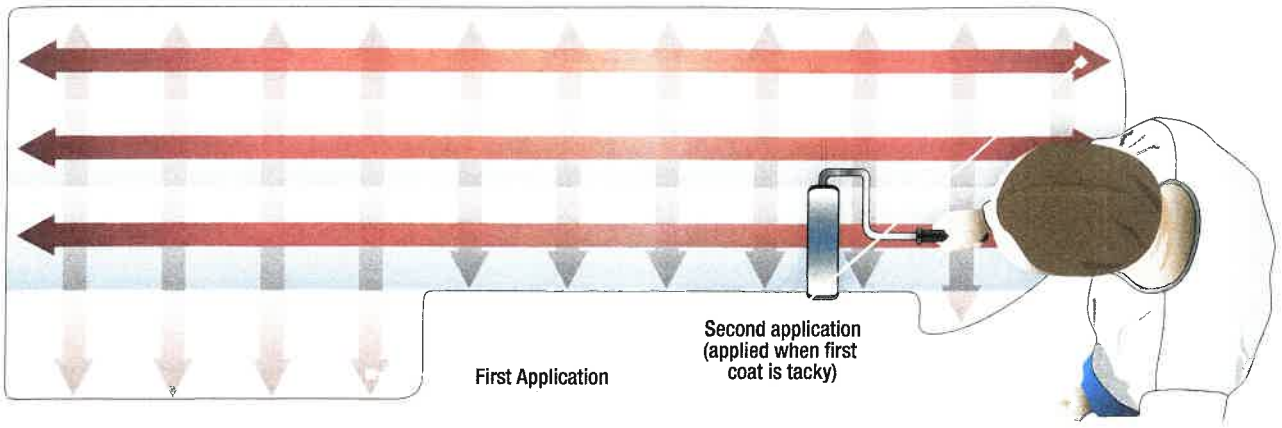


Figure 3-132. A single cross coat is made up of two coats of paint applied 90° to each other.

must meet the same criteria. TSO-15d also refers to another document, Society of Automotive Engineers (SAE) Aerospace Material Specification (AMS) 3806D, which details properties a fabric must contain to be an approved fabric for airplane cloth. Lighter weight fabrics typically adhere to the specifications in TSO-C14b, which refers to SAE AMS 3804C.

When a company is approved to manufacture or sell an approved aviation fabric, it applies for and receives a Parts Manufacturer Approval (PMA). Currently, only a few approved fabrics are used for aircraft coverings, such as the polyester fabrics Ceconite™, Stits/Poly-Fiber™, and Superflite™. These fabrics and some of their characteristics are shown in ***Figure 3-133.***

The holders of the Parts Manufacturer Approval (PMA) for these fabrics have also developed and gained approval for the various tapes, chords, threads, and liquids that are used in the covering process. These approved materials, along with the procedures for using them, constitute the STCs for each particular fabric covering process. Only the approved materials can be used. Substitution of other materials is forbidden and results in the aircraft being unairworthy.

Anti-Chafe Tape

Anti-chafe tape is used on sharp protrusions, rib caps, metal seams, and other areas to provide a smoother surface to keep the fabric from being torn. It is usually self-adhesive cloth tape and is applied after the aircraft is cleaned, inspected, and primed, but before the fabric is installed.

Aircraft Fabrics

Fabric Name or Type	Weight (oz/sq yd)	Count (warp x fill)	New Breaking Strength (lb) (warp, fill)	Minimum Deteriorated Breaking Strength	TSO
Ceconite™ 101	3.5	69 x 63	125,116	70% of original specified fabric	C-15d
Ceconite™ 102	3.16	60 x 60	106,113	70% of original specified fabric	C-15d
Polyfiber™ Heavy Duty-3	3.5	69 x 63	125,116	70% of original specified fabric	C-15d
Polyfiber™ Medium-3	3.16	60 x 60	106,113	70% of original specified fabric	C-15d
Polyfiber™ Uncertified Light	1.87	90 x 76	66,72	uncertified	
Superflight™ SF 101	3.7	70 x 51	80,130	70% of original specified fabric	C-15d
Superflight™ SF 102	2.7	72 x 64	90,90	70% of original specified fabric	C-15d
Superflight™ SF 104	1.8	94 x 91	75,55	uncertified	
Grade A Cotton	4.5	80 x 84	80,80	56 lb/in (70% of New)	C-15d

Figure 3-133. Approved fabrics for covering aircraft.

Reinforcing Tape

Reinforcing tape is most commonly used on rib caps after the fabric covering is installed to protect and strengthen the area for attaching the fabric to the ribs.

Rib Bracing

Rib bracing tape is used on wing ribs before the fabric is installed. It is applied spanwise and alternately wrapped around a top rib cap and then a bottom rib cap progressing from rib to rib until all are braced. (***Figure 3-134***) Lacing the ribs in this manner holds them in the proper place and alignment during the covering process.

Surface Tape

Surface tape, made of polyester material and often preshrunk, is obtained from the STC holder. This tape, also known as finishing tape, is applied after the fabric is installed. It is used over seams, ribs, patches, and edges. Surface tape can have straight or pinked edges and comes in various widths. For curved surfaces, bias cut tape is available, which allows the tape to be shaped around a radius.

Rib Lacing Cord

Rib lacing cord is used to lace the fabric to the wing ribs. It must be strong and applied as directed to safely transfer in-flight loads from the fabric to the ribs. Rib lacing cord is available in a round or flat cross section. The round cord is easier to use than the flat lacing, but if installed properly, the flat lacing results in a smoother finish over the ribs.

Sewing Thread

Sewing of polyester fabric is rare and mostly limited to the creation of prefitted envelopes used in the envelope method covering process. When a fabric seam must be made with no structure underneath it, a sewn seam could be used. Polyester threads of various specifications are used on polyester fabric. Different thread is specified for hand versus machine sewing. For hand sewing, the thread is typically a three ply, uncoated polyester with a 15 pound tensile strength. Machine thread is typically four-ply polyester with a 10 pound tensile strength.

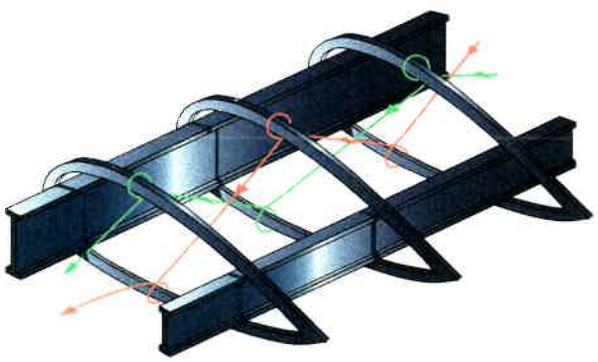


Figure 3-134. Inter-rib bracing holds the ribs in place during the covering process.

Special Fabric Fasteners

Each fabric covering job involves a method of attaching the fabric to wing and empennage ribs. The original manufacturer's method of fastening should be used. In addition to lacing the fabric to the ribs with approved rib lacing cord, special clips, screws, and rivets are employed on some aircraft. (***Figure 3-135***) The first step in using any of these fasteners is to inspect the holes into which they fit. Worn holes may have to be enlarged or re-drilled according to the manufacturer's instructions. Use of approved fasteners is mandatory. Use of unapproved fasteners can render the covering job unairworthy if substituted. Screws and rivets often incorporate the use of a plastic or aluminum washer. All fasteners and rib lacing are covered with finishing tape once installed to provide a smooth finish and airflow.

Grommets

Grommets are used to create reinforced drain holes in the aircraft fabric. Usually made of aluminum or plastic, they are glued or doped into place on the fabric surface. Once secured, a hole is created in the fabric through the center of the grommet. Often, this is done with a hot soldering pencil that also heat seals the fabric edge to prevent raveling. Seaplane grommets have a shield over the drain hole to prevent splashed water from entering the interior of the covered structure and to assist in siphoning out any water from within. (***Figure 3-136***) Drain holes using these grommets must be made before the grommets are put in place. Note that some drain holes do not require grommets if they are made through two layers of fabric.

Inspection Rings

The structure underneath an aircraft covering must be inspected periodically. To facilitate this in fabric covered aircraft, inspection rings are glued or doped to the fabric. They provide a stable rim around an area of fabric that can be cut to allow viewing of the structure underneath. The fabric remains uncut until

an inspection is desired. The rings are typically plastic or aluminum with an approximately three inch inside diameter. Spring clip metal panel covers can be fitted to close the area once the fabric inside the inspection ring has been cut for access. (***Figure 3-137***)

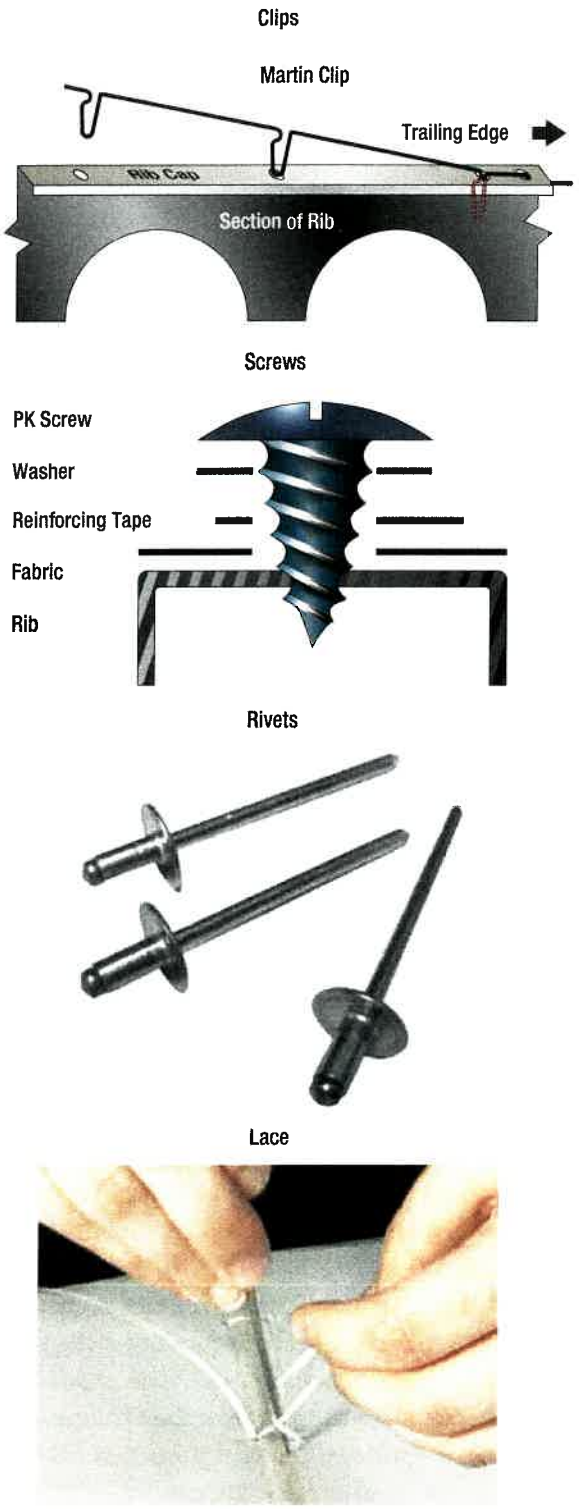


Figure 3-135. Clips, screws, rivets, or lace are used to attach the fabric to wing and empennage ribs.

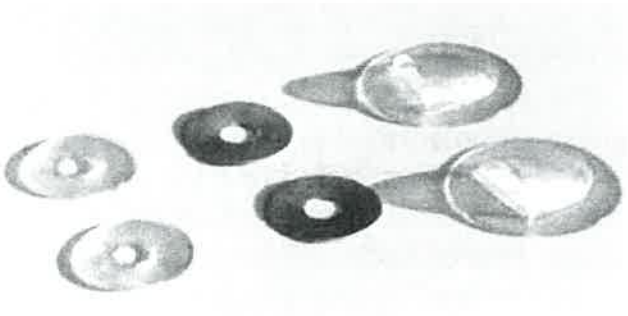


Figure 3-136. Plastic, aluminum, and seaplane grommets are used to reinforce drain holes in the fabric covering.

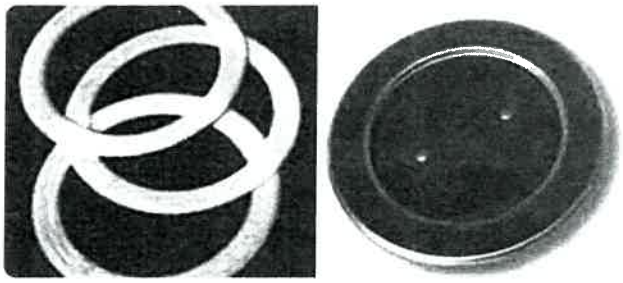


Figure 3-137. Inspection rings and an inspection cover.

The location of the inspection rings are specified by the manufacturer. Additional rings are sometimes added to permit access to important areas that may not have been fitted originally with inspection access.

Primer

The airframe structure of a fabric covered aircraft must be cleaned, inspected, and prepared before the fabric covering process begins. The final preparation procedure involves priming the structure with a treatment that works with the adhesive and first coats of fabric sealant that are to be utilized. Each STC specifies which primers, or if a wood structure, which varnishes are suitable. Most often, two part epoxy primers are used on metal structure and two part epoxy varnishes are used on wood structure. Utilize the primer specified by the manufacturer's or STC's instructions.

Fabric Cement

Modern fabric covering systems utilize special fabric cement to attach the fabric to the airframe. There are various types of cement. (***Figure 3-138***)

In addition to good adhesion qualities, flexibility, and long life, fabric cements must be compatible with the primer and the fabric sealer that are applied before and after the cement.

Aircraft Covering Systems							
APPROVED PROPRIETARY PRODUCT NAME							
Covering System	STC #	Allowable Fabrics	Base	Cement	Filler	UV Block	Topcoats
Air-Tech	SA7965SW	Ceconite™ Poly-Fiber™ Superflite™	Urethane Water	UA-55	PFU 1020 PFU 1030 PFUW 1050	PFU 1020 PFU 1030 PFUW 1050	CHSM Color Coat
Ceconite™/ Randolph System	SA4503NM	Ceconite™	Dope	New Super Seam	Nitrate Dope	Rand-O-Fill	Colored Butyrate Dope Ranthane Polyurethane
Stits/Poly-Fiber™	SA1008WE	Poly-Fiber™	Vinyl	Poly-tak	Poly-brush	Poly-spray	Vinyl Poly-tone, Aero-Thane, or Ranthane Polyurethane
Stewart System	SA01734SE	Ceconite™ Poly-Fiber™	Water-borne	EkoBond	EkoFill	EkoFIll	EkoPoly
Superflite™ • System1 • System VI	SA00478CH and others	Superflite™ 101,102 Superflite™ 101,102	Dope Urethane	U-500 U-500	Dacproofer SF6500	SrayFil SF6500	Tinted Butyrate Dope Superflite™ CAB

Figure 3-138. Current FAA-approved fabric covering processes.

Fabric Sealer

Fabric sealer surrounds the fibers in the fabric with a protective coating to provide adhesion and keep out dirt and moisture. The sealer is the first coat applied to the polyester fabric after it is attached to the airframe and heat shrunk to fit snugly. Dope based fabric coating systems utilize non-tautening nitrate dope as the primary fabric sealant. The application of tautening dope may cause the fabric to become too taut resulting in excess stress on the airframe that could damage it. Non-dope coating systems use proprietary sealers that are also non-tautening. (***Figure 3-138***)

Fillers

After the fabric sealer is applied, a filler is used. It is sprayed on in a number of cross coats as required by the manufacturer or the fabric covering process STC. The filler contains solids or chemicals that are included to block UV light from reaching the fabric. Proper fill coating is critical because UV light is the single most destructive element that causes polyester fabric to deteriorate. Dope based processes use butyrate dope fillers while other processes have their own proprietary formulas. When fillers and sealers are combined, they are known as fabric primers. Aluminum pastes and powders, formerly added to butyrate dope to provide the UV protection, have been replaced by premixed formulas.

Topcoats

Once the aircraft fabric has been installed, sealed, and fill coat protected, finishing or topcoats are applied to give the aircraft its final appearance. Colored butyrate dope is common in dope based processes, but various polyurethane topcoats are also available. It is important to use the topcoat products and procedures specified in the applicable STC to complete an airworthy fabric recovering job.

The use of various additives is common at different stages when utilizing the above products. The following is a short list of additional products that facilitate the proper application of the fabric coatings. Note again that only products approved under a particular STC can be used. Substitution of similar products, even though they perform the same basic function, is not allowed.

- A catalysts accelerates a chemical reaction. Catalysts are specifically designed for each product with which they are mixed. They are commonly used with epoxies and polyurethanes.
- A thinner is a solvent or mixture of solvents added to a product to give it the proper consistency for application, such as when spraying or brushing.
- A retarder is added to a product to slow drying time. Used mostly in dope processes and topcoats, a retarder allow more time for a sprayed coating to flow and level, resulting in a deeper, glossier finish. It is used when the working temperature is elevated slightly above the ideal temperature for a product. It also can be used to prevent blushing of a dope finish when high humidity conditions exist.
- An accelerators contains solvents that speed up the drying time of the product with which it is mixed. It is typically used when the application working temperature is below that of the ideal working temperature. It can also be used for faster drying when airborne contaminants threaten a coating finish.
- Rejuvenator, used on dope finishes only, contains solvents that soften coatings and allow them to flow slightly. Rejuvenator also contains fresh plasticizers that mix into the original coatings. This increases the overall flexibility and life of the coatings.
- Fungicide and mildewicide additives are important for organic fabric covered aircraft because fabrics, such as cotton and linen, are hosts for fungus and mildew. Since fungus and mildew are not concerns when using polyester fabric, these additives are not required. Modern coating formulas contain premixed anti-fungal agents, providing sufficient insurance against the problem of fungus or mildew.

AVAILABLE COVERING PROCESSES

The covering processes that utilize polyester fabric are the primary focus of this chapter. The approved aircraft covering processes are listed in ***Figure 3-138***. The processes can be distinguished by the chemical nature of the glue and coatings that are used. A dope-based covering process has been refined out of the cotton fabric era, with excellent results on polyester fabric. In particular, plasticizers added to the nitrate dope and butyrate dopes minimize the shrinking and tautening effects of the dope, establish flexibility, and allow esthetically pleasing tinted butyrate dope finishes that last indefinitely.

Durable polyurethane based processes integrate well with durable polyurethane topcoat finishes. Vinyl is the key ingredient in the popular Poly-Fiber™ covering system. Air Tech uses an acetone thinned polyurethane compatible system.

The most recent entry into the covering systems market is the Stewart Finishing System that uses waterborne technology to apply polyurethane coatings to the fabric. The glue used in the system is water based and nonvolatile. The Stewart Finishing System is Environmental Protection Agency (EPA) compliant and STC approved. Both the Stewart and Air Tech systems operate with any of the approved polyester fabrics as stated in their covering system STCs.

All the modern fabric covering systems listed in ***Figure 3-138*** result in a polyester fabric covered aircraft with an indefinite service life. Individual preferences exist for working with the different approved processes. A description of basic covering procedures and techniques common to most of these systems follows later in this chapter.

Ceconite™, Poly-Fiber™, and Superflight™ are STC approved fabrics with processes used to install polyester fabric coverings. Two companies that do not manufacturer their own fabric have gained STC approval for covering accessories and procedures to be used with these approved fabrics. The STCs specify the fabrics and the proprietary materials that are required to legally complete the recovering job.

The aircraft fabric covering process is a three step process. First, select an approved fabric. Second, follow the applicable STC steps to attach the fabric to the airframe and to protect it from the elements. Third, apply the approved topcoat to give the aircraft its color scheme and final appearance. Although Grade-A cotton can be used on all aircraft originally certificated to be covered with this material, approved aircraft cotton fabric is no longer available. Additionally, due to the shortcomings of cotton fabric coverings, most of these aircraft have been recovered with polyester fabric.

TYPES OF DEFECTS IN FABRICS

GENERAL

The fabric covering of an aircraft will deteriorate in time. The rate of deterioration depends to a large extent on the type of operation, climate, storage conditions, and the maintenance of a satisfactory surface finish. A great deal depends on how well the fabric has been protected from UV (ultraviolet). Due to water soakage, chafing against the structure and local wear, the covering will not deteriorate uniformly. In the case of the fabric covered components on large aircraft, an arbitrary life may be placed on the fabric. The fabric coverings of light aircraft should be checked at the periods specified in the approved maintenance schedule and before renewal of the Airworthiness Certificate. In general terms, all components of the covering should be regularly inspected for the following which are unacceptable:

- Loose finishing tape and reinforcing patches
- Chafing under fairings
- Brittleness, cracking, peeling, or deteriorated coatings
- Fabric tears and stone damage
- Broken or missing rib stringing
- Rodent and insect nests

The entire fabric covering should be uniformly taut with no loose or wrinkled areas. The fabric tension must not be excessive otherwise the underlying structure can be warped or damaged.

EXCESS TENSION

There are no methods of measuring fabric tension other than by observation. Excessive tension may warp longerons, wing ribs and trailing edges thus weakening the structure.

LOOSE FABRIC

Fabric finished with dope that flutters or ripples in the propeller slipstream, balloons, or is depressed excessively in flight is unacceptable. These issues may be caused by:

- Inadequate dope film
- Poor quality dope
- Fabric attached with excessive slack
- Bent or warped structure

Temporary wrinkles will develop in any fabric which is coated and finished with dope when moisture from rain, fog, or dew is absorbed into a poor quality dope film, causing the film to expand. Temporary wrinkles may also develop with any type of thick coating, in any type of fabric, when an aircraft is moved from a cold storage area to a warm hangar or parked in the warming sunshine causing rapid expansion of the coating. Loose or wrinkled polyester fabric covering finished with coatings other than dope may be caused by:

- Inadequate or excessive heat application during tautening
- Excess slack when the fabric was attached
- Bent broken or warped structure

Polyester fabric which does not meet aircraft quality specifications is likely to become loose after a short period of time.

Particular attention must be given to the area within the propeller slipstream. If failure is indicated by lifting of the fabric covering, the rib stringing cord and reinforcing tape must be reinstalled with double the number of original stitches.

COATING CRACKS

Fabric exposed through cracks in the coating should be inspected for deterioration by pressing firmly with a thumb to check the fabric's strength. Natural fibers deteriorate by exposure to ultraviolet radiation, mildew, fungus from moisture, high acid content, rain, dew, fog, pollution, and age. Polyester filaments will deteriorate by exposure to UV radiation.

Glass fabric will not deteriorate from exposure to UV, but will if exposed to acid rain, dew, or chafing if loose in the propeller slipstream area.

Polyester fabric that is coated with materials other than dope is dependent solely on the heat tautening characteristics of the polyester filaments to develop tension and transmit the airloads to the airframe without excessive deflection from the unloaded position.

Cracks in dope and other coatings will allow any type of exposed fabric to deteriorate. Cracks should be closed by sealing, or by removing the coatings in the immediate area and then replacing them with new coatings.

DETERIORATION

Polyester fabric deteriorates only by exposure to ultraviolet radiation. When approved coatings completely protect the fabric, its service life is infinite.

Therefore it is very important to thoroughly protect the structure beneath before covering and to provide adequate access in all areas of fabric covered components to allow inspection for corrosion, wood rot, and mice infestation. Multiple drain holes in the lower ends of all fabric covered sections also provide needed ventilation to remove condensation.

INSPECTION METHODS

DETERMINING FABRIC CONDITION; REPAIR OR RECOVER?

Recovering an aircraft with fabric is a major repair and should only be undertaken when necessary. Often a repair to the present fabric is sufficient to keep the aircraft airworthy. The original manufacturer's recommendations or the covering process STC should be consulted for the type of repair required for the damage incurred by the fabric covering.

Often a large area that needs repair is judged in reference to the overall remaining lifespan of the fabric on the aircraft. For example, if the fabric has reached the limit of its durability, it is better to recover the entire aircraft than to replace a large damaged area when the remainder of the aircraft would soon need to be recovered.

On aircraft with dope based covering systems, continued shrinkage of the dope can cause the fabric to become too tight. Overly tight fabric may require the aircraft to be recovered rather than repaired because excess tension on fabric can cause airframe structural damage. Loose fabric flaps in the wind during flight, affecting weight distribution and unduly stressing the airframe. It may also need to be replaced because of damage to the airframe.

Another reason to recover rather than repair occurs when dope coatings on fabric develop cracks. These cracks could expose the fabric beneath to the elements that can weaken it. Close observation and field testing must be used to determine if the fabrics are airworthy. If not, the aircraft must be recovered. If the fabric is airworthy and no other problems exist, a rejuvenator can be used per manufacturer's instructions. This product is usually sprayed on and softens the coatings with very powerful solvents. Plasticizers in the rejuvenator become part of the film that fills in the cracks. After the rejuvenator dries, additional coats of aluminum pigmented dope must be added and then final topcoats applied to finish the job. While laborious, rejuvenating a dope finish over strong fabric can save a great deal of time and money. Polyurethane-based finishes cannot be rejuvenated.

FABRIC STRENGTH

Deterioration of the strength of the present fabric covering is the most common reason to recover an aircraft. The strength of fabric coverings must be determined at every 100 hour and annual inspection. Minimum fabric breaking strength is used to determine if an aircraft requires recovering.

Fabric strength is a major factor in the airworthiness of an aircraft. Fabric is considered to be airworthy until it deteriorates to a breaking strength less than 70 percent of the strength of the new fabric required for the aircraft. For example, if an aircraft was certificated with Grade-A cotton fabric that has a new breaking strength of 80 pounds, it becomes unairworthy when the fabric strength falls to 56 pounds, which is 70 percent of 80 pounds. If polyester fabric, which has a higher new breaking strength, is used to recover this same aircraft, it would also need to exceed 56 pounds breaking strength to remain airworthy.

In general, an aircraft is certified with a certain fabric based on its wing loading and its Never Exceed Speed (VNE). The higher the wing loading and VNE, the stronger the fabric must be. On aircraft with wing loading of 9 pounds per square foot and over, or a VNE of 160 miles per hour (mph) or higher, fabric equaling or exceeding the strength of Grade-A cotton is required. This means the new fabric breaking strength must be at least 80 pounds and the minimum fabric breaking strength at which the aircraft becomes unairworthy is 56 pounds.

On aircraft with wing loading of 9 pounds per square foot or less, or a VNE of 160 mph or less, fabric equaling or exceeding the strength of intermediate grade cotton is required. This means the new fabric breaking strength must be at least 65 pounds and the minimum fabric breaking strength at which the aircraft becomes unairworthy is 46 pounds.

Lighter weight fabric may be found to have been certified on gliders or sailplanes and may be used on many uncertified aircraft. For aircraft with wing loading less than 8 pounds per square foot or less, or VNE of

Fabric Performance Criteria				
IF YOUR PERFORMANCE IS. . .		FABRIC STRENGTH MUST BE. . .		
Loading	V_{NE} Speed	Type	New Breaking Strength	Minimum Breaking Strength
> 9 lb/sq ft	> 160 mph	≥ Grade A	> 80 lb	> 56
< 9 lb/sq ft	< 160 mph	≥ Intermediate	> 65 lb	> 46
< 8 lb/sq ft	< 135 mph	≥Lightweight	> 50 lb	> 35

Figure 3-139. Aircraft performance affects fabric selection.

135 mph or less, the fabric is considered unairworthy when the breaking strength has deteriorated to below 35 pounds (new minimum strength of 50 pounds). ***Figure 3-139*** summarizes these parameters.

How Fabric Breaking Strength is Determined

Manufacturer's instructions should always be consulted first for fabric strength inspection methodology. These instructions are approved data and may not require removal of a test strip to determine airworthiness of the fabric. In some cases, the manufacturer's information does not include any fabric inspection methods.

The test strip method for the breaking strength of aircraft covering fabrics uses standards published by the American Society for Testing and Materials (ASTM) for the testing of various materials. Breaking strength is determined by cutting a 1¼ inch by 4–6 inch strip of fabric from the aircraft covering. This sample should be taken from an area that is exposed to the elements; usually an upper surface. It is also wise to take the sample from an area that has a dark colored finish since this has absorbed more of the sun's UV rays and degraded faster. All coatings are then removed and the edges raveled to leave a 1 inch width. One end of the strip is clamped into a secured clamp and the other end is clamped such that a suitable container may be suspended from it. Weight is added to the container until the fabric breaks. The breaking strength of the fabric is equal to the weight of the lower clamp, the container, and the weight added to it. If the breaking strength is still in question, a sample should be sent to a qualified testing laboratory and breaking strength tests made in accordance with ASTM publication D5035.

NOTE: The fabric test strip must have all coatings removed from it for the test. Soaking and cleaning the test strip in Methyl Ethyl Ketone (MEK) usually removes all the coatings.

Properly installed and maintained polyester fabric should give years of service before appreciable fabric strength degradation occurs. Aircraft owners often prefer not to have test strips cut out of the fabric, especially when the aircraft or the fabric covering is relatively new, because removal of a test strip damages the integrity of an airworthy component if the fabric passes. The test strip area then must be repaired, costing time and money. To avoid cutting a strip out of airworthy fabric, the IA makes a decision based on knowledge, experience, and available nondestructive techniques as to whether removal of a test strip is warranted to ensure that the aircraft can be returned to service.

An aircraft made airworthy under an STC is subject to the instructions for continued airworthiness in that STC. Poly-Fiber™ and Ceconite™ re-covering process STCs contain their own instructions and techniques for determining fabric strength and airworthiness. Therefore, an aircraft covered under those STCs may be inspected in accordance with this information. In most cases, the aircraft can be approved for return to service without cutting a strip from the fabric covering.

The procedures in the Poly-Fiber™ and Ceconite™ STCs outlined in the following paragraphs are useful when inspecting any fabric covered aircraft. However, following these procedures alone on aircraft not recovered under these STCs does not make the aircraft airworthy. The technician must add his or her own knowledge, experience, and judgment to make a final determination of the strength of the fabric and whether it is airworthy.

Exposure to UV radiation appreciably reduces the strength of polyester fabric and forms the basis of the Poly-Fiber™ and Ceconite™ fabric evaluation process. All approved covering systems utilize fill coats applied to the fabric to protect it from UV. If installed according to the STC, these coatings should be sufficient to protect the fabric from the sun and should last indefinitely. Therefore, most of the evaluation of the strength of the fabric is actually an evaluation of the condition of its protective coating(s).

Upon a close visual inspection, the fabric coating(s) should be consistent, contain no cracks, and be flexible, not brittle. Pushing hard against the fabric with a knuckle should not damage the coating(s). It is recommended the inspector check in several areas, especially those most exposed to the sun. Coatings that pass this test can move to a simple test that determines whether or not UV light is passing through the coatings.

This test is based on the assumption that if visible light passes through the fabric coatings, then UV light can also. To verify whether or not visible light passes through the fabric coating, remove an inspection panel from the wing, fuselage, or empennage. Have someone hold an illuminated 60 watt lamp one foot away from the exterior of the fabric. No light should be visible through the fabric. If no light is visible, the fabric has not been weakened by UV rays and can be assumed to be airworthy. There is no need to perform the fabric strip strength test. If light is visible through the coatings, further investigation is required.

FABRIC TESTING DEVICES

Mechanical devices used to test fabric by pressing against or piercing the finished fabric are not approved and are used at the discretion of the certificated mechanic to form an opinion on the general fabric condition. Punch test accuracy depends on the individual device calibration, total coating thickness, brittleness, and types of coatings and fabric. If the fabric tests in the lower breaking strength range with the mechanical punch tester or if the overall fabric cover conditions are poor, then more accurate field tests may be made.

The test should be performed on exposed fabric where there is a crack or chip in the coatings. If there is no crack or chip, coatings should be removed to expose the fabric wherever the test is to be done.

The Maule punch tester, a spring loaded device with its scale calibrated in breaking strength, tests fabric strength by pressing against it while the fabric is still on the aircraft. It roughly equates strength in pounds per square inch (psi) of resistance to breaking strength. The tester is pushed squarely against the fabric until the scale reads the amount of maximum allowable degradation. If the tester does not puncture the fabric, it may be considered airworthy. Punctures near the breaking strength should be followed with further testing, specifically the strip breaking strength test described above. Usually, a puncture indicates the fabric is in need of replacement.

A second type of punch tester, the Seyboth, is not as popular as the Maule because it punctures a small hole in the fabric when the mechanic pushes the shoulder of the testing unit against the fabric. A pin with a color coded calibrated scale protrudes from the top of the tester and the mechanic reads this scale to determine fabric strength. Since this device requires a repair regardless of the strength of the fabric indicated, it is not widely used.

Seyboth and Maule fabric strength testers designed for cotton and linen covered aircraft, not to be used on modern Dacron fabrics. Mechanical devices, combined with other information and experience, help the certificated mechanic judge the strength of the fabric. (***Figure 3-140***)

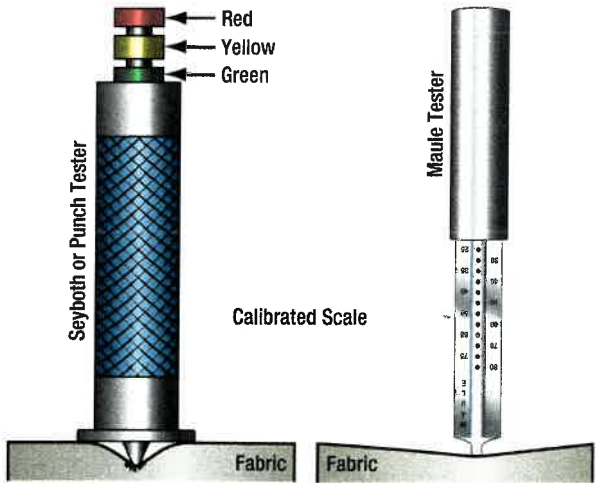


Figure 3-140. Seyboth and Maule fabric strength testers.

REPAIR OF FABRIC COVERING

This section describes steps common to various STC and manufacturer covering processes, as well as the differences of some processes. To aid in proper performance of fabric covering and repair procedures, STC holders produce illustrated, step by step instructional manuals and videos that demonstrate the correct covering procedures. These training aids are invaluable to the inexperienced technician.

Since modern fabric coverings last indefinitely, a rare opportunity to inspect the aircraft exists during the recovering process. Inspectors and owner-operators should use this opportunity to perform a thorough inspection of the aircraft before new fabric is installed. The method of fabric attachment should be identical, as far as strength and reliability are concerned, to the method used by the manufacturer of the aircraft being recovered or repaired. Carefully remove the old fabric from the airframe, noting the location of inspection covers, drain grommets, and method of attachment. Either the envelope method or blanket method of fabric covering is acceptable, but a choice must be made prior to beginning the recovering process.

Blanket Method Versus Envelope Method

In the blanket method of recovering, multiple flat sections of fabric are trimmed and attached to the airframe. Certified greige polyester fabric for covering an aircraft can be up to 70 inches in width and used as it comes off the bolt. Each aircraft must be considered individually to determine the size and layout of blankets needed to cover it. A single blanket cut for each small surface (i.e., stabilizers and control surfaces) is common. Wings may require two blankets that overlap. Fuselages are covered with multiple blankets that span between major structural members, often with a single blanket for the bottom. Very large wings may require more than two blankets of fabric to cover the entire top and bottom surfaces. In all cases, the fabric is adhered to the airframe using the approved adhesives, following specific rules for the covering process being employed. (***Figure 3-141***)

An alternative method of recovering, the envelope method, saves time by using precut and presewn envelopes of fabric to cover the aircraft. The envelopes must be sewn with approved machine sewing thread, edge distance, fabric fold, etc., or an STC. Patterns are made and fabric is cut and stitched so that each major surface, including the fuselage and wings, can be covered with a single, close fitting envelope. Since envelopes are cut to fit, they are slid into position, oriented with the seams in the proper place, and attached with adhesive to the airframe. Envelope seams are usually located over airframe structure in inconspicuous places, such as the trailing edge structures and the very top and bottom of the fuselage, depending on airframe construction. Follow the manufacturer's or STC's instructions for proper location of the sewn seams of the envelope when using this method. (***Figure 3-142***)

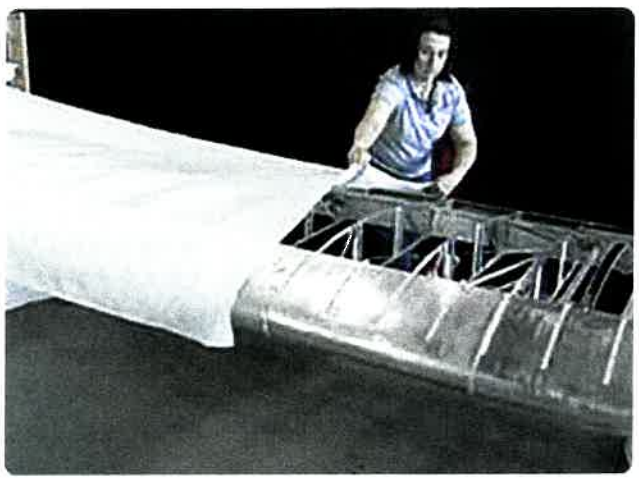


Figure 3-141. Laying out fabric during a blanket method recovering job.



Figure 3-142. Inspection rings and an inspection cover.

Preparation for Fabric Covering Work

Proper preparation for recovering a fabric aircraft is essential. First, assemble the materials and tools required to complete the job. The holder of the STC usually supplies a materials and tools list either separately or in the STC manual. Control of temperature, humidity, and ventilation is needed in the work environment. If ideal environmental conditions cannot be met, additives are available that compensate for this for most recovering products.

Rotating work stands for the fuselage and wings provide easy, alternating access to the upper and lower surfaces while the job is in progress. (***Figure 3-143***) They can be used with sawhorses or sawhorses can be used alone to support the aircraft structure while working. A workbench or table, as well as a rolling cart and storage cabinet, are also recommended. ***Figure 3-144*** shows a well conceived fabric covering workshop. A paint spray booth for sprayed-on coatings and space to store components awaiting work is also recommended.

Many of the substances used in most recovering processes are highly toxic. Proper protection must be used to avoid serious short and long term adverse health effects. Eye protection, a proper respirator, and skin protection are vital. As mentioned in the beginning of this chapter, nitrate dope is very flammable. Proper ventilation and a rated fire extinguisher should be on hand when working with this and other covering process

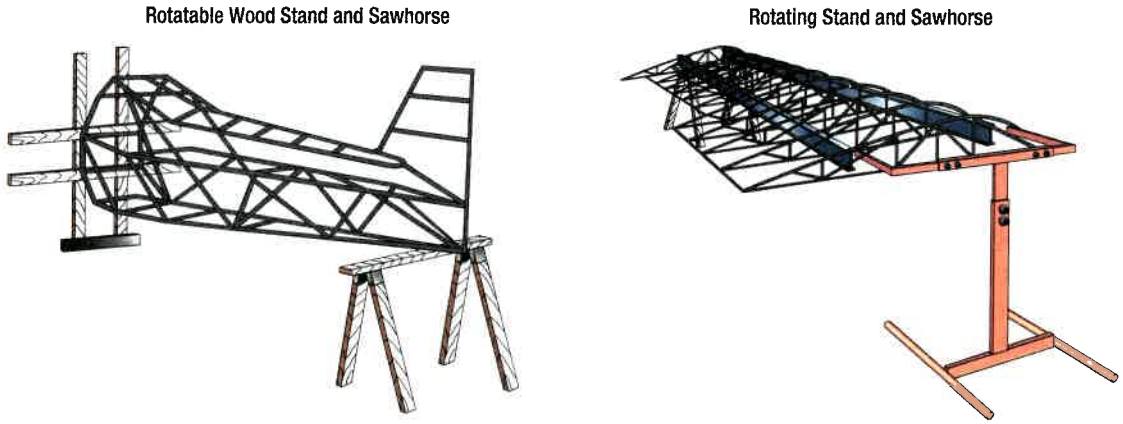


Figure 3-143. Rotating stands and sawhorses facilitate easy access to top and bottom surfaces during the fabric covering process.

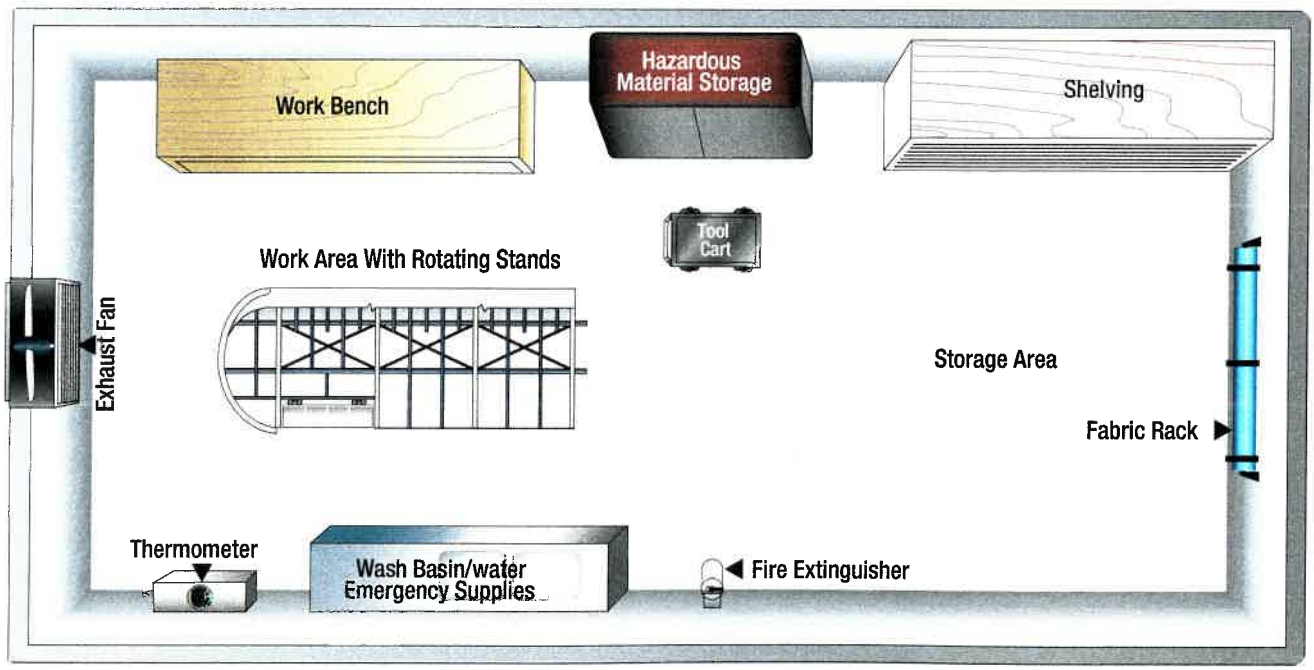


Figure 3-144. Some components of a work area for covering an aircraft with fabric.

materials. Grounding of work to prevent static electricity buildup may be required. All fabric recovering processes also involve multiple coats of various products that are sprayed onto the fabric surface. Use of a High-Volume, Low-Pressure (HVLP) sprayer is recommended. Good ventilation is needed for all of the processes.

Removal of Old Fabric Coverings

Removal of the old covering is the first step in replacing an aircraft fabric covering. Cut away the old fabric from the airframe with razor blades or utility knife. Care should be taken to ensure that no damage is done to the airframe. (***Figure 3-145***) To use the old covering for templates in transferring the location of inspection panels, cable guides, and other features to the new covering, the old covering should be removed in large sections. Note: any rib stitching fasteners, if used to attach the fabric to the structure, should be removed before the fabric is pulled free of the airframe. If fasteners are left in place, damage to the structure may occur during fabric removal.

Preparation of the Airframe Before Covering

Once the old fabric has been removed, the exposed airframe structure must be thoroughly cleaned and inspected. The technician collaborating on the job should be involved in this step of the process. Details of the inspection should follow the manufacturer's guidelines, or the STC. All of the old adhesive must be completely removed from the airframe with solvent, such as MEK. A thorough inspection must be done and various components may be selected to be removed for cleaning, inspection, and testing. Any repairs that are required, including the removal and treatment of all corrosion, must be done at this time. If the airframe is steel tubing, many technicians take the opportunity to grit blast the entire airframe at this stage.

The leading edge of a wing is a critical area where airflow diverges and begins its laminar flow over the wing's surfaces, which results in the generation of lift. It is beneficial to have a smooth, regular surface in this area. Plywood leading edges must be sanded until smooth, bare wood is exposed. If oil or grease spots exist, they must be cleaned with naphtha or other specified cleaners. If there are any chips, indentations, or irregularities, approved filler may be spread into these areas and sanded smooth. The entire leading edge should be cleaned before beginning the fabric covering process.

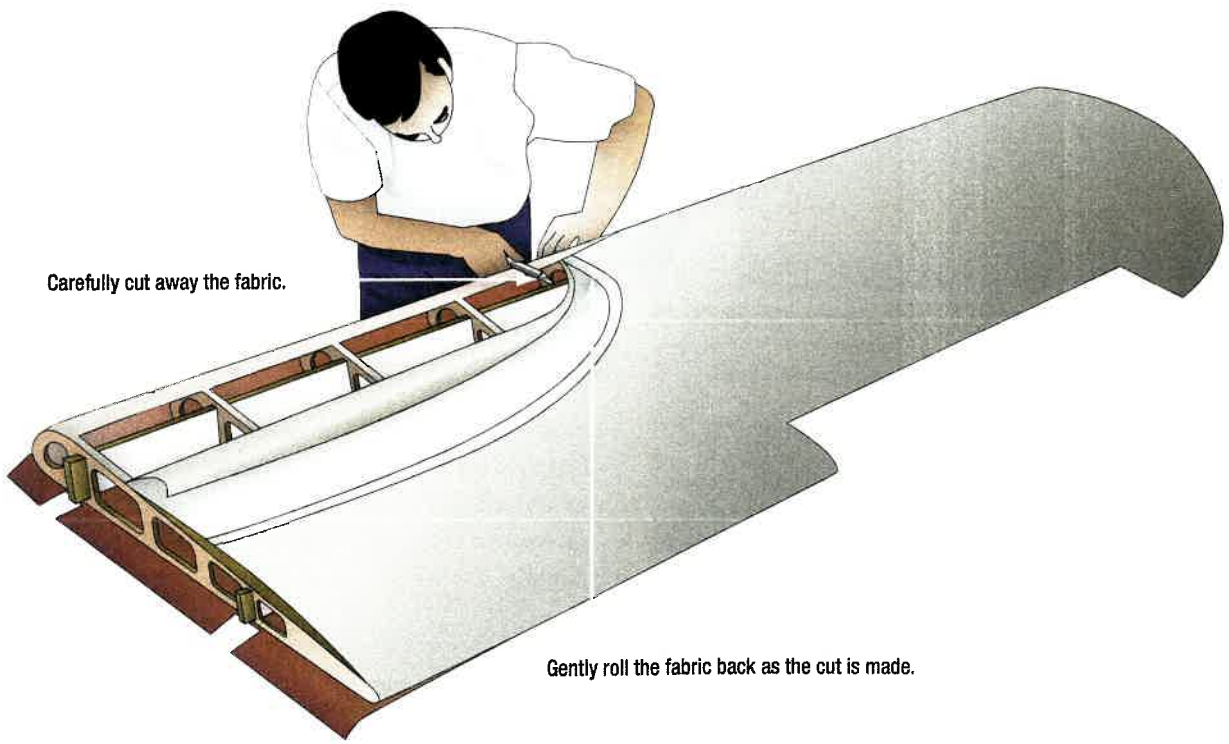


Figure 3-145. Old fabric coverings are cut off in large pieces to preserve them as templates for locating various airframe features. Sharp blades and care must be used to avoid damaging the structure.

To obtain a smooth finish on fabric covered leading edges of aluminum wings, a sheet of felt or polyester padding may be applied before the fabric is installed. This should only be done with the material specified in the STC under which the technician is working. The approved padding ensures compatibility with the adhesives and first coatings of the covering process. When a leading edge pad is used, check the STC process instructions for permission to make a cemented fabric seam over the padding. (***Figure 3-146***)

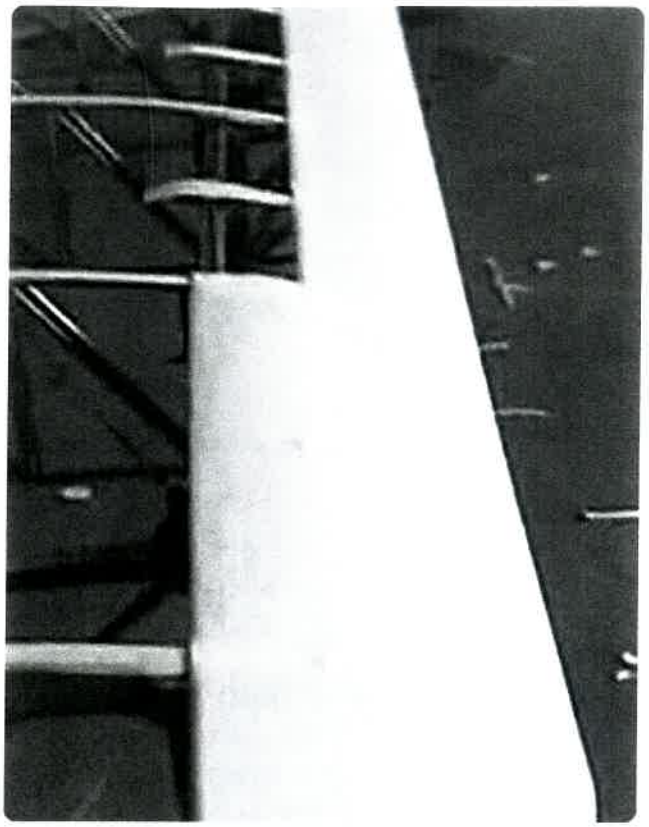


Figure 3-146. The use of specified felt or padding over the wing leading edges before the fabric is installed results in a smooth regular surface.

When completely cleaned, inspected, and repaired, an approved primer, or varnish if it is a wood structure, should be applied to the airframe. This step is sometimes referred to as dope proofing. Exposed aluminum must first be acid etched. Use the product(s) specified by the manufacturer or in the STC to prepare the metal before priming. Two part epoxy primers and varnishes, which are not affected by the fabric adhesive and subsequent coatings, are usually specified. One part primers, such as zinc chromate and spar varnish, are typically not acceptable. The chemicals in the adhesives dissolve the primers, and adhesion of the fabric to the airframe is lost.

Sharp edges, metal seams, the heads of rivets, and any other feature on the aircraft structure that might cut or wear through the fabric should be covered with anti-chafe tape. As described above, this cloth sticky back tape is approved and should not be substituted with masking or any other kind of tape. Sometimes, rib cap strips need to have anti-chafe tape applied when the edges are not rounded over. (***Figure 3-147***)

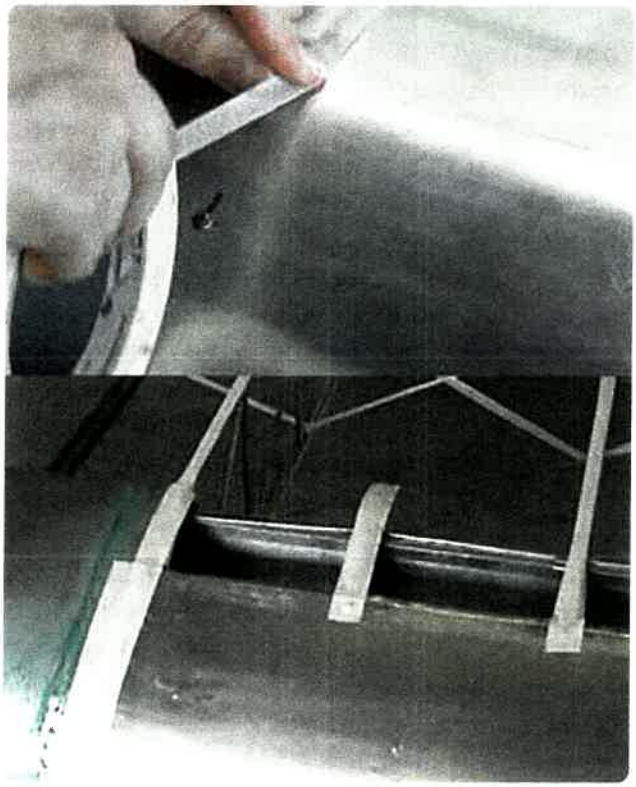


Figure 3-147. Anti-chafe tape is applied to all features that might cut or wear through the fabric.

Inter-rib bracing must also be accomplished before the fabric is installed. It normally does not have an adhesive attached to it and is wrapped only once around each rib. The single wrap around each rib is enough to hold the ribs in place during the covering process but allows small movements during the fabric shrinking process. (***Figure 3-148***)

ATTACHING POLYESTER FABRIC TO THE AIRFRAME

Inexperienced technicians are encouraged to construct a test panel upon which they can practice with the fabric and various substances and techniques to be used on the aircraft. It is often suggested to cover smaller surfaces first, such as the empennage and control surfaces. Mistakes on these can be corrected and are less costly

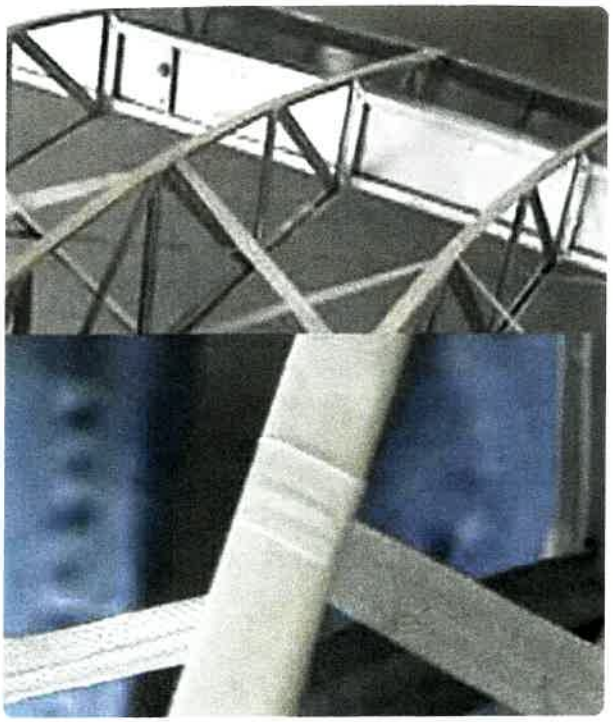


Figure 3-148. Inter-rib bracing holds the ribs in place during the re-covering process.

if they occur. The techniques employed for all surfaces, including the wings and fuselage, are basically the same. Once dexterity has been established, the order in which one proceeds is often a personal choice.

When approved, attachment of the fabric may begin. The manufacturer's or STC's instructions must be followed without deviation for the job to be airworthy. The following are the general steps taken. Each approved process has its own nuances.

Seams

During installation, the fabric is overlapped and seamed together. Primary concerns for fabric seams are strength, elasticity, durability, and good appearance. Whether using the blanket method or envelope method, position all fabric seams over the airframe structure to which the fabric is to be adhered during the covering process, whenever possible. Unlike the blanket method, fabric seam overlap is predetermined in the envelope method. Seams are sewn to the specifications in the STC under which the work is being performed, or the manufacturer's instructions should perform adequately.

Most covering procedures for polyester fabric rely on doped or glued seams as opposed to sewn seams. They are simple and easy to make and provide excellent strength, elasticity, durability, and appearance. When using the blanket method, seam overlap is specified in the covering instructions and the certificated mechanic must adhere to these specifications. Typically, a minimum of two to four inches of fabric overlap seam is required where ends of fabric are joined in areas of critical airflow, such as the leading edge of a wing. One to two inches of overlap is often the minimum in other areas. When using the blanket method, options exist for deciding where to overlap the fabric for coverage.

Function and the final appearance of the covering job should be considered. For example, fabric seams made on the wing's top surface of a high wing aircraft are not visible when approaching the aircraft. Seams on low wing aircraft and many horizontal stabilizers are usually made on the bottom of the wing for the same reason. (***Figure 3-149***)

Fabric Cement

A polyester fabric covering is cemented or glued to the airframe structure at all points where it makes contact. Special formula adhesives have replaced nitrate dope for adhesion in most covering processes. The adhesive (as well as all subsequent coating materials) should be mixed for optimum characteristics at the temperature at which the work is being performed. Follow the manufacturer's or STC's guidance when mixing.

To attach the fabric to the airframe, first pre-apply two coats of adhesive to the structure at all points the fabric is to contact it. (It is important to follow the manufacturer's or STC's guidance as all systems are different.) Allow these to dry. The fabric is then spread over the surface and clamped into position. It should not be pulled tighter than the relaxed but not wrinkled condition it assumes when lying on the structure. Clamps or clothespins are used to attach the fabric completely around the perimeter. The Stewart System STC does not need clamps because the glue assumes a tacky condition when precoated and dried. There is sufficient adhesion in the precoat to position the fabric.

The fabric should be positioned in all areas before undertaking final adhesion. Final adhesion often involves lifting the fabric, applying a wet bed of cement, and pressing the fabric into the bed. An additional coat of cement over the top of the fabric is common. Depending on the process, wrinkles and excess cement

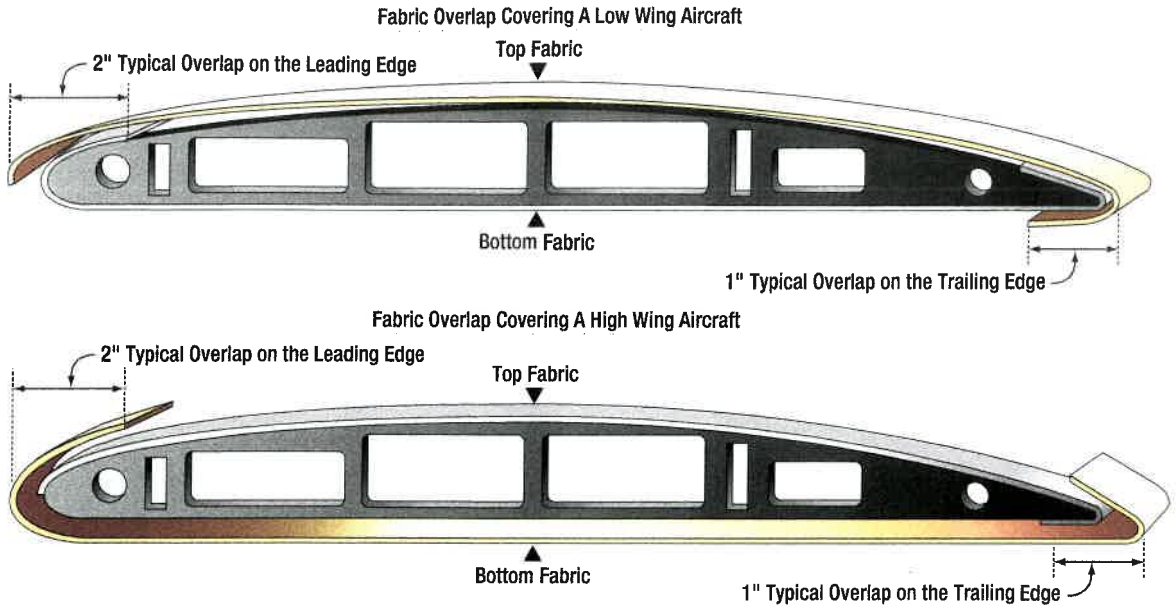


Figure 3-149. For appearance, fabric can be overlapped differently on high wing and low wing aircraft.

are smoothed out with a squeegee or are ironed out. The Stewart System calls for heat activation of the cement precoats through the fabric with an iron while the fabric is in place. Follow the approved instructions for the covering method being used.

FABRIC HEAT SHRINKING

Once the fabric has been glued to the structure, it can be made taut by heat shrinking. This process is done with an ordinary household iron that the technician calibrates before use. A smaller iron is also used to iron in small or tight places. (***Figure 3-150***)



Figure 3-150. Irons used during the fabric covering process.

The iron is run over the entire surface of the fabric. Follow the instructions for the work being performed. Some processes avoid ironing seams while other processes begin ironing over structure and move to spanned fabric or visa versa. It is important to shrink the fabric evenly. Starting on one end of a structure and progressing sequentially to the other end is not recommended. Skipping from one end to the other, and then to the middle, is more likely to evenly draw the fabric tight. (***Figure 3-151***)

The amount polyester fabric shrinks is directly related to the temperature applied. Polyester fabric can shrink nearly 5 percent at 120°C and 10 percent at 175°C. It is customary to shrink the fabric in stages, using a lower temperature first, before finishing with the final temperature setting. The first shrinking is used to remove wrinkles and excess fabric. The final shrinking gives the finished tautness desired. Each process has its own temperature regime for the stages of tautening. Typically ranging from 107°C to 120°C, it is imperative to follow the process instructions. Not all fabric covering processes use the same temperature range and maximum temperature. Ensure irons are calibrated to prevent damage at high temperature settings.

ATTACHING FABRIC TO THE WING RIBS

Once the fabric has been tautened, covering processes vary. Some require a sealing coat be applied to the fabric at this point. It is usually put on by brush to ensure the fibers are saturated. Other processes seal the fabric later. Whatever the process, the fabric on wings must be

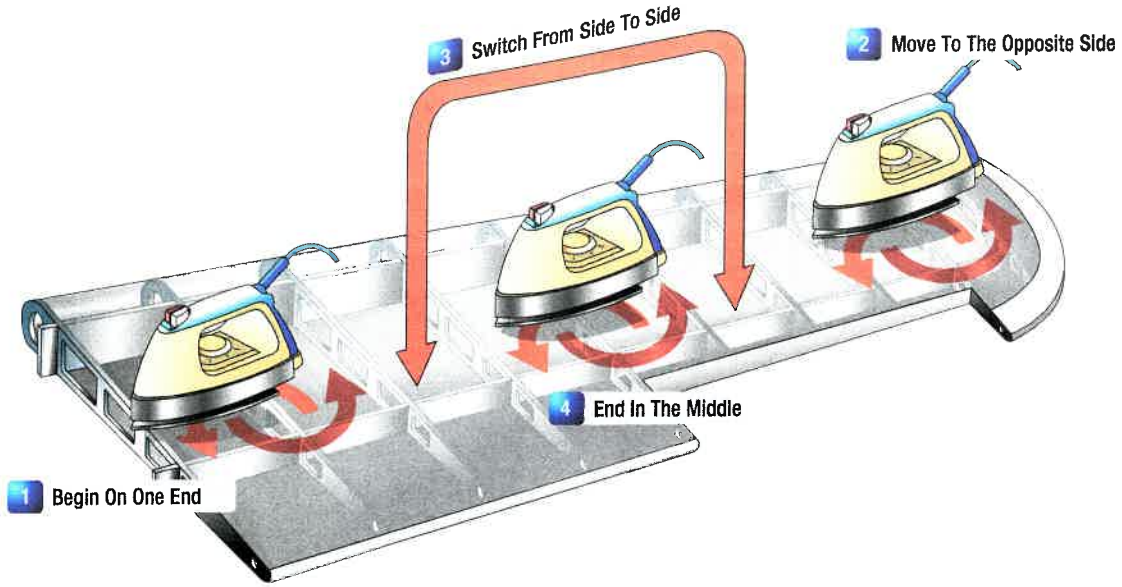


Figure 3-151. An example of a wing fabric ironing procedure designed to evenly taughten the fabric.

secured to the wing ribs with more than just cement. The forces caused by the airflow over the wings are too great for cement alone to hold the fabric in place. As described in the materials section, screws, rivets, clips and lacing hold the fabric in place on manufactured aircraft. Use the same attach method as used by the original aircraft manufacturer. Deviation requires a field approval.

NOTE: The fuselage and empennage attachments may be used on some aircraft. Follow the methodology for wing rib lacing described below and the manufacturer's instructions for attach point locations and any possible variations to what is presented here.

Care must always be taken to identify and eliminate any sharp edges that might wear through the fabric. Reinforcing tape of the exact same width as the rib cap is installed before any of the fasteners. This approved sticky-back tape helps prevent the fabric from tearing. (***Figure 3-152***) Then, screws, rivets, and clips simply attach into the predrilled holes in the rib caps to hold the fabric to the caps. Rib lacing is a more involved process whereby the fabric is attached to the ribs with cord.

Rib Lacing

There are two kinds of rib lacing cord. One has a round cross section and the other flat. Which to use is a matter of preference based on ease of use and final appearance. Only approved rib lacing cord can be used. Unless a rib is unusually deep from top to bottom, rib lacing uses a single length of cord that passes completely through the wing from the upper surface to the lower surface thereby attaching the top and bottom skin to the rib simultaneously. Holes are laid out and prepunched through the skin as close to the rib caps as possible to accept the lacing cord. (***Figure 3-153***)

This minimizes leverage the fabric could develop while trying to pull away from the structure and prevents tearing. The location of the holes is not arbitrary. The spacing between lacing holes and knots must adhere to manufacturer's instructions, if available. STC lacing guidance refers to manufacturer's instructions or to that shown on the chart in ***Figure 3-154***.

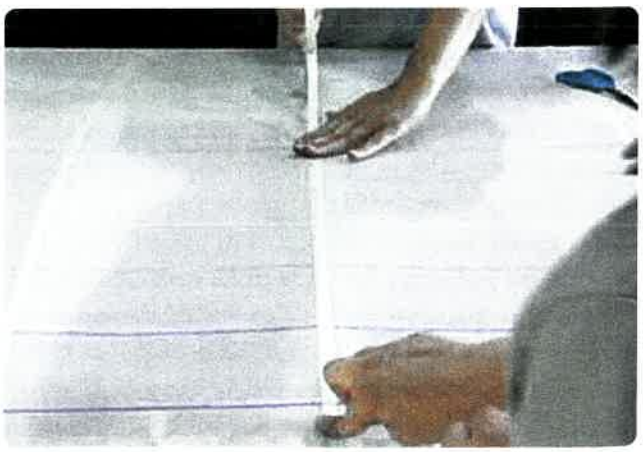


Figure 3-152. Reinforcing tape the same width as the wing ribs is applied over all wing ribs.

Notice that because of greater turbulence in the area of the propeller wash, closer spacing between the lacing is required there. This slipstream is considered to be the width of the propeller plus one additional rib. Ribs are normally laced from the leading edge to the trailing edge of the wing. Rib lacing is done with a long curved needle to guide the cord in and out of holes and through the depth of the rib. The knots are designed not to slip under the forces applied and can be made in a series out of a single strand of lacing. Stitching can begin at the leading edge or trailing edge. A square knot with a half hitch on each side is typically used for the first knot when lacing a rib. (***Figure 3-155***)

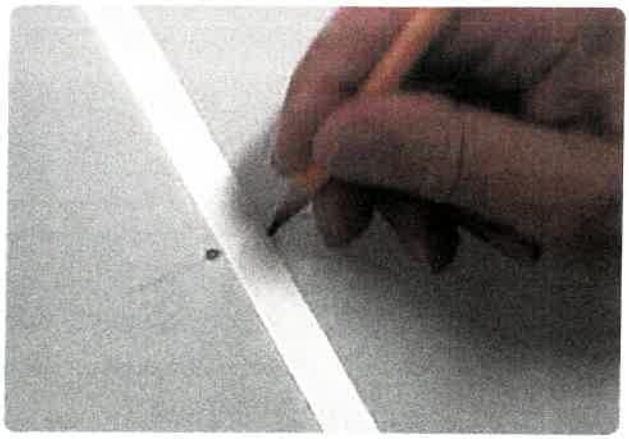


Figure 3-153. A premarked location for a lacing hole, which is punched through the fabric with a pencil.

This is followed by a series of modified seine knots until the final knot is made and secured with a half hitch. (***Figure 3-156***) Hidden modified seine knots are also used. These knots are placed below the fabric surface so only a single strand of lacing is visible across the rib cap. (***Figure 3-157***)

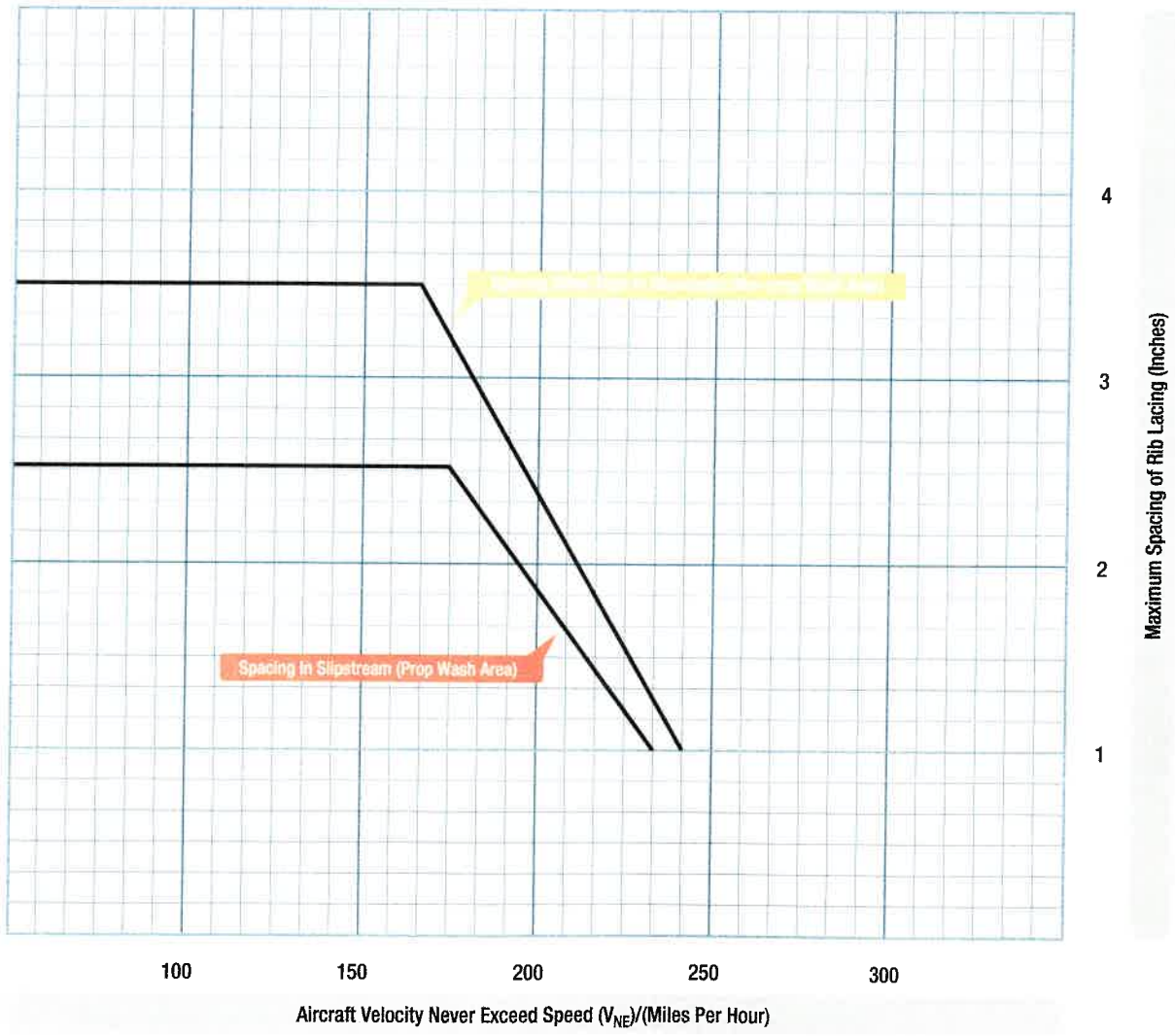


Figure 3-154. A rib lacing spacing chart. Unless manufacturer data specifies otherwise, use the spacing indicated.

Structure and accessories within the wing may prevent a continuous lacing. Ending the lacing and beginning again can avoid these obstacles. Lacing that is not long enough to complete the rib may be ended and a new starting knot can be initiated at the next set of holes. The lacing can also be extended by joining it with another piece of lacing using the splice knot shown in ***Figure 3-158.***

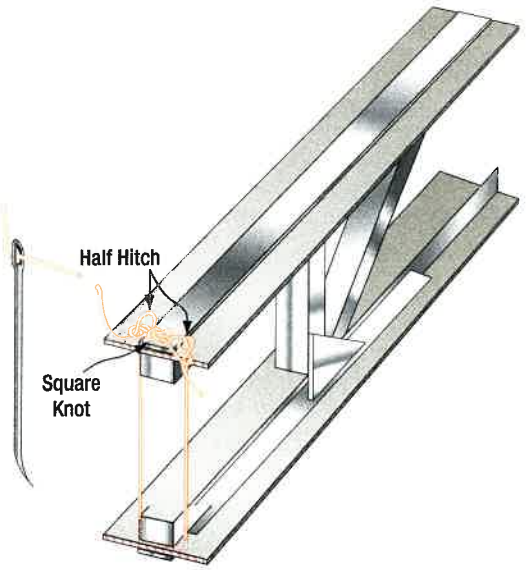


Figure 3-155. A starter knot for rib lacing can be a square knot with a half hitch on each side.

Occasionally, lacing to just the rib cap is employed without lacing entirely through the wing and incorporating the cap on the opposite side. This is done where ribs are exceptionally deep or where through lacing is not possible, such as in an area where a fuel tank is installed. Changing to a needle with a tighter radius facilitates threading the lacing cord in these areas. Knotting procedures remain unchanged.

Technicians inexperienced at rib lacing should seek assistance to ensure the correct knots are being tied. STC holder videos are invaluable in this area. They present repeated closeup visual instruction and guidance to ensure airworthy lacing.

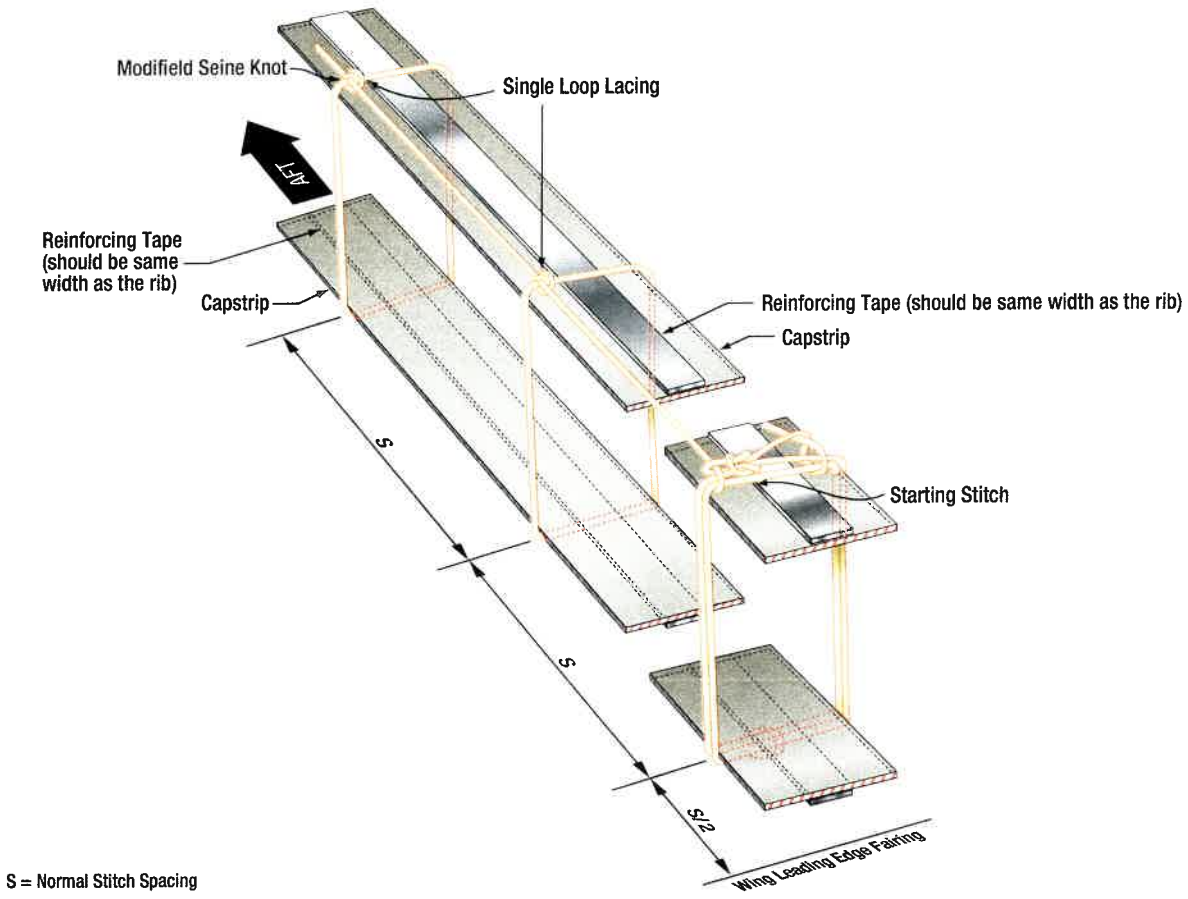


Figure 3-156. In this example of rib lacing, modified seine knots are used and shown above the fabric surface. Hidden modified seine knots are common. They are made so that the knots are pushed or pulled below the fabric surface.

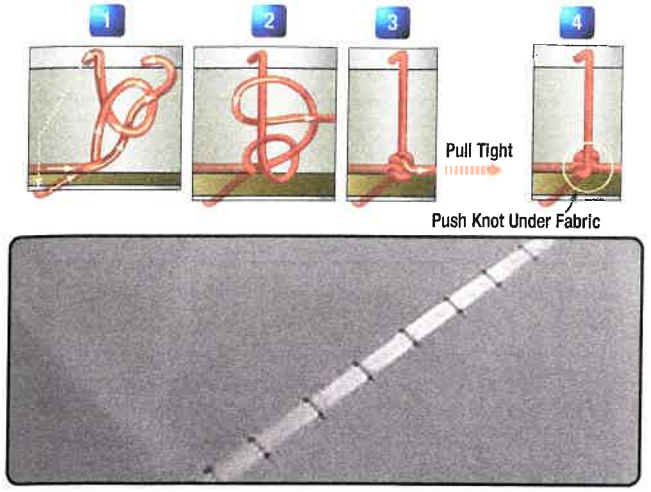


Figure 3-157. Hiding rib lacing knots below the fabric surface results in a smooth surface.

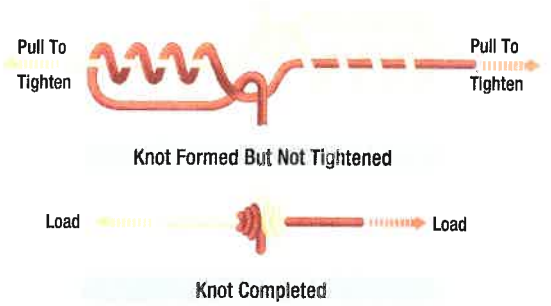


Figure 3-158. The splice knot can be used to join two pieces of rib lacing cord.

RINGS, GROMMETS, AND GUSSETS

When the ribs are laced and the fabric covering completely attached, the various inspection rings, drain grommets, reinforcing patches, and finishing tapes are applied. Inspection rings aid access to critical areas of the structure (pulleys, bell cranks, drag/anti-drag wires, etc.) once the fabric skin is in place. They are plastic or aluminum and normally cemented to the fabric using the approved cement and procedures. The area inside the ring is left intact. It is removed only when inspection or maintenance requires access through that ring. Once removed, preformed inspection panels are used to close the opening. The rings should be positioned as specified by the manufacturer. Lacking that information, they should be positioned as they were on the previous covering fabric. Additional rings should be installed by the technician if it is determined a certain area would benefit from access in the future. (***Figure 3-159***)

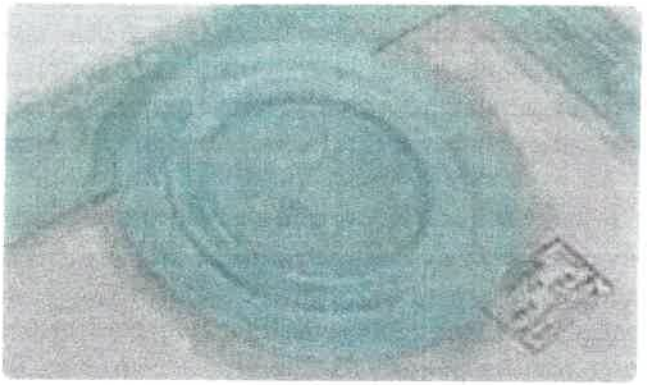


Figure 3-159. Cement is brushed through a four-inch tape during installation over the fabric seam on a wing leading edge. Two-inch tapes cover the wing rib and rib lacing.

Water from rain and condensation can collect under the fabric covering and needs a way to escape. Drain grommets serve this purpose. There are a few different types as described in the materials section above. All are cemented into position in accordance with the approved process under which the work is being performed. Locations for the drain grommets should be ascertained from manufacturer's data. Each fabric covering STC may also give recommendations. Typically, drain grommets are located at the lowest part of each area of the structure (e.g., bottom of the fuselage, wings, empennage). (***Figure 3-160***)

Each rib bay of the wings is usually drained with one or two grommets on the bottom of the trailing edge. Note that drain holes without grommets are sometimes approved in reinforced fabric. It is possible that additional inspection rings and drain grommets have been specified after the manufacture of the aircraft. Check the Airworthiness Directives (ADs) and Service Bulletins for the aircraft being re-covered to ensure required rings and grommets have been installed.

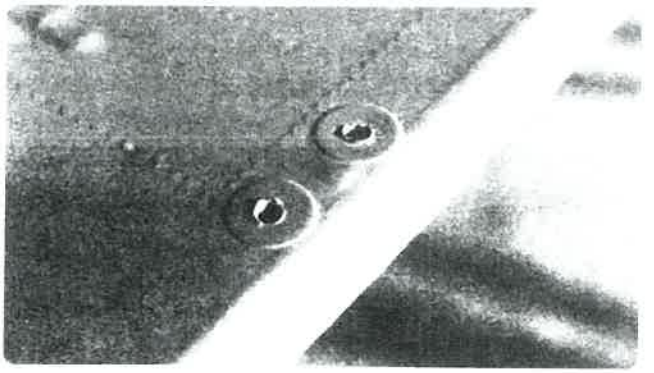


Figure 3-160. Drain grommets cemented into place on the bottom side of a control surface.

Cable guide openings, strut-attach fitting areas, and similar features, as well as any protrusions in the fabric covering, are reinforced with fabric gussets. These are installed as patches in the desired location. They should be cut to fit exactly around the feature they reinforce to support the original opening made in the covering fabric. (***Figure 3-161***) Gussets made to keep protrusions from coming through the fabric should overlap the area they protect. Most processes call for the gusset material to be preshrunk and cemented into place using the approved covering process cementing procedures.

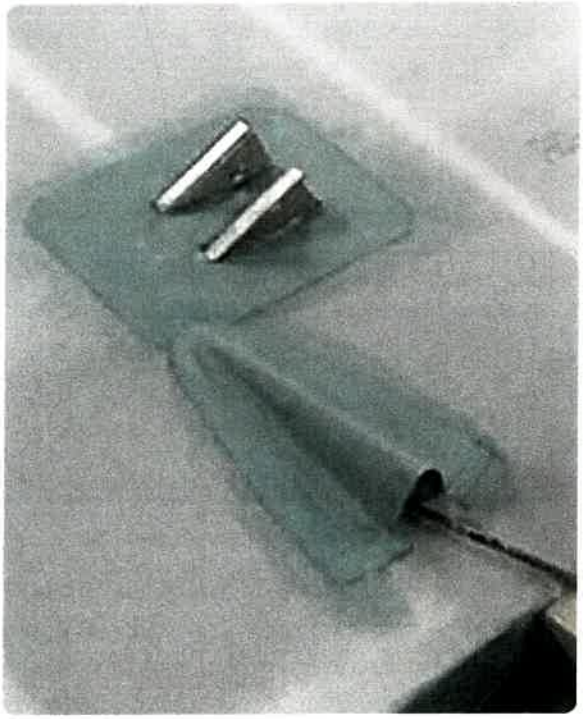


Figure 3-161. A strut fitting and cable guide with reinforcing fabric gussets cemented in place.

FINISHING TAPES

Finishing tapes are applied to all seams, edges, and over the ribs once all of the procedures above have been completed. They are used to protect these areas by providing smooth aerodynamic resistance to abrasion. The tapes are made from the same polyester material as the covering fabric. Use of lighter weight tapes is approved in some STCs. Preshrunk tapes are preferred because they react to exposure to the environment in the same way the as the fabric covering. This minimizes stress on the adhesive joint between the two. Straight edged and pinked tapes are available. The pinking provides greater surface area for adhesion of the edges and a smoother transition into the fabric covering. Only tapes approved in the STC under which work is being accomplished may be used to be considered airworthy.

Finishing tapes from one to six inches in width are used. Typically, two inch tapes cover the rib lacing and fuselage seams. Wing leading edges usually receive the widest tape with four inches being common. (***Figure 3-162***) Bias cut tapes are often used to wrap around the curved surfaces of the airframe, such as the wing tips and empennage surface edges. They lay flat around the curves and do not require notching.

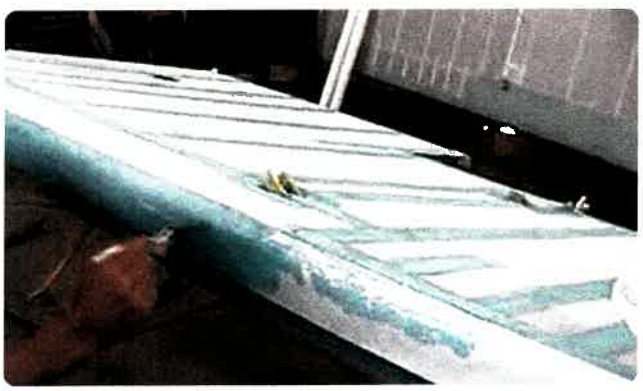


Figure 3-162. Cement is brushed through a four-inch tape during installation over the fabric seam on a wing leading edge. Two-inch tapes cover the wing rib and rib lacing.

Finishing tapes are attached with the process adhesive or the nitrate dope sealer when using a dope-based process. Generally, all chordwise tapes are applied first followed by the span wise tapes at the leading and trailing edges. Follow the manufacturer's STC instructions.

COATING THE FABRIC

The sealer coat in most fabric covering processes is applied after all finishing tapes have been installed unless it was applied prior to rib lacing as in a dope based finishing process. This coat saturates and completely surrounds the fibers in the polyester fabric, forming a barrier that keeps water and contaminants from reaching the fabric during its life. It is also used to provide adhesion of subsequent coatings. Usually brushed on in a cross coat application for thorough penetration, two coats of sealer are commonly used but processes vary on how many coats and whether spray coating is permitted.

With the sealer coats installed and dried, the next step provides protection from UV light, the only significant cause of deterioration of polyester fabric. Designed to prevent UV light from reaching the fabric and extend

the life of the fabric indefinitely, these coating products, or fill coats, contain aluminum solids premixed into them that block the UV rays. They are sprayed on in the number of cross coats as specified in the manufacturer's STC instructions under which work is done. Two to four cross coats is common. Note that some processes may require coats of clear butyrate before the blocking formula is applied.

Fabric primer is a coating used in some approved covering processes that combines the sealer and fill coatings into one. Applied to fabric after the finishing tapes are installed, these fabric primers surround and seal the fabric fibers, provide good adhesion for all of the following coatings, and contain UV blocking agents. One modern primer contains carbon solids and others use chemicals that work similarly to sun block for human skin. Typically, two to four coats of fabric primer are sufficient before the top coatings of the final finish are applied. (***Figure 3-163***)

The certificated mechanic must strictly adhere to all instructions for thinning, drying times, sanding, and cleaning. Small differences in the various processes exist and what works in one process may not be acceptable and could ruin the finish of another process. STCs are issued on the basis of the holder having successfully proven the effectiveness of both the materials and the techniques involved. When the fill coats have been applied, the final appearance of the fabric covering job is crafted with the application of various topcoats. Due to the chemical nature of the fill coating upon which topcoats are sprayed, only specified materials can be used for top coating to ensure compatibility. Colored butyrate dope and polyurethane paint finishes are most common. They are sprayed on according to instructions.



Figure 3-163. Applying a primer with UV blocking by spraying cross coats.

Once the topcoats are dry, the trim (ID numbers, stripes, etc.) can be added. Strict observation of drying times and instructions for buffing and waxing are critical to the quality of the final finish. Also, note that STC instructions may include insight on finishing the nonfabric portions of the airframe to best match the fabric covering finish.

POLYESTER FABRIC REPAIRS

Applicable Instructions

Repairs to aircraft fabric coverings are inevitable. Always inspect a damaged area to ensure the damage is confined to the fabric and does not involve the structure below. A technician who needs to make a fabric repair must first identify which approved data was used to install the covering that needs to be repaired. Consult the logbook where an entry and reference to manufacturer data, an STC, or a field approval should be recorded. The source of approved data for the covering job is the same source of approved data used for a repair.

This section discusses general information concerning repairs to polyester fabric. It is the responsibility of the holder of an STC to provide maintenance instructions for the STC alteration in addition to materials specifications required to do the job.

Repair Considerations

The type of repair performed depends on the extent of the damage and the process under which the fabric was installed. The size of the damaged area is often a reference for whether a patch is sufficient to do the repair or whether a new panel should be installed. Repair size may also dictate the amount of fabric to fabric overlap required when patching and whether finishing tapes are required over the patch. Many STC repair procedures do not require finishing tapes. Some repairs require the use of tape up to six inches wide.

While many cotton fabric repairs involve sewing, nearly all repairs of polyester fabric are made without sewing. It is possible to apply the sewing repair techniques to polyester fabric, but they were developed primarily for

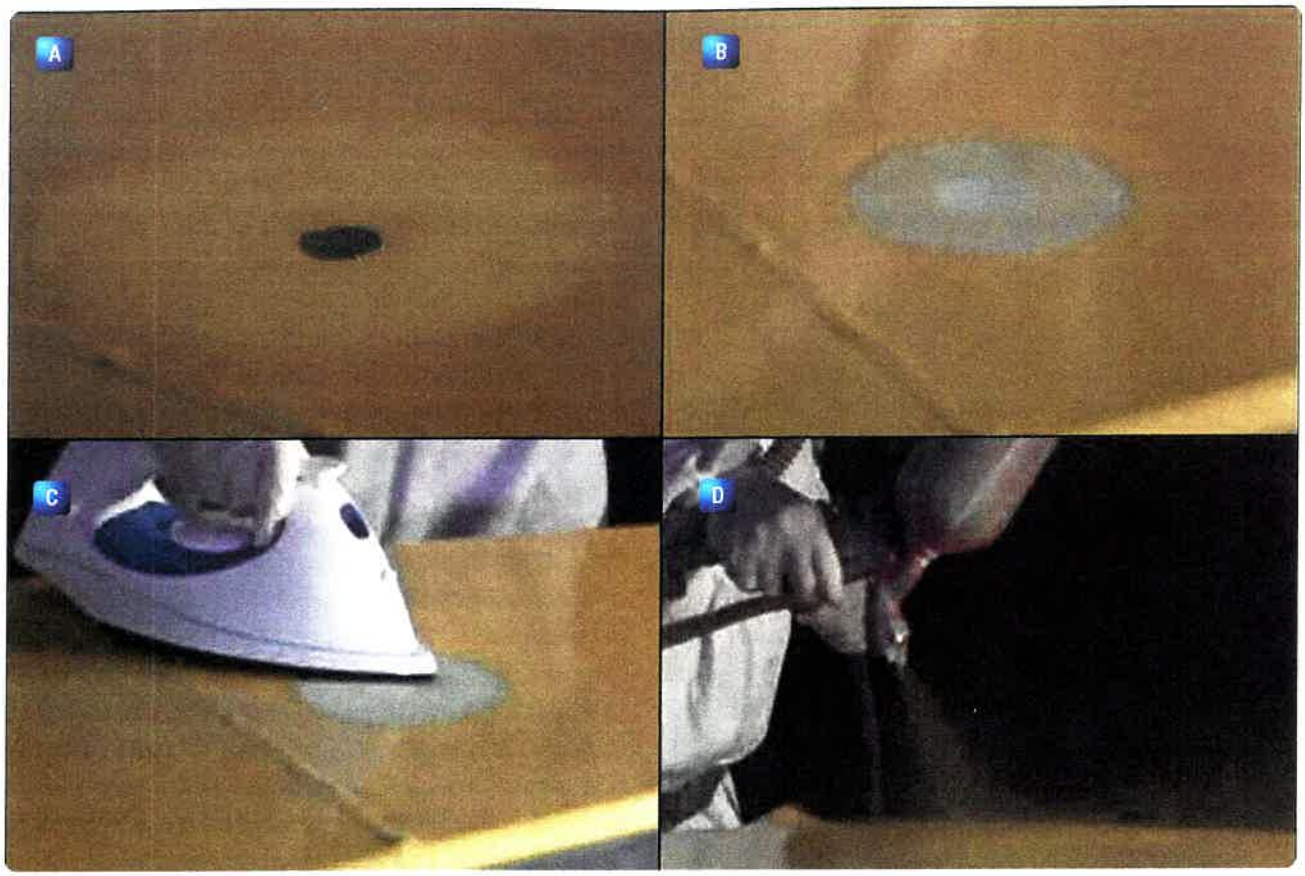


Figure 3-164. A patch over this small hole on a polyurethane top coat is repaired in accordance with the repair instructions in the STC under which the aircraft was recovered. It requires only a two inch fabric overlap and scuffing into the top coat before cementing and refinishing. Other STC repair instructions may not allow this repair.

cotton and linen fabrics. STC instructions for repairs to polyester fabric are for cemented repairs which most technicians prefer as they are generally considered easier than sewn repairs. There is no compromise to the strength of the fabric with either method.

Patching or replacing a section of the covering requires prepping the fabric area around the damage where new fabric is to be attached. Procedures vary widely. Dope based covering systems tend toward stripping off all coatings to cement raw fabric to raw fabric when patching or seaming in a new panel. From this point, the coatings are reapplied and finished as in the original covering process. Some polyurethane based coating processes require only a scuffing of the topcoat with sandpaper before adhering small patches that are then refinished. (***Figure 3-164***)

Still, other processes may remove the topcoats and cement a patch into the sealer or UV blocking coating. In some repair processes, preshrunk fabric is used and in others, the fabric is shrunk after it is in place. Varying techniques and temperatures for shrinking and gluing the fabric into a repair also exist. These deviations in procedures underscore the critical nature of identifying and strictly adhering to the correct instructions from the approved data for the fabric covering in need of repair. A patch or panel replacement technique for one covering system could easily create an unairworthy repair if used on fabric installed with a different covering process.

Large section panel repairs use the same proprietary adhesives and techniques and are only found in the instructions for the process used to install the fabric covering. A common technique for replacing any large damaged area is to replace all of the fabric between two adjacent structural members (e.g., two ribs, two longerons, between the forward and rear spars).

COTTON-COVERED AIRCRAFT

You may encounter a cotton fabric covered aircraft. In addition to other airworthiness criterion, the condition of the fabric under the finished surface is paramount as the cotton can deteriorate even while the aircraft is

stored in a hanger. Inspection, in accordance with the manufacturer maintenance manual should be diligent. If the cotton covering is found to be airworthy, repairs to the fabric can be made under those specifications. This includes sewn in and doped in patches, as well as sewn in and doped in panel repairs. Due to the very limited number of airworthy aircraft that may still be covered with cotton, this handbook does not cover specific information on recovering with cotton or cotton fabric maintenance and repair procedures.

FIBERGLASS COVERINGS

References to fiberglass surfaces in aircraft covering STCs, and other maintenance literature address techniques for finishing and maintaining this kind of surface. However, this is typically limited to fiberglass ray domes and fiberglass reinforced plywood surfaces and parts that are still in service. Use of dope based processes on fiberglass is well established. Repair and apply coatings and finishes on fiberglass in accordance with manufacturer data or STC instructions.

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Question: 3-1
Name two primary classifications of transparent plastics.

Question: 3-2
Name the three main advantages of the use of advanced composite materials on aircraft.

Question: 3-3
Light weight, strong and tough _____ fibers are used in components in areas prone to impact.

Question: 3-4
The surface of an external composite component often consists of a ply or layer of conductive material for _____ because composite materials are less conductive than aluminum.

Question: 3-5
Thermosetting resins use a _____ to cure.

Question: 3-6
_____ has high bending stiffness at minimal weight in comparison to aluminum and composite laminate construction.

Question: 3-7
Since composite structures are affected by ultra violet light (UV) they are typically protected by _____.

Question: 3-8
In ultrasonic testing of composites, disrupted or diminished sonic energy is picked up by a _____ and converted into a display on an oscilloscope or a chart recorder.

ANSWERS

Answer: 3-1
Thermoplastic.
Thermosetting.

Answer: 3-2
Light weight.
High strength
Corrosion resistance.

Answer: 3-3
aramid

Answer: 3-4
lightning strike protection

Answer: 3-5
chemical reaction

Answer: 3-6
Honeycomb sandwich construction

Answer: 3-7
a top coating (paint).

Answer: 3-8
receiving transducer

Question: 3-9
Name three types of NDI that are used to detect water inside a composite structure.

Question: 3-10
An _____ controls the pressure, temperature and time profiles of a part that is being cured.

Question: 3-11
What can happen when a composite repair is not cured at the proper temperature?

Question: 3-12
_____ resins are preferred for vacuum bag molding.

Question: 3-13
Excessive polishing in one location of thermoplastics can cause _____.

Question: 3-14
Glue deterioration on wood aircraft structure can be caused by chemical factors, mechanical forces, or _____ growth.

Question: 3-15
The purpose of repairing all wood structural components is to obtain a structure as strong as _____.

Question: 3-16
When mixing, the recommended proportions of glue, catalyst, and water or other solvent should be determined by _____ of each component.

ANSWERS

Answer: 3-9
X-ray.
Thermography.
RF moisture meter.

Answer: 3-10
autoclave

Answer: 3-11
weak patches and/or bonding surfaces which can result in failure during service.

Answer: 3-12
High-flow

Answer: 3-13
visual distortion (due to high temperature softening the surface).

Answer: 3-14
fungal

Answer: 3-15
the original.

Answer: 3-16
the weight

Question: 3-17
The slope of the scarf in any stressed part, such as a spar, should not be steeper than _____ to 1.

Question: 3-18
All holes drilled in a wood structure to receive bolts or bushings should be of such size that inserting the bolt or bushing requires _____.

Question: 3-19
Holes drilled in wood structure should be _____ after being drilled.

Question: 3-20
The surface finish on a wood structure or repair is the final line of defense to prevent the destructive effects of _____ entry into the structure.

Question: 3-21
A cut, fold or seam made diagonally to the warp or fill threads is called _____.

Question: 3-22
Modern aircraft fabric covering systems utilize special _____ to attach the fabric to the airframe.

Question: 3-23
Fabric is considered to be airworthy until it deteriorates to a breaking strength less than _____ percent of the strength of the new fabric required for the aircraft.

Question: 3-24
What are the primary concerns for fabric seams?

ANSWERS

Answer: 3-17
15

Answer: 3-18
a light tapping with a wood or rawhide mallet.

Answer: 3-19
sealed

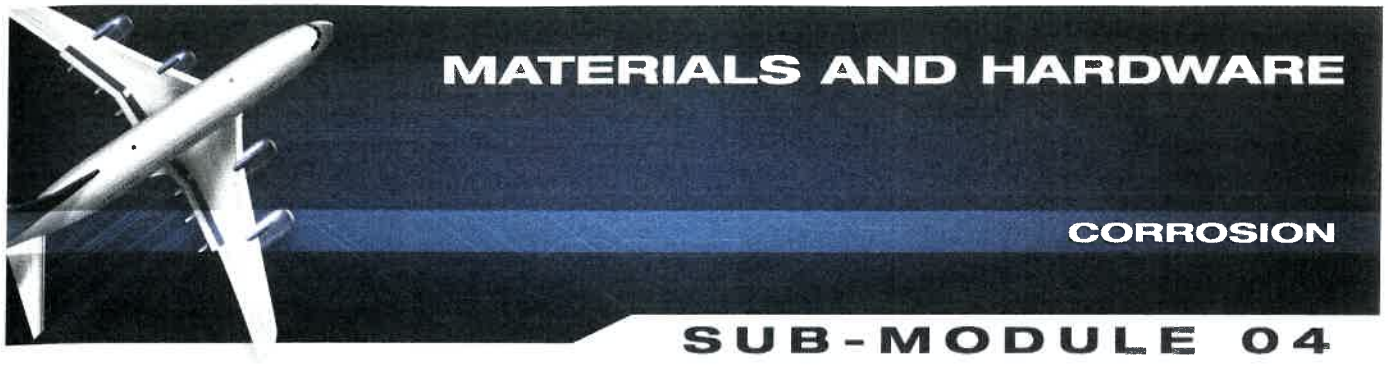
Answer: 3-20
moisture

Answer: 3-21
the bias.

Answer: 3-22
fabric cement

Answer: 3-23
70

Answer: 3-24
Strength.
Elasticity.
Durability.
Good Appearance.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY → B1

Sub-Module 04

CORROSION

Knowledge Requirements

4.1 - Corrosion

(a)	Chemical fundamentals; Formation by, galvanic action process, microbiological, stress;	1
(b)	Types of corrosion and their identification; Causes of corrosion; Material types, susceptibility to corrosion.	3

6.4(A) - CHEMICAL FUNDAMENTALS OF CORROSION

INTRODUCTION TO CORROSION

Corrosion is the electrochemical deterioration of a metal because of its chemical reaction with the surrounding environment. While new and better materials are continuously being developed, the progress is offset, in part, by a more aggressive operational environment. This problem is compounded by the fact that corrosion is a complex phenomenon. It can take many different forms and the resistance of aircraft materials to corrosion drastically change with only a small environmental change.

Corrosion is most often thought of as a slow process of material deterioration, taking place over a significant period of time (examples being general corrosion, pitting, exfoliation, etc.). Other forms of corrosion degradation can occur very quickly, in days or even hours, with catastrophic results. These forms (such a stress corrosion cracking, environmental embrittlement, and corrosion fatigue) depend on both the chemical and mechanical aspects of the environment and cause catastrophic structural failure without warning.

The possibility of an in flight mishap or excessive down times for structural repairs necessitates an active corrosion prevention and control program. The type and aggressiveness of the corrosion prevention and control program depends on the operational environment of the aircraft. Aircraft exposed to salt air, heavy atmospheric industrial pollution, and/or over water operations will require a more stringent corrosion prevention and control program than an aircraft that is operated in a dry environment.

COMMON CORROSIVE AGENTS

Substances that cause corrosion of metals are called corrosive agents. The most common corrosive agents are acids, alkalies, and salts. The atmosphere and water, the two most common media for these agents, may also act as corrosive agents.

- Acids—moderately strong acids severely corrode most of the alloys used in airframes. The most destructive are sulfuric acid (battery acid), halogen acids (hydrochloric, hydrofluoric, and hydrobromic), nitrous oxide compounds, and organic acids found in the wastes of humans and animals.
- Alkalies—as a group, alkalies are not as corrosive as acids. Aluminum and magnesium alloys are exceedingly prone to corrosive attack by many alkaline solutions unless the solutions contain a corrosion inhibitor. Substances particularly corrosive to aluminum are washing soda, potash (wood ashes), and lime (cement dust). Ammonia, an alkali, is an exception because aluminum alloys are highly resistant to it.
- Salts—most salt solutions are good electrolytes and can promote corrosive attack. Some stainless steel alloys are resistant to attack by salt solutions but aluminum alloy, magnesium alloys, and other steels are extremely vulnerable. Exposure of airframe materials to salts or their solutions is extremely undesirable.
- Atmosphere—the major atmospheric corrosive agents are oxygen and airborne moisture. Corrosion often results from the direct action of atmospheric oxygen and moisture on metal, and the presence of additional moisture often accelerates corrosive attack, particularly on ferrous alloys. However, the atmosphere may also contain other corrosive gases and contaminants, particularly industrial and marine salt spray.
- Water—the corrosiveness of water depends on the type and quantity of dissolved mineral and organic impurities and dissolved gasses (particularly oxygen) in the water. One characteristic of water that determines its corrosiveness is the conductivity or ability to act as an electrolyte and conduct a current. Physical factors, such as water temperature and velocity, also have a direct bearing on its corrosiveness.

CORROSION FORMATION

GALVANIC ACTION

Corrosion is a natural occurrence that attacks metal by chemical or electrochemical action and converts it back to a non-metallic compound. Four conditions must exist before electrochemical corrosion can occur. (***Figure 4-1***)

They are: (1) A metal subject to corrosion (Anode); (2) A dissimilar conductive material (Cathode), which has less tendency to corrode; (3) Presence of a continuous, conductive liquid path (Electrolyte); and (4) Electrical

Simplified corrosion cell showing conditions that must exist for electrochemical corrosion.

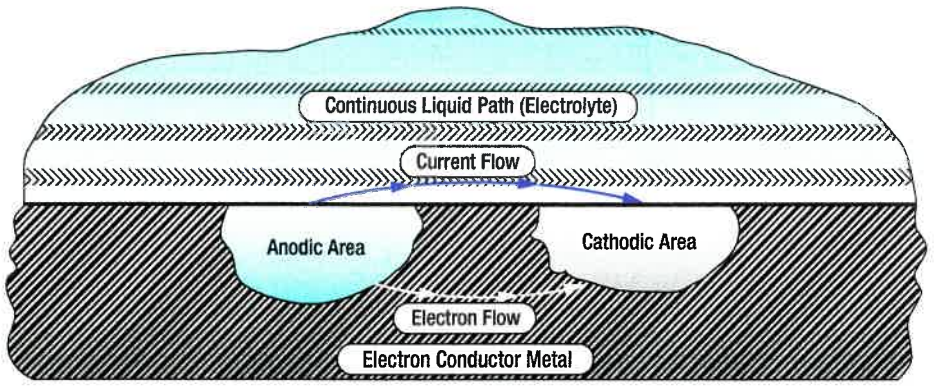


Figure 4-1. Simplified corrosion cell showing conditions which must exist for electrochemical corrosion.

Elimination of corrosion by application of an organic Film to metal surface.

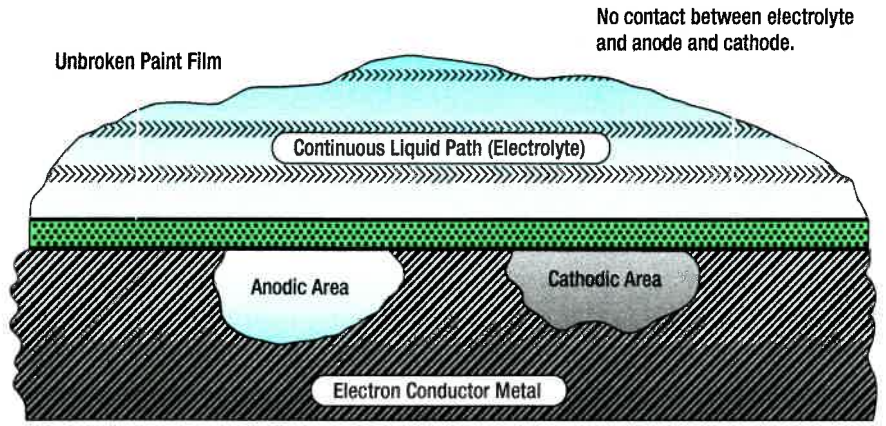


Figure 4-2. Elimination of corrosion by application of an organic film to metal surface.

contact between the anode and the cathode, usually in the form of metal to metal contact such as rivets, bolts, and corrosion. Elimination of any one of these conditions will stop electrochemical corrosion. (***Figure 4-2***)

Note: Paint can mask the initial stages of corrosion. Since corrosion products occupy more volume than the original metal, painted surfaces should be inspected often for irregularities such as blisters, flakes, chips, and lumps. Some factors that influence metal corrosion and the rate of corrosion are:

1. Type of metal;
2. Heat treatment and grain direction;
3. Presence of a dissimilar, less corrodible metal;
4. Anodic and cathodic surface areas (in galvanic corrosion);
5. Temperature;
6. Presence of electrolytes (hard water, salt water, battery fluids, etc.);
7. Availability of oxygen;
8. Presence of biological organisms;
9. Mechanical stress on the corroding metal;
10. Time of exposure to a corrosive environment;
11. Lead/graphite pencil marks on aircraft surface metals.

Most pure metals are not suitable for aircraft construction and are used only in combination with other metals to form alloys. Most alloys are made up entirely of small crystalline regions, called grains. Corrosion can occur on surfaces of those regions which are less resistant and also at boundaries between regions, resulting in the formation of pits and intergranular corrosion. Metals have a wide range of corrosion resistance. The most active metals, (those which lose electrons easily), such as magnesium and aluminum, corrode easily. The most noble metals (those which do not lose electrons easily), such as gold and silver, do not corrode easily.

Corrosion is quickened by high temperature environments that accelerate chemical reactions and increase the concentration of water vapor in the air. Electrolytes (electrically conducting solutions) form on surfaces when condensation, salt spray, rain, or rinse water accumulate. Dirt, salt, acidic gases, and engine exhaust gases can dissolve on wet surfaces, increasing the electrical conductivity of the electrolyte, thereby increasing the rate of corrosion. When some of the electrolyte on a metal surface is partially confined, (such as between surfaces that have been joined together (faying surfaces) or in a deep crevice) the metal around this area corrodes more rapidly. This type of corrosion is called an oxygen concentration cell. Corrosion occurs more rapidly because the reduced oxygen content of the confined electrolyte causes the adjacent metal to become anodic to other metal surfaces on the same part that are immersed in electrolyte or exposed to air.

Slime, molds, fungi, and other living organisms (some microscopic) can grow on damp surfaces. Once they are established, the area usually remains damp, increasing the possibility of corrosion. Manufacturing processes such as machining, forming, welding, or heat treatment can leave residual stress in aircraft parts and can cause cracking in a corrosive environment.

MICROBIOLOGICAL CORROSION

Bacteria may be either aerobic or anaerobic. Aerobic bacteria require oxygen to live. They accelerate corrosion by oxidizing sulfur to produce sulfuric acid. Bacteria living adjacent to metals may promote corrosion by depleting the oxygen supply or by releasing metabolic products. Anaerobic bacteria, on the other hand, can survive only when free oxygen is not present. The metabolism of these bacteria requires them to obtain part of their sustenance by oxidizing inorganic compounds, such as iron, sulfur, hydrogen, and carbon monoxide. The resultant chemical reactions cause corrosion.

Fungi are the microorganisms that feed on organic materials. While low humidity does not kill microbes, it slows their growth and may prevent corrosion damage. Ideal growth conditions for most microorganisms are temperatures from 20-40°C and relative humidity from 85 to 100 percent.

Damage resulting from microbial growth can occur when any of three basic mechanisms, or a combination of these, are brought into play. First, fungi have a tendency to hold moisture which contributes to other forms of corrosion. Second, because fungi are living organisms, they need food to survive. This food is obtained from the material on which the fungi are growing. Third, these micro organisms secrete corrosive fluids that attack many materials, including some that are not fungi nutrient.

Microbial growth must be removed completely to avoid corrosion. Microbial growth should be removed by hand with a firm non-metallic bristle brush and water. Removal of microbial growth is easier if the growth is kept wet with water. Microbial growth may also be removed with steam at 100 psi. Protective clothing must be used when using steam for removing microbial growth. (***Figure 4-3***)

STRESS CORROSION

Stress corrosion is the growth of crack formations in a corrosive environment. It can lead to unexpected sudden failure of normally ductile metals which have been subjected to stress, especially at elevated temperatures. The chemical environment that causes stress corrosion cracking is often one which is only mildly corrosive to the metal. Hence, metal parts with severe stress corrosion can appear bright and shiny while being filled with microscopic cracks. This factor makes it common for stress corrosion to go undetected prior to failure. Stress corrosion often progresses rapidly, and is more common among alloys than pure metals. The specific environment is of crucial importance and only small concentrations of certain chemicals are needed to produce catastrophic results and unexpected failure. See more on this topic later in this chapter.



Figure 4-3. An example of severe microbial corrosion.

6.4(B) - TYPES OF CORROSION AND THEIR IDENTIFICATION

Alloy	Type of attack to which alloy is susceptible	Appearance of corrosion product
Magnesium	Highly susceptible to pitting	White, powdery, snow-like mounds and white spots on surface
Low alloy steel (4,000–8,000 series)	Surface oxidation and pitting, surface, and intergranular	Reddish–brown oxide (rust)
Aluminum	Surface pitting, intergranular, exfoliation stress–corrosion and fatigue cracking, and fretting	White–to–grey powder
Titanium	Highly corrosion resistant; extended or repeated contact with chlorinated solvents may result in degradation of the metal's structural properties at high temperature	No visible corrosion products at low temperature. Colored surface oxides develop above 700 °F (370 °C)
Cadmium	Uniform surface corrosion; used as sacrificial plating to protect steel	From white powdery deposit to brown or black mottling of the surface
Stainless steels (300–400 series)	Crevice corrosion; some pitting in marine environments; corrosion cracking; intergranular corrosion (300 series); surface corrosion (400 series)	Rough surface; sometimes a uniform red, brown, stain
Nickel–base (Inconel, Monel)	Generally has good corrosion resistant qualities; susceptible to pitting in sea water	Green powdery deposit
Copper–base Brass, Bronze	Surface and intergranular corrosion	Blue or blue–green powdery deposit
Chromium (Plate)	Pitting (promotes rusting of steel where pits occur in plating)	No visible corrosion products; blistering of plating due to rusting and lifting
Silver	Will tarnish in the presence of sulfur	Brown–to–black film
Gold	Highly corrosion resistant	Deposits cause darkening of reflective surfaces
Tin	Subject to whisker growth	Whisker–like deposit

Figure 4-4. Corrosion of metals.

All corrosive attacks begin on the surface of the metal making the classification of corrosion by physical appearance a convenient means of identification. (***Figure 4-4***)

GENERAL SURFACE CORROSION

General surface corrosion (also referred to as Uniform Etch or Uniform Attack Corrosion) is the most common form of corrosion and results from a direct chemical attack on a metal surface and involves only the metal surface. (***Figure 4-5***)

General surface corrosion usually occurs over a wide area and is more or less equal in dispersion. On a polished surface, this type of corrosion is first seen as a general dulling of the surface, and if allowed to continue, the surface becomes rough and possibly frosted in appearance. The discoloration or general dulling of metal created by exposure to elevated temperatures is not to be considered general surface corrosion.

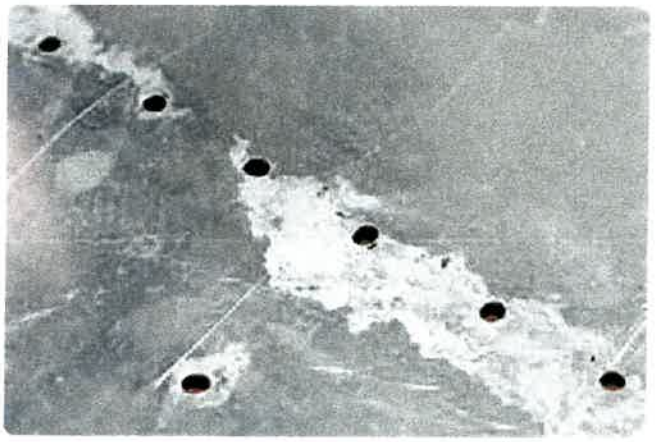


Figure 4-5. Surface corrosion.

Sometimes corrosion spreads under the surface coating and can not be recognized by either the roughening of the surface or a frosted appearance. Instead closer inspection reveals the paint or plating is lifted off the surface in small blisters that result from the pressure of the underlying accumulation of corrosion products.

PITTING CORROSION

Pitting corrosion is one of the most destructive and intense forms of corrosion. It can occur in any metal but is most common on metals that form protective oxide films, such as aluminum and magnesium alloys. It is first noticeable as a white or gray powdery deposit, similar to dust, which blotches the surface. When the deposit is cleaned away, tiny holes or pits can be seen in the surface. (***Figures 4-6***)

These small surface openings may penetrate deeply into structural members and cause damage completely out of proportion to its surface appearance.

CONCENTRATION CELL CORROSION

Concentration cell corrosion, (also known as crevice corrosion) is corrosion of metals in a metal to metal joint, corrosion at the edge of a joint even though the joined metals are identical, or corrosion of a spot on the metal surface covered by a foreign material. Metal ion concentration cells,oxygen concentration cells, and active passive cells are three general types of concentration cell corrosion.

Metal Ion Concentration Cells

The solution may consist of water and ions of the metal which is in contact with water. A high concentration of the metal ions will normally exist under faying surfaces where the solution is stagnant, and a low concentration of metal ions will exist adjacent to the crevice which is created by the faying surface. An electrical potential will exist between the two points. The area of the metal in contact with the low concentration of metal ions will be anodic and corrode, and the area in contact with the high metal ion concentration will be cathodic and not show signs of corrosion. (***Figure 4-7***)

Oxygen Concentration Cells

The solution in contact with the metal surface will normally contain dissolved oxygen. An oxygen cell can develop at any point where the oxygen in the air is not allowed to diffuse into the solution, thereby creating a difference in oxygen concentration between two points. Typical locations of oxygen concentration cells are under gaskets, wood, rubber, and other materials in contact with the metal surface. Corrosion will occur at the area of low oxygen concentration (anode). Alloys such as stainless steel are particularly susceptible to this type of crevice corrosion. (***Figure 4-8***)

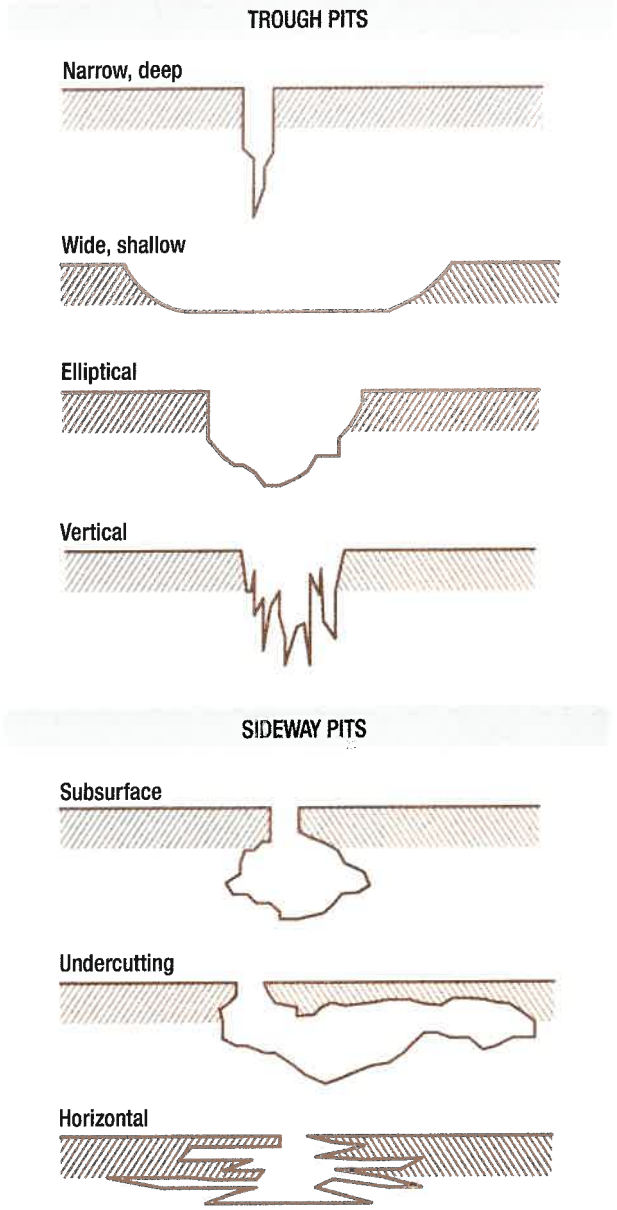


Figure 4-6. Types of pitting corrosion.

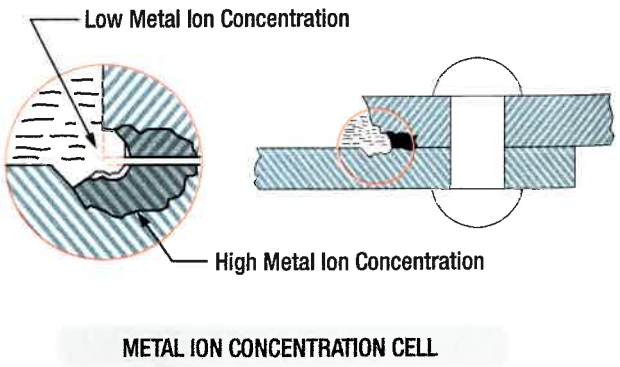
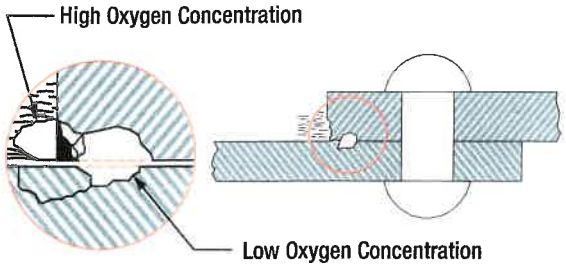


Figure 4-7. Metal ion concentration cell.

Active Passive Cells

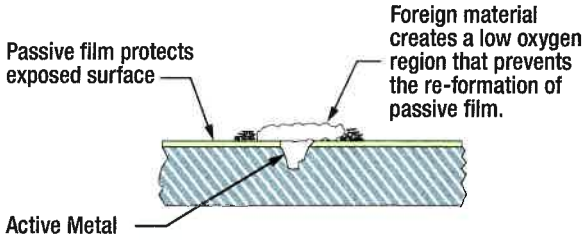
Metals which depend on a tightly adhering passive film, usually an oxide, for corrosion protection are prone to rapid corrosive attack by active passive cells. Active passive cells are often referred to as a type of concentration cell corrosion. However, the active passive cell is actually two forms of corrosion working in conjunction. The corrosive action usually starts as an oxygen concentration cell. As an example, salt deposits on the metal surface in the presence of water containing oxygen can create the oxygen cell. The passive film will be broken beneath the salt crystals. Once the passive film is broken, the active metal beneath the film will be exposed to corrosive attack. (***Figure 4-9***)

Rapid pitting of the active metal will result. This reaction can become locally intense due to several factors. First the reaction is augmented by the affected area, since the proportion of the exposed base metal is small compared to the surrounding nonreactive metal. This effectively concentrates the focal point of the reaction, often resulting in deep pits in a short time and a greater rate of corrosion.



OXYGEN CONCENTRATION CELL

Figure 4-8. Oxygen concentration cell.



ACTIVE – PASSIVE CELL

Figure 4-9. Active-passive cell.

FILIFORM CORROSION

Filiform corrosion is a special form of oxygen concentration cell which occurs on metal surfaces having an organic coating system. It is recognized by its characteristic worm like trace of corrosion products beneath the paint film. (***Figure 4-10***)

Polyurethane finishes are especially susceptible to filiform corrosion. Filiform occurs when the relative humidity of the air is between 78 to 90 percent and the surface is slightly acidic. (***Figure 4-11***) This corrosion usually attacks steel and aluminum surfaces. The traces never cross on steel, but they will on aluminum which makes the damage deeper and more severe for aluminum. If the corrosion is not removed, the area treated, and a protective finish applied; the corrosion can lead to intergranular corrosion, especially around fasteners and seams. Filiform corrosion can be removed using glass bead blasting material with abrasive blasting equipment or sanding. Filiform can be prevented by storing aircraft in an environment with a relative humidity below 70 percent, using coating systems having a low rate of diffusion for oxygen and water vapors, and by washing the aircraft to remove acidic contaminants from the surface.

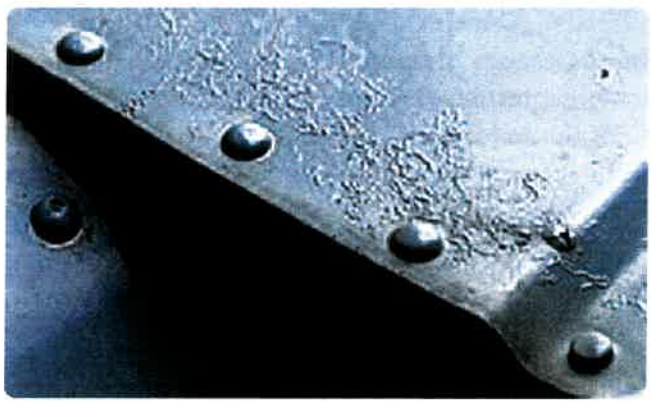


Figure 4-10. Filiform corrosion.

INTERGRANULAR CORROSION

Intergranular corrosion is an attack on the grain boundaries of a metal. A highly magnified cross section of any commercial alloy shows the granular structure of the metal. It consists of quantities of individual grains, and each of these tiny grains has a clearly defined boundary which chemically differs from the metal within the grain. The grain boundary and the grain center can react with each other as anode and cathode when in contact with an electrolyte. (***Figure 4-12***)

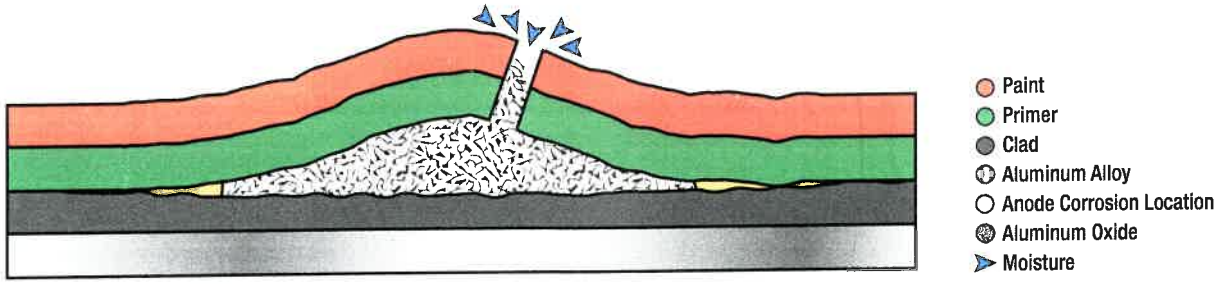


Figure 4-11. Filiform corrosion.

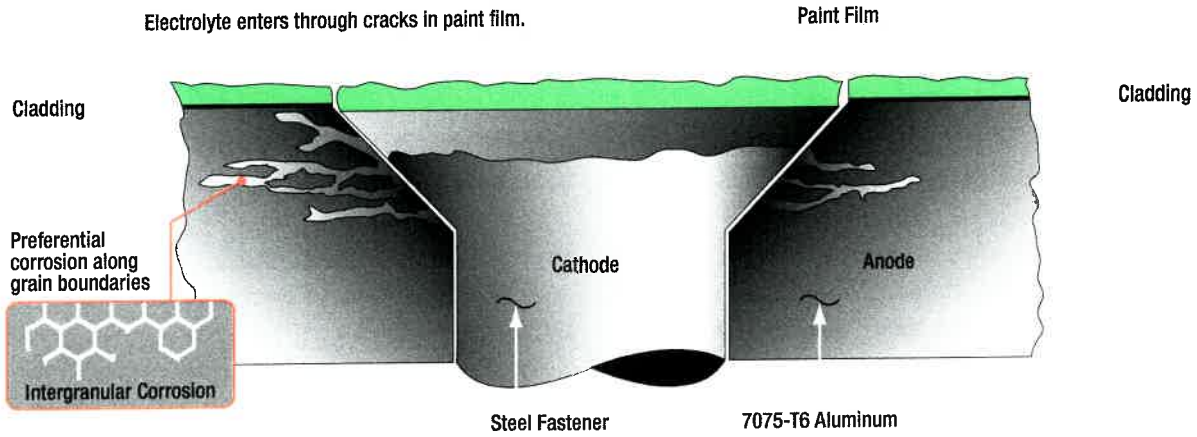


Figure 4-12. Intergranular corrosion of 7075-T6 aluminum adjacent to steel fastener.

Rapid selective corrosion of the grain boundaries can occur. High strength aluminum alloys such as 2014 and 7075 are more susceptible to intergranular corrosion if they have been improperly heat treated and then exposed to a corrosive environment. (***Figure 4-13***)

EXFOLIATION CORROSION

Exfoliation corrosion is an advanced form of intergranular corrosion and shows itself by lifting up the surface grains of a metal by the force of expanding corrosion products occurring at the grain boundaries just below the surface. (***Figure 4-14***) It is visible evidence of intergranular corrosion and is most often seen on extruded sections where grain thickness are usually less than in rolled forms.

This type of corrosion is difficult to detect in its initial state. Extruded components such as spars can be subject to this type of corrosion. Ultrasonic and eddy current inspection methods are being used to detect early signs of exfoliation with a great deal of success.

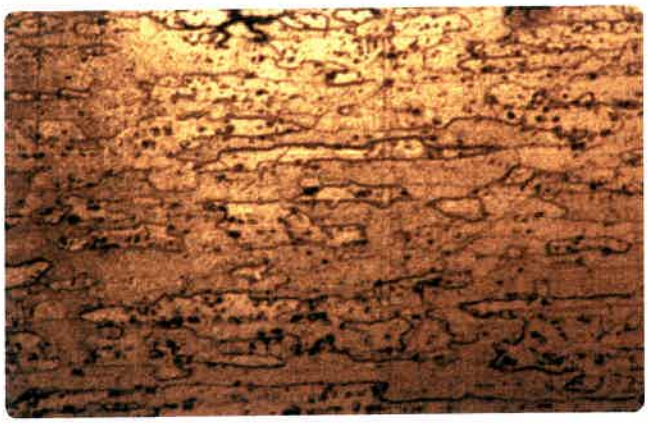


Figure 4-13. Intergranular corrosion present on 6061-T6 aluminum.



Figure 4-14. Exfoliation corrosion.

GALVANIC CORROSION

Galvanic corrosion occurs when two dissimilar metals make contact in the presence of an electrolyte. (***Figure 4-15***) The rate which corrosion occurs depends on the difference in activities. The greater the difference in activity, the faster the corrosion occurs. The rate of galvanic corrosion also depends on the size of the parts in contact. If the surface area of the corroding metal is smaller than the surface area of the less active metal, corrosion is rapid and severe. When the corroding metal is larger than the less active metal, corrosion is slow and superficial.



Figure 4-15. Galvanic corrosion.

STRESS CORROSION CRACKING

This form of corrosion involves a constant or cyclic stress acting in conjunction with a damaging chemical environment. The stress may be caused by internal or external loading. (***Figure 4-16***)

Internal stress may be trapped in a part of structure during manufacturing processes, such as cold working or by unequal cooling from high temperatures. Most manufacturers follow these processes with a stress relief operation. Even so, sometimes stress remains trapped. The stress may be externally introduced in part structure by riveting, welding, bolting, clamping, press fit, etc. If a slight mismatch occurs or a fastener is over torqued, internal stress is present. Internal stress is more important than design stress, because stress corrosion is difficult to recognize before it has overcome the design safety factor. The level of stress varies from point to point within the metal. Stresses near the yield strength are generally necessary to promote stress corrosion cracking. However, failures may occur at lower stresses.

Specific environments have been identified that cause stress corrosion cracking of certain alloys.

1. Salt solutions and sea water cause stress corrosion cracking of high strength, heat treated steel and aluminum alloys.
2. Methyl alcohol hydrochloric acid solutions cause stress corrosion cracking of some titanium alloys.
3. Magnesium alloys may stress corrode in moist air.

Stress corrosion may be reduced by applying protective coatings, stress relief heat treatments, using corrosion inhibitors, or controlling the environment. Shot peening a metal surface increases resistance to stress corrosion cracking by creating compressive stresses on the surface which should be overcome by applied tensile stress before the surface sees any tension load. Therefore, the threshold stress level is increased.

FATIGUE CORROSION

Fatigue corrosion involves cyclic stress and a corrosive environment. Metals may withstand cyclic stress for an infinite number of cycles so long as the stress is below the endurance limit of the metal. Once the limit has

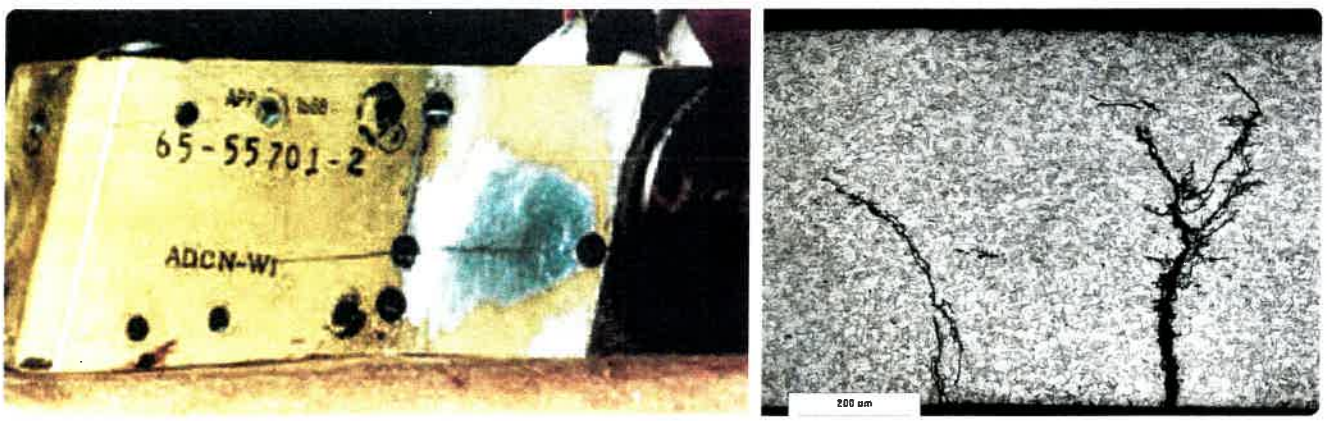


Figure 4-16. Stress corrosion cracking.

been exceeded, the metal will eventually crack and fail from metal fatigue. However, when the part or structure undergoing cyclic stress is also exposed to a corrosive environment, the stress level for failure may be reduced many times. Thus, failure occurs at stress levels that can be dangerously low depending on the number of cycles assigned to the life limited part.

Fatigue corrosion failure occurs in two stages. During the first stage the combined action of corrosion and cyclic stress damages the metal by pitting and crack formations to such a degree that fracture by cyclic stress will occur, even if the corrosive environment is completely removed.

The second stage is essentially a fatigue stage in which failure proceeds by propagation of the crack (often from a corrosion pit or pits). It is controlled primarily by stress concentration effects and the physical properties of the metal. Fracture of a metal part due to fatigue corrosion, generally occurs at a stress level far below the fatigue limit of an uncorroded part, even though the amount of corrosion is relatively small.

FRETTING CORROSION

Fretting corrosion is a particularly damaging form of corrosive attack that occurs when two mating surfaces normally at rest with respect to one another are subject to slight relative motion. It is characterized by pitting of the surfaces and the generation of considerable quantities of finely divided debris. Since the restricted movements of the two surfaces prevent the debris from easily escaping, an extremely localized abrasion occurs. (***Figure 4-17***)



Figure 4-17. Fretting corrosion.

The presence of water vapor greatly increases this type of deterioration. If the contact areas are small and sharp, deep grooves resembling brinell markings or pressure indentations may be worn into the rubbing surface. As a result, this type of corrosion on bearing surfaces has also been called false brinelling.

The most common example of fretting corrosion is the smoking rivet found on engine cowlings and wing skins. This is one corrosion reaction that is not driven by an electrolyte, and in fact moisture may inhibit the reaction. A smoking rivet is identified by a black ring around the rivet.

CAUSES OF CORROSION

The degree of severity, the cause, and the type of corrosion depend on many factors, including the size or thickness of the part, the material, heat treatment of the material, protective finishes, environmental conditions, preventative measures, and design.

Thick structural sections are generally more susceptible to corrosive attack because of variations in their composition, particularly if the sections are heat treated during fabrication. When large sections are machined or chem-milled after heat treatment, the corrosion characteristics of thinner sections may be different from those of thicker areas. Section size is based on structural requirements and cannot be changed for the purpose of controlling corrosion. From a maintenance standpoint, the correct approach is one of recognizing the need to ensure the integrity and strength of major structural parts and maintaining permanent protection over such areas at all times.

In service stresses and field repairs may affect the rates and types of corrosion. Aircraft structure under high cyclic stresses, such as helicopter main rotors, are particularly subject to stress corrosion cracking. Also areas adjacent to weld repaired items often have corrosion due to insufficient removal of the weld flux, or, for some steels, buildup of a magnetic field. Areas such as these should be closely inspected for signs of corrosion and, when found, proper treatment

Since corrosion is the deterioration of metals resulting from reaction between metals and their environment, some corrosion control or means to minimize corrosion when the aircraft enters operational service should be

introduced during the design phase. The corrosion issues discussed in this sub-module provide information to reduce the rate of corrosive attack by corrosion control measures introduced early in design.

The nature of the material is a fundamental factor in corrosion. High strength, heat treatable aluminum and magnesium alloys are very susceptible to corrosion, while titanium and some stainless steel alloys are less susceptible in atmospheric environment. The aircraft manufacturer selects material for the aircraft based on material strength, weight, and cost, while corrosion resistance is often a secondary consideration. However, corrosion control should be considered as early as possible during the preliminary design phase.

The use of more corrosion resistant materials in any design normally involves additional weight to achieve required strength. Since weight consideration is a major factor in the construction of airframes, the primary means of preventing corrosion is by use of protective coatings and proper maintenance procedures.

The use of corrosion resistant alloys is not a cure all for corrosion prevention. A common mistake is to replace a corroded part with a corrosion resistant alloy only to find that the corrosion has now shifted to another part and increased in severity.

The problem of protection against corrosion is minimized if the material to be protected is intrinsically resistant to corrosion. Aluminum copper alloys are known to have better stress corrosion resistance and better fatigue strength properties than aluminum zinc alloys; therefore, they are often used as the primary structural materials.

Galvanic corrosion is created by dissimilar metals being in contact with each other. The galvanic series of metals and alloys (***Figure 4-18***) is a factor that should be considered in the repair of aircraft. The further apart the metals listed in ***Figure 4-18*** are, the greater the tendency will be for galvanic corrosion. The metals grouped together in ***Figure 4-19*** have little differences in electrical potential; thus they are relatively safe to use in contact with one another. However, the coupling of metals from different groups will result in corrosion of the group having a lower number.

ELECTRODE POTENTIAL OF VARIOUS METALS AND ALLOYS

METAL / ALLOY	POTENTIAL (VOLTS) 0.1 CALOMEL SCALE
Magnesium	-1.73
Zinc	-1.10
7072, Alclad 3003, Alclad 6061, Alclad 7075	-0.96
5056, 7079-T6, 5456, 5083, 214, 218	-0.87
5052, 5652, 5086, 1099	-0.85
3004, B214, 1185, 1060, 1260, 5050	-0.84
1100, 3003, 615, 6053, 6061-T6, 6062-T6, 6063, 6363, Alclad 2014, Alclad 2024	-0.83
Cadmium	-0.82
7075-T6, 356, 360	-0.81
2024-T81, 6061-T4, 6062-T4	-0.80
355-T6	-0.79
2014-T6, 113, 750-T5	-0.78
2014-T4, 2017-T4, 2024-T3, 2024-T4	-0.68 to -0.70
Mild steel	-0.58
Lead	-0.55
Tin	-0.49
Copper	-0.20
Bismuth	-0.18
Stainless Steel (series 300, type 430)	-0.09
Silver	-0.08
Nickel	-0.07
Chromium	-0.49 to +0.018

Figure 4-18. Galvanic series of metals and alloys.

METALS AND ALLOYS GROUPED BY POTENTIAL

Group I	Magnesium and Magnesium Alloys
Group II	Aluminum, Aluminum Alloys, Zinc, Cadium and Cadium-Titanium Plate
Group III	Iron, Steels - Except Stainless Steels; Lead, Tin and thier Alloys
Group IV	Copper, Brass, Bronze, Copper-Berylium, Copper-Nickel, Chromium, Nickel, Nickel Base Alloys, Cobalt Base Alloys, Carbon Graphite, Stainless Steels, Titanium and Titanium Alloys

Note:
1. *Metals listed in the same group are considered similar to one another.*
2. *Metals listed in different groups are considered dissimilar to one another.*

Figure 4-19. Grouping of Metals and Alloys by potential.

Protective finishes are divided into 2 separate categories, sacrificial and non-sacrificial. Sacrificial coatings include cadmium, zinc, and aluminum. Non-sacrificial coatings include hard plating (chromium and nickel), chemical conversion coatings, sealant, primers, and top coat.

Aircraft systems exposed to marine atmospheres, moisture, acid rain, tropical temperature conditions, industrial chemicals, and soils and dust in the atmosphere can simply be disadvantaged by geographic or environmental factors. Limit, whenever possible, the requirement for operation of aircraft in adverse environments. Moisture is present in the air as a gas (water vapor) or as finely divided droplets of liquid (mist or fog) and often contains contaminants such as chlorides, sulfates, and nitrates, which increase its corrosive effects. Condensed moisture which evaporates will leave its contaminants behind. Condensed moisture and its contaminants can also be trapped in close fitting, wettable joints, such as faying surfaces and be drawn along poor bond lines bicapillary action.

Salt particles, when dissolved in water, form strong electrolytes. Normal sea winds carry dissolved salt which makes coastal environments highly corrosive. Industrial pollutants (such as carbon, nitrates, ozone, sulfur dioxide, and sulfates) contribute to the deterioration of nonmetallic materials and can cause severe corrosion of metals. Warm, moist air, normally found in tropical climates accelerates corrosion while cold, dry air normally found in arctic climates reduces corrosion. Proper heat treatment of materials is a vital factor in maximizing resistance to corrosion.

MATERIAL TYPES

Corrosion occurs in many different materials. The look and characteristics of corrosion varies with each metal. In the following section, common metals and environmental conditions found in aerospace are discussed.

ALUMINUM AND ALUMINUM ALLOYS

Aluminum and aluminum alloys are the most widely used material for aircraft construction. Aluminum appears high in the electro chemical series of elements and corrodes very easily. However, the formation of a tightly adhering oxide film offers increased resistance under most corrosive conditions. Most metals in contact with aluminum form couples that undergo galvanic corrosion attack. The alloys of aluminum are subject to pitting, intergranular corrosion, and intergranular stress corrosion cracking. In some cases, the corrosion products of metal in contact with aluminum are corrosive to aluminum. Therefore, aluminum and its alloys must be cleaned and protected.

Corrosion on aluminum surfaces is usually quite obvious, since the products of corrosion are white and generally more voluminous than the original base metal. Even in its early stages, aluminum corrosion is evident as general etching, pitting, or roughness of the aluminum surfaces.

Note: Aluminum alloys commonly form a smooth surface oxidation that is from 0.001" to 0.0025" thick. This is not considered detrimental. The coating provides a hard shell barrier to the introduction of corrosive elements. Such oxidation is not to be confused with the severe corrosion discussed in this paragraph.

General surface attack of aluminum penetrates relatively slowly, but speeds up in the presence of dissolved salts. Considerable attack can usually take place before serious loss of structural strength develops.

At least three forms of attack on aluminum alloys are particularly serious: the penetrating pit type corrosion through the walls of aluminum tubing, stress corrosion cracking of materials under sustained stress, and intergranular corrosion, which is characteristic of certain improperly heat treated aluminum alloys.

MAGNESIUM AND MAGNESIUM ALLOYS

Magnesium and magnesium alloys are the most chemically active of the metals used in aircraft construction and are the most difficult to protect. However, corrosion on magnesium surfaces is probably the easiest to detect in its early stages. Since magnesium corrosion products occupy several times the volume of the original magnesium metal destroyed, initial signs show a lifting of the paint films and white spots on the magnesium surface. These rapidly develop into snow like mounds or even white whiskers. The prompt and complete correction of the coating failure is imperative if serious structural damage is to be avoided.

FERROUS METALS

One of the most familiar types of corrosion is ferrous oxide (rust), generally resulting from atmospheric oxidation of steel surfaces. Some metal oxides protect the underlying base metal, but rust is not a protective coating in any sense of the word. Its presence actually promotes additional attack by attracting moisture from the air and acting as a catalyst for additional corrosion. If complete control of the corrosive attack is to be realized, all rust must be removed from steel surfaces.

Rust first appears on bolt heads, hold down nuts, or other unprotected aircraft hardware. (***Figure 4-20***) Its presence in these areas is generally not dangerous and has no immediate effect on the structural strength of any major components. The residue from the rust may also contaminate other ferrous components, promoting corrosion of those parts. The rust is indicative of a need for maintenance and of possible corrosive attack in more critical areas. It is also a factor in the general appearance of the equipment. When paint failures occur or mechanical damage exposes highly stressed steel surfaces to the atmosphere, even the smallest amount of rusting is potentially dangerous in these areas and must be removed and controlled. Rust removal from structural components, followed by an inspection and damage assessment, must be done as soon as feasible. (***Figure 4-21***)

NOBLE METALS

The noble metals silver, platinum and gold are used in aircraft assemblies because of their resistance to ordinary surface attack and their improved electrical or heat conductivity. A brown or black sulfide tarnish may occur that is normally not cleaned in the field by the technician. Copper and copper alloys are also relatively corrosion resistant. Attacks on copper components are usually limited to staining and tarnish. Such change in surface condition is not dangerous and should ordinarily have no effect on the function of the part. Serious copper corrosion is evident by the accumulation of green to blue copper salts on the corroded part. These are typically mechanically removed and then the area is treated with a chromic acid solution.

CHROMIUM AND NICKEL PLATED PARTS

Nickel and chromium platings are used extensively as protective and wear resistant coatings over high strength steel parts (landing gear journals, shock strut pistons,



Figure 4-20. Rust.



Figure 4-21. Rust on structural components.

etc.). Chromium and nickel plate provide protection by forming a somewhat impervious physical coat over the underlying base metal. When breaks occur in the surface, the protection is destroyed.

CADMIUM AND ZINC PLATED PARTS

Cadmium plating is used extensively in aircraft construction as a protective finish over both steel and copper alloys. Protection is provided by a sacrificial process in which the cadmium is attacked rather than the underlying base material. Properly functioning cadmium surface coatings may show mottling, ranging from white to brown to black spots on their surfaces. These show the sacrificial protection being offered by the cadmium coat, and under no condition should such spotting be removed merely for appearance sake. In fact, cadmium will continue to protect even when actual breaks in the coating develop and bare steel or exposed copper surfaces appear.

TITANIUM AND TITANIUM ALLOYS

Attack on titanium surfaces is generally difficult to detect. Titanium is, by nature, highly corrosion resistant, but it may show deterioration from the presence of salt deposits and metal impurities, particularly at high temperatures. Therefore, the use of steel wool, iron scrapers, or steel brushes for cleaning or for the removal of corrosion from titanium parts is prohibited.

If titanium surfaces require cleaning, hand polishing with aluminum polish or a mild abrasive is permissible if fiber brushes only are used and if the surface is treated following cleaning with a suitable solution of sodium dichromate. Wipe the treated surface with dry cloths to remove excess solution, but do not use a water rinse.

SUSCEPTIBILITY TO CORROSION (CORROSION PRONE AREAS)

This section lists corrosion problem areas that are common to most aircraft. Corrosion prone areas should be cleaned, inspected, and treated more frequently than less corrosion prone areas. The list is not necessarily complete for any specific aircraft, but could be used to set up a maintenance inspection program. Also included in this chapter are factors in corrosion control and preventative maintenance.

EXHAUST TRAIL AREAS

Both jet and reciprocating engine exhaust deposits are very corrosive and give particular trouble where gaps, seams, hinges, and fairings are located downstream from the exhaust pipes or nozzles. (***Figure 4-22***)

Deposits may be trapped and not reached by normal cleaning methods. Pay special attention to areas around rivet heads and in skin lap joints and other crevices. Remove and inspect fairings and access plates in the exhaust areas. Do not overlook exhaust deposit buildup in remote areas, such as the empennage surfaces. Buildup in these areas is slower and may not be noticed until corrosive damage has begun.

BATTERY COMPARTMENTS AND BATTERY VENT OPENINGS

In spite of protective paint systems and extensive sealing and venting provisions, battery compartments continue to be corrosion problem areas. Fumes from overheated battery electrolyte are difficult to contain and will spread to internal structure. Unprotected surfaces will be subjected to corrosive attack. For lead acid batteries, frequent cleaning and neutralization of acid deposits with sodium bicarbonate solution will minimize corrosion. If the battery installation includes external vent openings on the aircraft skin, these areas should be included in the inspection and maintenance procedure.

If aircraft batteries with electrolytes of either sulfuric acid or potassium hydroxide are in use, their leakage will cause corrosion. Consult the applicable maintenance manuals for the particular aircraft to determine which type battery was installed and the recommended maintenance practices for each. Cleaning of nickel

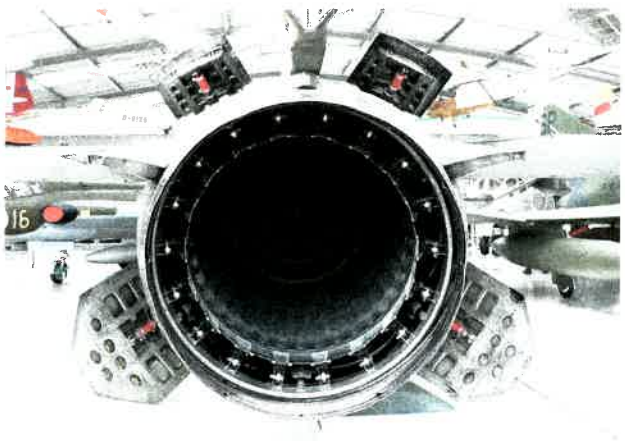


Figure 4-22. Exhaust nozzle area.



Figure 4-23. (A) Typical lavatory floor damages shown with the toilet is removed. (B) Corrosion caused by water and urine collecting on a lavatory attach bracket.

cadmium compartments should be done with ammonia or boric acid solution, allowed to dry thoroughly, and then painted with an alkali resistant varnish.

LAVATORIES, BUFFETS, AND GALLEYS

These areas, particularly deck areas behind lavatories, sinks, and ranges, where spilled food and waste products may collect if not kept clean, are potential trouble spots. Even if some contaminants are not corrosive in themselves, they will attract and retain moisture and, in turn, cause corrosive attack. Pay attention to bilge areas located under galleys and lavatories. Clean these areas frequently and maintain the protective sealant and paint finishes. (***Figure 4-23, A and B***)

BILGE AREAS

On all aircraft, the bilge area is a common trouble spot. The bilge is a natural sump or collection point for waste hydraulic fluids, water, dirt, loose fasteners, drill chips, and other odds and ends of debris. Residual oil quite often masks small quantities of water which settle to the bottom and set up a hidden potential corrosion cell. With the exception of water displacing corrosion preventative compounds, keeping bilge areas free of all extraneous material including water and oil will insure the best protection against corrosion. A good vacuum cleaner and clean wiping cloths are necessary to clean such areas.

WHEEL WELLS AND LANDING GEAR

The wheel well area probably receives more punishment than any other area of the aircraft. It is exposed to mud, water, salt, gravel, and other flying debris from runways during flight operations (***Figure 4-24***).



Figure 4-24. The landing gear area should be cleaned and inspected more frequently than other areas.

Frequent cleaning, lubrication and paint touch up are needed on aircraft wheels and on wheel well areas. Because of the many complicated shapes, assemblies, and fittings in the area, complete coverage with a protective paint film is difficult to attain. Thus, preservative coatings tend to mask trouble rather than prevent it. Because of the heat generated from braking, preservative coatings cannot be used on aircraft landing gear wheels. During inspection of this area, particular attention should be given to the following trouble spots:

1. High strength steel.
2. Exposed surfaces of struts, oleos, arms, links, and attaching hardware (bolts, pins, etc.).
3. Axle interiors.
4. Exposed position indicator switches and other electrical equipment.
5. Crevices between stiffeners, ribs, and lower skin surfaces which are typical water and debris traps.

6. Magnesium wheels, particularly around bolt heads, lugs, and wheel web areas.
7. Exposed rigid tubing, especially at "B" nuts and ferrules under clamps and tubing identification tapes.

EXTERNAL SKIN AREAS

External aircraft surfaces are ordinarily covered with protective finishes. In addition, paint coatings may be applied. The affected external aircraft surfaces are readily visible or available for inspection and maintenance. Much emphasis has been given to these areas in the past, and maintenance procedures are well established. Even here, certain types of configurations or combinations of materials become troublesome and require special attention if serious corrosion difficulties are to be avoided. Some of the common trouble areas, other than those attributed to engine exhaust deposits, are given below.

Fasteners and areas around steel, titanium, Corrosion Resistant Stainless Steel (CRES) and nickel alloy fasteners are trouble spots. These areas are subject to high operational loads, moisture intrusion, and dissimilar metal corrosion (***Figure 4-25***).

Similar to corrosion around fasteners, corrosion in faying surfaces, seams, and joints is caused by the intrusion of moisture and other corrosive agents. The effect of this type of corrosion is usually detectable by bulging of the skin surface.

Properly surface treated, insulated, and painted magnesium skin surfaces give relatively little trouble from a corrosion standpoint if the original surface is maintained. However, trimming, drilling and riveting destroy some of the original surface treatment which may not be completely restored by touch up procedures.

Some aircraft have steel fasteners installed through magnesium skin with only protective finishes under the fastener heads, and fillet sealant or tape over the surface for insulation. Further, all paint coatings are inherently thin at abrupt changes in contour, such as at trimmed edges. With magnesium's sensitivity to moisture, all of these conditions add up to a potential corrosion problem whenever magnesium is used. Any inspection for corrosion should include all magnesium skin surfaces, as well as other magnesium fittings or structural components, with special attention to edges, areas around skin edges and fasteners, and cracked, chipped, or missing paint. Corrosion of spot welded skin construction is chiefly the result of the entrance and entrapment of moisture or other corrosive agents between layers of the metal (***Figure 4-26***).

Spot welded assemblies are particularly corrosion prone. Corrosive attack causes skin buckling or spot weld bulging, and eventually spot weld fracture. Some of the corrosion may be caused originally by fabricating processes, but its progress to the point of skin bulging and spot weld fracture is the direct result of moisture or other corrosive agents working its way through open

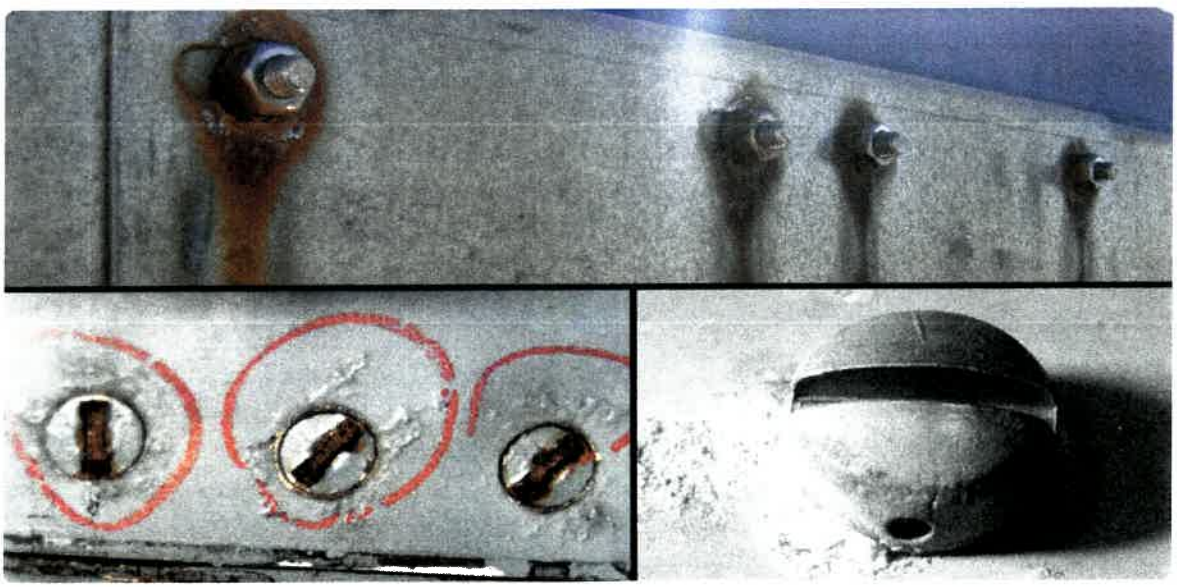


Figure 4-25. Corrosion around fasteners due to dissimilar metal contact.

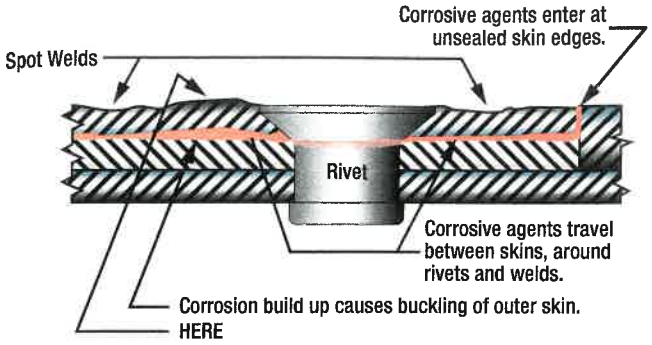


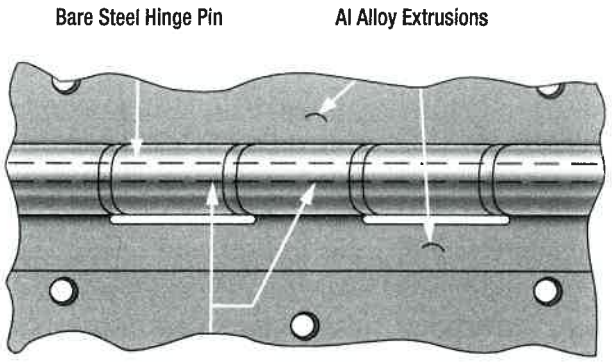
Figure 4-26. Spot-welded skin corrosion.

gaps or seams. The use of weld through sealing materials is expected to minimize this problem, but many inservice aircraft still have unsealed spot weld skin installed. This type of corrosion is evidenced by corrosion products appearing at the crevices through which the corrosive agents enter.

Corrosion may appear at either external or internal faying surfaces, but it is usually more prevalent on external areas. More advanced corrosive attack causes skin buckling and eventual spot weld fracture. Skin buckling in the early stage may be detected by sighting or feeling along spot welded seams or by using a straight edge. To prevent these conditions, keep potential moisture entry points including gaps, seams, and holes created by broken welds filled with noncorrosive sealant.

Piano type hinges are prime spots for corrosion due to dissimilar metal contact between the steel pin and the aluminum hinge tabs. They are also natural traps for dirt, salt and moisture. Where this type of hinge is used on access doors or plates, and actuated only when opened during an inspection, they tend to corrode and freeze in the closed position between inspections. When the hinge is inspected, it should be lubricated and actuated through several cycles to ensure complete penetration of the lubricant (***Figures 4-27 and 4-28***).

Heavy or thick sections of most heat treated aluminum alloys are susceptible to pitting or intergrannular corrosion and exfoliation of the metal. When inspecting external skin surfaces, especially around countersunk fastener heads, look for white or grey powder deposits or metal exfoliation. This is usually first evident as small raised areas or bumps under paint film.



Hidden corrosion occurs here. Joint freezes and lugs break off when hinge is actuated.

Figure 4-27. Hinge corrosion points.

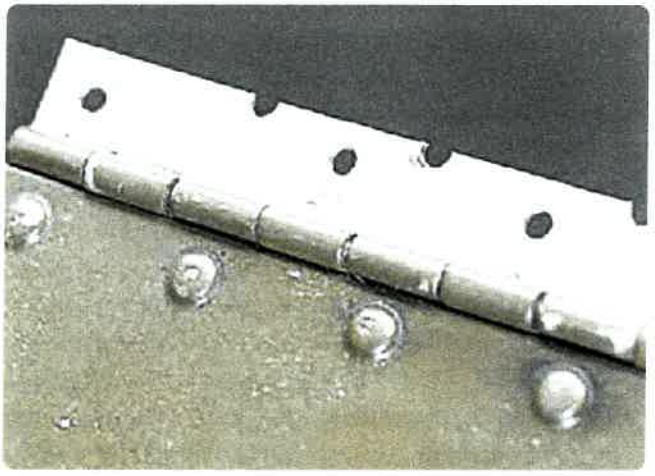


Figure 4-28. Hinge corrosion points. You must be able to recognize and take corrective action whenever corrosion is detected or suspected.

Organic composites used in aircraft can cause different corrosion problems than those normally associated with all metal structure. Composite such as graphite/epoxy act as a very noble (cathodic) material, creating the potential for galvanic corrosion. The galvanic corrosion potential coupled with different methods of attachment (i.e., adhesive bonding, stepped structures, locking mechanical fasteners, etc.) lead to multicomponent galvanic couples with the problem being particularly aggravated by high humidity and salt water environments. Application of aircraft sealants over the dissimilar metal/composite junctions will prevent galvanic corrosion if moisture is completely excluded. However, since complete exclusion of moisture is virtually impossible under extended periods of flight the most effective method of eliminating the voltage potential is to provide a nonconductive layer such as fiberglass/epoxy and/or sealant between the composite and dissimilar metal surfaces.

WATER ENTRAPMENT AREAS

Corrosion will result from the entrapment of moisture. With the exception of sandwich structures, design specifications usually require that the aircraft have low point drains installed in all areas where moisture and other fluids can collect. In many cases, these drains are ineffective either due to location or because they are plugged by sealants, extraneous fasteners, dirt, grease, and debris. Potential entrapment areas are not a problem when properly located drains are functioning, and the aircraft is maintained in a normal ground attitude. However, the plugging of a single drain hole or the altering of the level of the aircraft can result in a corrosion problem if water becomes entrapped in one of these "bathtub" areas. Daily inspection of low point drains is a recommended practice.

ENGINE FRONTAL AREAS AND COOLING AIR VENTS

Constant abrasion by airborne dirt and dust, bits of gravel from runways, and rain tends to remove the protective surfaces from these areas. Furthermore, cores of radiator coolers, reciprocating engine cylinder fins, etc., due to the requirement for heat dissipation, may not be painted. Engine accessory mounting bases usually have small areas of unpainted magnesium or aluminum on the machined mounting surfaces. With moist and salt or industrial pollutant laden air constantly flowing over these surfaces, they are prime sources of corrosive attack.

Inspection of such areas should include all sections in the cooling air path with special attention to obstructions and crevices where salt deposits may build up during marine operations (***Figures 4-29 and 4-30***).

ELECTRONIC PACKAGE COMPARTMENTS

Electronic and electrical package compartments cooled by ram air or compressor bleed air are subjected to the same conditions common to engine and accessory cooling vents and engine frontal areas. While the degree of exposure is less because of a lower volume of air passing through and special design features incorporated to prevent water formation in enclosed spaces, this is still a trouble area that requires special attention.

Circuit breakers, contact points, and switches are extremely sensitive to moisture and corrosive attack and should be inspected for these conditions as thoroughly as design permits. If design features hinder examination while in the installed condition, inspection should be accomplished after component removal for other reasons.

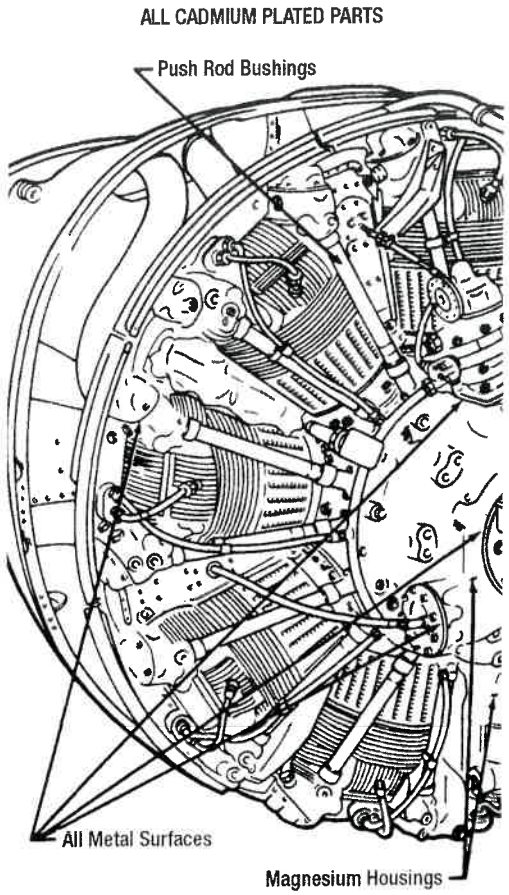


Figure 4-29. Reciprocating engine frontal area corrosion.

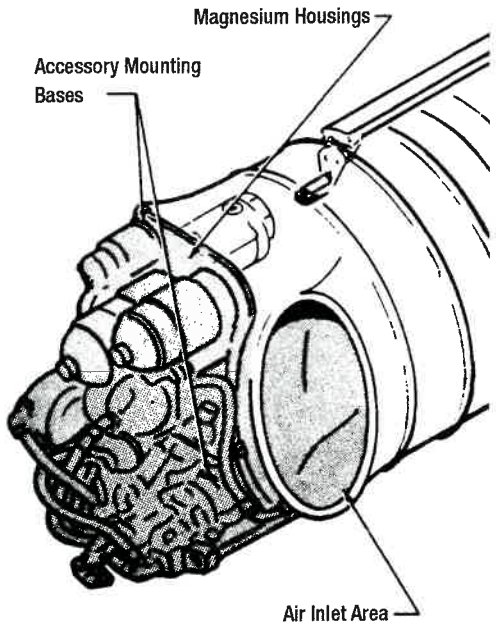


Figure 4-30. Turbine engine frontal area corrosion points.

Treatment of corrosion in electrical and electronic components should be done only by, or under the direction of qualified personnel familiar with the function of the unit involved as conventional corrosion treatment may be detrimental to some units.

Most corrosion that occurs on avionic equipment is similar to that which occurs on the basic airframe structure. The difference between avionic and airframe corrosion is that minute amounts of corrosion in avionic equipment can cause serious degradation or complete failure, while it would be unnoticed on larger structures.

Smog, smoke, soot, and other airborne contaminants are extremely corrosive to exposed avionic equipment. Many fumes and vapors emitted from industrial complexes are highly acidic and greatly accelerate corrosion. An example is the corrosive effect of ozone, a product of many welding machines and large electrical motors. Complete degradation of rubber seals and damage to delicate components have occurred in equipment stored near ozone producing equipment. Avionic shops and storage areas should have a filtered air conditioning system.

Another man made atmosphere is the aircraft environmental control system. These systems induce cooling air to the equipment. They may include a filter system that extracts moisture, and in some cases contaminants, from the air that enters the equipment. Failure to replace and/or clean a filter, or eliminate a leaky environmental seal, may cause a moisture or contaminant buildup that could cause a corrosive atmosphere within the equipment.

The control of corrosion in avionic systems is similar to that in airframes, with appropriate modifications. The general differences in construction and procedures between airframe and avionics relative to corrosion control are:

1. Less durable protection systems.
2. Small amounts of corrosion can make equipment inoperative.
3. Dissimilar metals are often in electrical contact.
4. Stray currents can cause corrosion.
5. Active metals and dissimilar metals in contact are often unprotected.
6. Closed boxes· can produce condensation during normal temperature changes during flight.
7. Avionic systems have many areas to trap moisture.
8. Hidden corrosion is difficult to detect in many avionic systems.
9. Many materials used in avionic systems are subject to attack by bacteria and fungi.
10. Organic materials are often used which, when overheated or improperly or incompletely cured, can produce vapors which are corrosive to electronic components and damaging to coatings and insulators.

MISCELLANEOUS TROUBLE AREAS

A variety of additional trouble spots exists. Some are covered by manufacturer's' publications. Examine all flexible hose assemblies for chafing, weather checking, hardening, discoloration, evidence of fungus, and torn weather protective coatings or sleeves. Replace hoses that are found to be discrepant.

Trimmed edges of sandwich panels and drilled holes should have some type of corrosion protection. A brush treatment with an inhibitor solution or the application of a sealant along the edge, or both, is recommended. Any gaps or cavities where moisture, dirt, or other foreign material can be trapped should be filled with a sealant. The adjacent structure (not the sandwich) should have sufficient drainage to prevent moisture accumulation. Damage or punctures in panels should be sealed as soon as possible to prevent additional moisture entry, even if permanent repair has to be delayed.

Control cables may present a corrosion problem whether carbon steel or stainless steel is used. The presence of bare spots in the preservative coating is one of the main contributing factors in cable corrosion. Cable condition should be determined by cleaning the cable assembly, inspection for corrosion, and application of an approved preservative if no corrosion is found. If external corrosion is found, relieve tension on the cable and check internal strands for corrosion. Cables with corrosion on internal strands should be replaced. Pay particular attention to sections passing through fairleads, around sheaves, and grooved bellcrank arms. External corrosion should be removed by a clean, dry, coarse rag or fiber brush. After complete corrosion removal, apply a preservative.

Topcoating materials (Buna-N, Polyurethane, and Epoxy) used in integral fuel cells are impervious to fuel but not completely impervious to moisture absorption.

Since it is impossible to keep fuel completely free of water, moisture can penetrate through the topcoating materials and sometimes causes pitting or intergranular corrosion on aircraft structural parts. It has also been found that microorganisms which live in the water entrained by fuel, particularly jet propellant types, feed on fuel hydrocarbon and hydrocarbon type elastomeric coatings materials. These microorganisms excrete organic acids, and dead micro organisms act as a gelatinous acidified sponge which can deteriorate integral tank coating and corrode the aircraft structure. Microbial corrosion can be minimized by preventing as much water contamination of the fuel as possible with well managed storage facilities, adequate filtration of fuel, and drainage of water contamination from integral fuel cells which keeps the water moving and reduces the chance for the colonies of microorganisms to develop. Micro organic activity can be reduced by using a biocide additive such as "Biobor JF" or equivalent. Solution strength and application frequency should be in accordance with the manufacturer's instructions.

Electrical connectors/components may be potted with a sealing compound to provide more reliability of equipment. The sealing compound prevents entrance of moisture into the area of connectors where the wires are attached to the pins. Rubber o-rings may also be used to seal moisture out of the mating area of pin connections and to prevent loss of pressurization in compartments containing bulkhead connectors.

Moisture intrusion into electrical connectors can cause corrosion and an electrical failure. Suspected plugs should be disconnected, dissembled, solvent cleaned, and inspected for corrosion. When sealing provisions are not designed into the electrical component, these components can have moisture intrusion and internal corrosion.

Severe corrosion damage to the rear pressure bulkhead below the floor may occur as a result of contamination by fluids. Inspection for rear bulkhead corrosion may require extensive disassembly of components and fixtures to allow a thorough visual inspection. When inspection access holes are available, inspection by fiber optics is useful. Other Nondestructive Inspection (NDI) methods (x-ray, ultrasonic, and eddy current) are also available. However, these inspection techniques require specially trained personnel, NDI comparison standards, and suitable access. A regular inspection of the rear pressure bulkhead (both front and rear faces) below the floor level should be accomplished to prevent serious corrosion from occurring between the bulkhead and periphery doubler at the floor level. Such corrosion could weaken the bulkhead skin and cause sudden cabin pressure loss.

Some older aircraft have developed delaminations in cold bonded joints. Corrosion between the delaminated surfaces is caused by moisture intrusion along the edge of the mating parts or around fasteners securing the mating parts together. Localized bulging of the skin or internal structural component, usually around the fasteners, is the first indication of a corrosion problem (***Figure 4-31***).

Skin cracks or dished or missing fastener heads may also indicate severe corrosion in bonded joints. Corrosion which occurs between skins, doublers, and stringers or frames will produce local bulging or pulled rivets (***Figure 4-32***).

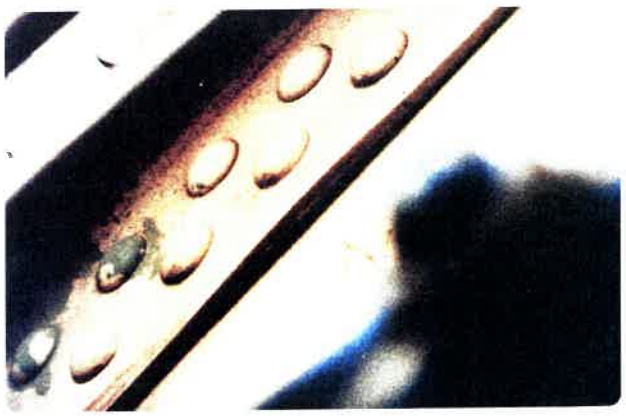


Figure 4-31. Spar chord lifting caused by corrosion products.



Figure 4-32. Skin bulging around fasteners.

Corrosion that occurs between the skins and doublers or tear straps away from backup structure such as stringer or frame will not produce local bulging. An external low frequency eddy current inspection may be used to determine the extent of corrosion in the skin. Lap joints should be opened with wedges to determine the full extent of corrosion damage (***Figures 4-33 and 4-34***).

Internal visual inspection should be used to detect delaminated doublers or tear straps. A penetrating water displacement corrosion inhibitor should be applied to faying surfaces after corrosion removal and repair.

Flap and slat recesses and equipment installed in these areas, which are normally closed, may corrode unnoticed unless special inspections are performed.

SEAPLANES AND FLOODING

A special problem is encountered in the conversion of land planes to seaplanes. In general, land planes do not receive corrosion proofing to the same extent as do seaplanes. Corrosion proofing standards for land planes converted to seaplanes are divided into two classes, necessary minimum precautions and recommended precautions. Regardless of such precautions, it is imperative that the exterior surfaces of seaplanes be washed with clear fresh water immediately following extended water operation, or at least once a day when operated in salty or brackish water. Wash interior surfaces of seaplanes exposed to spray, taking care to prevent damage to electrical circuits or other items subject to injury.

Aircraft recovered from partial or total immersion in standing water or flash floods require an indepth inspection and cleaning of both the exterior and interior areas. Water immersion increases the probability of corrosive attack, it removes lubricants, deteriorates aircraft materials, and destroys electrical and avionics components. Sea water, because of salt content, is more corrosive than fresh water. However, fresh water may also contain varying amounts of salt and, as drying occurs, the salt concentration is increased and corrosive attack accelerated.

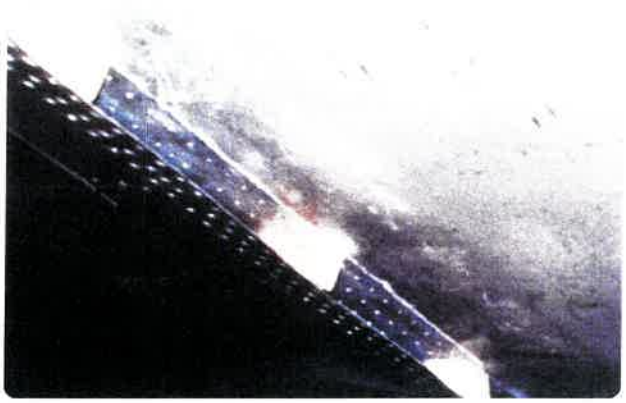


Figure 4-33. Opening of a corroded lap joint for repair.

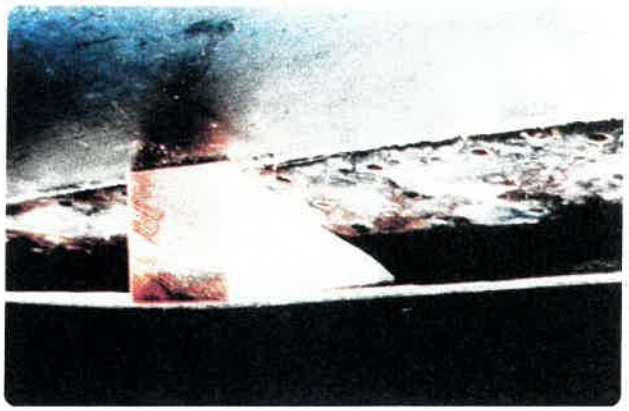


Figure 4-34. Close-up view of a corroded lap joint.

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Question: 4-1
Corrosion is the _____ deterioration of a metal.

Question: 4-2
Substances that cause corrosion are called corrosive _____.

Question: 4-3
Fretting corrosion is also known as _____ corrosion.

Question: 4-4
Cadmium plating used extensively in aircraft construction provides corrosion protection through a _____ process.

Question: 4-5
Name six corrosion prone areas on an aircraft.

Question: 4-6
Airborne particle abrasion contributes to corrosion by removing _____ surfaces exposing bare metal to the elements.

Question: 4-7
Electrical connectors may be potted with sealing compound or may use _____ to keep moisture from entering the area where wire attach to the connector pins.

Question: 4-8
Galvanic corrosion is created by _____ metals being in contact with each other.

ANSWERS

Answer: 4-1
electrochemical.

Answer: 4-2
agents.

Answer: 4-3
wear.

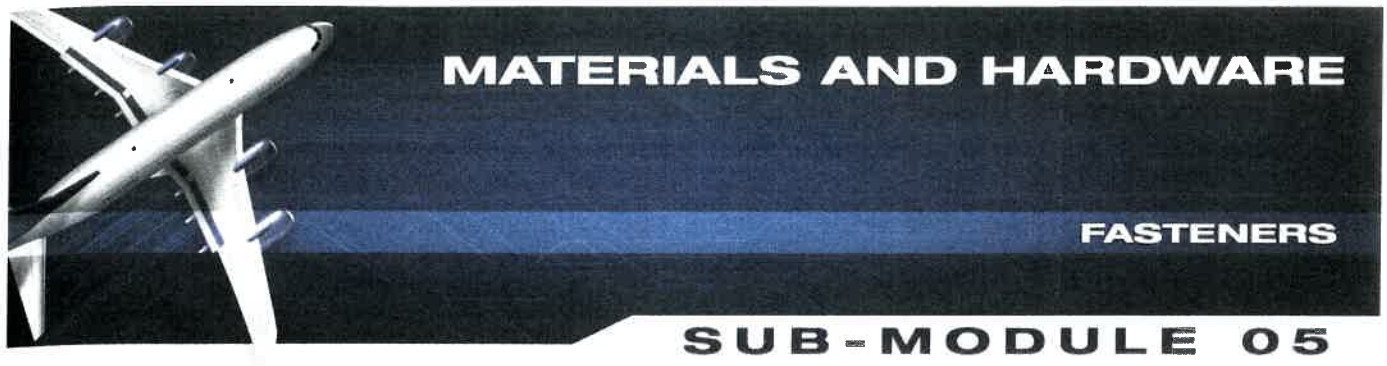
Answer: 4-4
sacrificial.

Answer: 4-5
Exhaust trail areas.
Battery compartments and battery vent openings.
Lavatories, buffets and galleys.
Bilge areas.
Wheel wells and landing gear.
External skin areas.
Water entrapment areas.
Engine frontal areas and cooling vents.
Electronics bays.

Answer: 4-6
protective.

Answer: 4-7
o-rings.

Answer: 4-8
dissimilar.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY → B1

Sub-Module 05

FASTENERS

Knowledge Requirements

6.5 - Fasteners

6.5.1 - Screw Threads

Screw nomenclature; Thread forms, dimensions and tolerances for standard threads used in aircraft; Measuring screw threads.	2

6.5.2 - Bolts, Studs And Screws

Bolt types: specification, identification and marking of aircraft bolts, international standards; Nuts: self locking, anchor, standard types; Machine screws: aircraft specifications; Studs: types and uses, insertion and removal; Self tapping screws, dowels.	2

6.5.3 - Locking devices

Tab and spring washers, locking plates, split pins, pal-nuts, wire locking, quick release fasteners, keys, circlips, cotter pins.	2

6.5.4 - Aircraft rivets

Types of solid and blind rivets: specifications and identification, heat treatment.	2

6.5 - FASTENERS

Aircraft hardware is the term used to describe the various types of fasteners and miscellaneous small items used in the manufacture and repair of aircraft. The importance of aircraft hardware is often overlooked because of its small size; however, the safe and efficient operation of any aircraft is greatly dependent upon the correct selection and use of aircraft hardware.

An aircraft, even though made of the best materials and strongest parts, would be of doubtful value unless those parts were firmly held together. Several methods are used to hold metal parts together; they include riveting, bolting, brazing, and welding. The process used must produce a union that will be as strong as the parts that are joined.

Aircraft are nothing more than a multitude of parts flying in formation. While this old saying is meant as a joke, it also points out that the numerous parts and assemblies that make up an aircraft must be fastened together properly to operate successfully.

Fasteners are often used to assemble aircraft parts and assemblies. Each fastener in an aircraft has been considered by design engineers for the type of loads it will encounter. It has been selected to perform a specific function and, should it need to be replaced, must be replaced with the exact same fastener. Substitution of fasteners installed by the manufacturer may only be done with fasteners that have all of the same characteristics as the original. This requires an in depth knowledge of the engineering behind the fastener choice and complete knowledge of each fastener's characteristics. To this extent, substitution of fasteners with other than those originally installed should be avoided.

Fasteners are manufactured to standards. Each county may have several organizations that set standards for aircraft hardware and there are many organizations that maintain international standards. Aircraft manufacturer's choose fasteners based on a variety of reasons. Among other things, they rely on these standards to select fasteners with the correct properties for each installation. The technician in the field learns of these standards largely through working with the various fasteners aircraft and component manufacturer's have installed.

This chapter is written with the inclusion of hardware used in aircraft manufactured in the United States of America. As such, many of the standards used in the U.S. are presented. Manufacturer's of aircraft in other countries may or may not use the same hardware. While there is a movement towards international standardization, slight differences in dimensions and even nomenclature of aircraft fasteners may exist. Moreover, identification numbers and markings may differ on aircraft hardware depending on which standards are followed. The technician is charged with utilizing the approved hardware of the manufacturer regardless which national or international standard is employed. Keep in mind that processes are also standardized and may be included in an organization's standards.

For recognition, some of the U.S. standards and international standardization organizations are listed:

MS - Military Standard
NAS - National Aerospace Standards
AN - Air Force Navy
ISO - International Organization for Standardization
AS - Standards Australia
SAE - Society of Automotive Engineers
ASTM - American Society for Testing and Materials
ANSI - American National Standards Institute
BSI - British Standards Institute

6.5.1 - SCREW THREADS

SCREW PRINCIPLES

Two of the most common fasteners used in aviation are the bolt and the screw. They are similar in that they both have threads. ***Figure 5-1*** illustrates the parts of the threads on a threaded fastener.

In general, screws differ from bolts in the following ways. Screw usually have lower material strength and a looser thread fit than a bolt. The shank of a screw is typically threaded along its entire length without a clearly defined grip. Most screws thread into the material they are intended to fasten and do not utilize a nut. A bolt, on the other hand, requires a nut for it to function.

Threaded fasteners allow parts to be fastened together with all of the strength that unthreaded fasteners provide. However unlike rivets and pins, threaded fasteners may be dissembled and reassembled an almost infinite number of times. Due to the large range of different available fasteners, great care must be taken to select the correct fastener for each particular installation.

THE INCLINED PLANE

The value of the wedge as a means of transmitting motion is well known. For a constant effort applied in driving a wedge, a smaller angle of inclination between the planes will cause a greater force to be exerted through a shorter distance. Conversely, a larger angle will cause less force to be exerted through a greater distance.

The action of the wedge may also be reversed, with the wedge moving when a force is applied to the inclined surface. This is seen when the angle is large. The larger the angle becomes, the more readily the motion is reversed. But no matter how small the angle may be, the resulting forces will still tend to produce movement.

When a continuous inclined plane is cut around the outside or inside of a cylinder, then a spiral known as a helix is produced. The helix angle is important because it dictates the number of threads which can be cut per axial increment in the cylinder.

SCREW NOMENCLATURE

The difference between a bolt and a screw are sometimes disputed, but it is more often agreed that a bolt is considered to be a threaded fastener which has a definite plain portion on the shank between its head and the beginning of the thread and is used together with a nut, while a screw is threaded all the way to the head. However because there are so many variations in terminology between different manufacturers, the only safe way to identify a fastener is to use the identical terminology as in the manufacturer's illustrated parts catalog when selecting or ordering replacements.

BOLT AND SCREW TERMINOLOGY

Figure 5-1 illustrates the parts of a bolt which is generally the same for a screw. A common screw and a machine screw are illustrated in ***Figure 5-2***. Note that there are several types of structural screws such as the machine screw which also uses a nut like a bolt. Also note that the length of a countersunk screw is measured from the top of the head.

SCREW THREAD TERMINOLOGY

Figure 5-3 illustrates the parts of the threads on a threaded fastener.

- Major Diameter – the crest to crest distance of the thread is known as the major diameter.
- Minor Diameter – the root to root distance of the thread is known as the minor diameter.

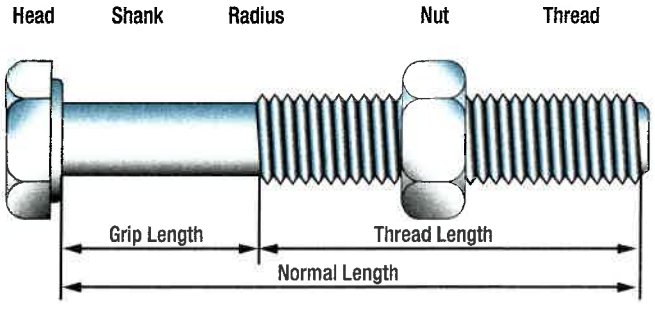


Figure 5-1. Bolt and screw terminology.

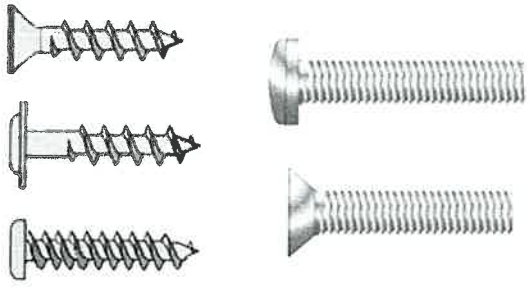


Figure 5-2. Common screws and machine screws.

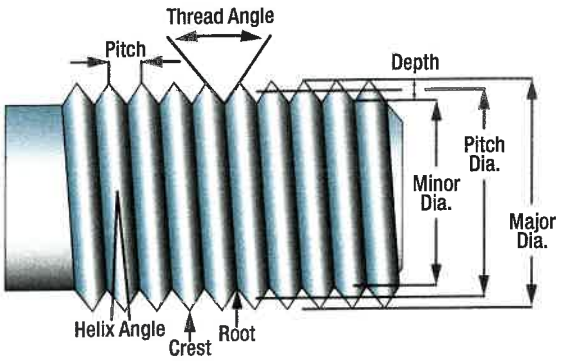


Figure 5-3. Threaded fastener nomenclature.

- Pitch – The distance from the center of one crest to the center of the next as measured parallel to the axis.
- Flank – the surface of the thread which connects the root to the crest.
- Lead – The distance a screw moves axially in one complete turn.
- Runout – the part of the thread where the minor diameter increases until it equals the major diameter and merges with the plain portion of the shank.

THREAD FORMS, DIMENSIONS AND TOLERANCES

COARSE AND FINE PITCH THREADS

Two screw threads may have the same major diameter and similar form but different depths of thread. The screw with deep thread will have fewer threads per inch and so pitch is said to be coarser than the screw with the shallow thread. The coarse pitch thread will have greater lead than the fine pitch thread but the minor diameter will be less and so less strength. A screw with fine pitch thread will have stronger crest position, tighter grip, finer adjustment and resistance to vibration. (***Figure 5-4***)

SINGLE AND MULTI START THREADS

A single start thread is when there is only one screw thread cut in the material. A multi-start thread consists of two or more separate parallel threads cut into the material carrying the thread. (***Figure 5-5***)

SCREW THREAD PROFILE

Forms of screw threads vary according to the purpose for which they are used and also according to the country in which they are manufactured. Most threads are V form, but some are square form, or modifications of either, or both.

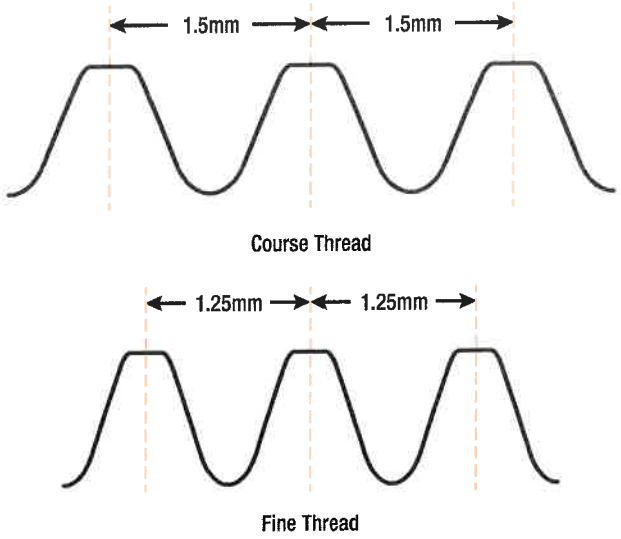


Figure 5-4. Course and fine pitch threads.

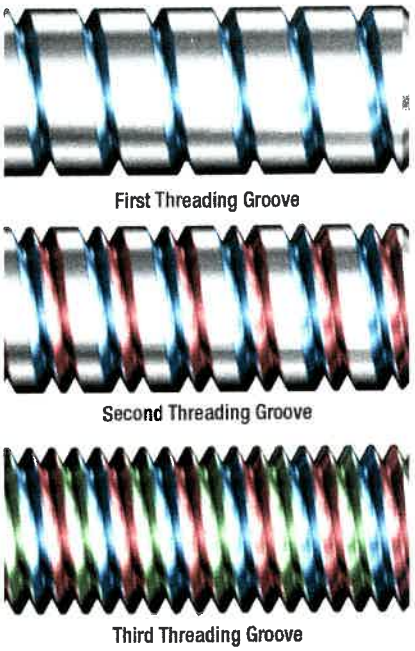


Figure 5-5. Single and multi start threads.

INTERNATIONAL THREAD SYSTEM

Aircraft screws, bolts, and nuts are manufactured to many different international standards and in a variety of thread forms. Most aircraft now use unified or metric threads but some older aircraft use the obsolete British Standard or Whitworth thread forms. None of these are compatible with the unified or metric forms.

Unified Screw Thread

These threads are of two basic series in which diameter is related to pitch. Both use the same thread form and are designated as United Coarse Thread (UNC) and

United Fine Thread (UNF). The UNC series covers sizes ranging from ¼" major diameter to 4". The UNF are available in fractional sizes from ¼" to ½".

Metric Screw Thread

Metric screw threads have a thread angle of 60°. Those used in aerospace use the MJ profile which is suitable for highly stressed applications. The MJ profile is a metric version similar to UNJ thread. Aerospace screws, bolts, and nuts are manufactured in diameter and pitch variations from 1.6 mm to 39 mm. The metric thread is identified by the letters MJ to identify metric followed by the size and pitch in millimeters and followed by the tolerance class. For example MJ6 × 1-3

CHOOSING THREADED FASTENERS

There are a wide variety of fastener styles, each with their own properties. It is important that only the fastener specified for a particular location be used in that location. Installation of the incorrect type of fastener may have catastrophic consequences. The illustrated parts catalog for the aircraft or equipment specifies each fastener by part number. The part number must always be used to identify the parts needed for any job. When a part has been superseded, an alternate part can only be substituted under suitable authority. One source of information on acceptable alternatives is the Structural Repair Manual (SRM) of that aircraft which gives a recognized substitutions table.

Note that thread types are not interchangeable. For example, a UNF screw must be used only in a UNF nut of the same size. The size of the hole to be drilled for tapping an internal thread and holes for screw clearance are given in the applicable national standard. However tapping and clearance drill sizes are reproduced in readily available charts such as ZEUS reference tables.

THREAD TOLERANCES

Threads are also designated by Class of fit. The Class of a thread indicates the tolerance allowed in manufacturing. Class 1 is a loose fit, Class 2 is a free fit, Class 3 is a medium fit, and Class 4 is a close fit. Aircraft bolts are almost always manufactured in the Class 3, medium fit.

A Class 4 fit requires a wrench to turn the nut onto a bolt, whereas a Class 1 fit can easily be turned with the fingers. Generally, aircraft screws are manufactured with a Class 2 thread fit for ease of assembly. Bolts and nuts are also produced with right hand and left hand threads. A right hand thread tightens when turned clockwise; a left hand thread tightens when turned counterclockwise.

MEASURING SCREW THREADS

SCREW THREAD PITCH

Screw thread pitch on a screw, bolt, or tapped hole can be verified and measured with a screw thread pitch gauge as shown in ***Figure 5-6***. Gauges are available for most thread patterns as discussed above. To use, simply match the threads on any screw or bolt to the threaded edge of the a leaf. If the threads align perfectly you have identified the thread pitch correctly.

GO-NOGO GAUGES

Go-NoGo gauges such as shown in ***Figure 5-7*** provide a quick way to judge whether a threaded plug will properly fit into a threaded hole within tolerance limits. Be careful not to force the gauge into the test hole, especially on thin walled parts, as an incorrect attempt could easily distort the receptacle.

It is important that all thread gauges should be checked periodically to ensure that they are not worn beyond permissible limits or otherwise damaged. Checking is normally done by skilled inspectors. Gauges used continuously should be checked daily. Those used intermittently should be checked weekly.

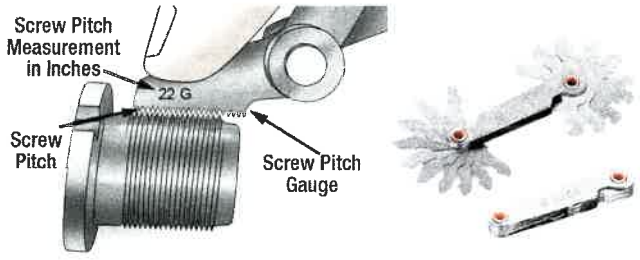


Figure 5-6. Screw thread pitch tools.



Figure 5-7. Go-NoGo gauges.

BOLT TYPES: SPECIFICATIONS, IDENTIFICATION, AND MARKING OF AIRCRAFT BOLTS, INTERNATIONAL STANDARDS

AIRCRAFT BOLT TYPES

Aircraft bolts are fabricated from cadmium or zincplated corrosion resistant steel, unplated corrosion resistant steel, or anodized aluminum alloys. Most bolts used in aircraft structures are either general purpose, AN bolts, or NAS internal wrenching or close tolerance bolts, or MS bolts. In certain cases, aircraft manufacturer's make bolts of different dimensions or greater strength than the standard types. Such bolts are made for a particular application, and it is of extreme importance to use like bolts in replacement. Special bolts are usually identified by the letter "S" stamped on the head.

AN bolts come in three head styles; hex head, clevis, and eyebolt. NAS bolts are available in hex head, internal wrenching, and countersunk head styles. MS bolts come in hex head and internal wrenching styles. (***Figure 5-8***)

IDENTIFICATION AND CODING

Bolts are manufactured in many shapes and varieties. A clear cut method of classification is difficult. Bolts can be identified by the shape of the head, method of securing, material used in fabrication, or the expected usage. AN-type aircraft bolts can be identified by the code markings on the bolt heads.

The markings generally denote the bolt manufacturer, the material of which the bolt is made, and whether the bolt is a standard AN-type or a special purpose bolt. AN standard steel bolts are marked with either a raised dash or asterisk; corrosion resistant steel is indicated by a single raised dash; and AN aluminum alloy bolts are marked with two raised dashes. Additional information, such as bolt diameter, bolt length, and grip length may be obtained from the bolt part number.

For example, in the bolt part number AN3DD5A, the "AN" designates that it is an Air Force-Navy Standard bolt, the "3" indicates the diameter in sixteenths of an inch (3/16), the "DD" indicates the material is 2024 aluminum alloy. The letter "C" in place of the "DD" would indicate corrosion resistant steel, and the absence of the letters would indicate cadmium plated steel. The "5" indicates the length in eighths of an inch (5/8), and the "A" indicates that the shank is undrilled. If the letter "H" preceded the "5" in addition to the "A" following it, the head would be drilled for safetying.

Close tolerance NAS bolts are marked with either a raised or recessed triangle. The material markings for NAS bolts are the same as for AN bolts, except that they may be either raised or recessed. Bolts inspected magnetically (Magnaflux) or by fluorescent means (Zyglo) are identified by means of colored lacquer, or a head marking of a distinctive type.

GENERAL PURPOSE BOLTS

The hex head aircraft bolt (AN-3 through AN-20) is an all-purpose structural bolt used for general applications involving tension or shear loads where a light drive fit is permissible (0.006-inch clearance for a 5/8-inch hole, and other sizes in proportion). Alloy steel bolts smaller than No. 10-32 and aluminum alloy bolts smaller than 1/4 inch in diameter are not used in primary structures. Aluminum alloy bolts and nuts are not used where they will be repeatedly removed for purposes of maintenance and inspection. Aluminum alloy nuts may be used with cadmium-plated steel bolts loaded in shear on land airplanes, but are not used on seaplanes due to the increased possibility of dissimilar metal corrosion.

The AN-73 drilled head bolt is similar to the standard hex bolt, but has a deeper head which is drilled to receive wire for safetying. The AN-3 and the AN-73 series bolts are interchangeable, for all practical purposes, from the standpoint of tension and shear strengths.

CLOSE TOLERANCE BOLTS

This type of bolt is machined more accurately than the general purpose bolt. Close tolerance bolts may be hex headed (AN-173 through AN-186) or have a 100° countersunk head (NAS-80 through NAS-86). They are used in applications where a tight drive fit is required. (The bolt will move into position only when struck with a 12 to 14 ounce hammer.)

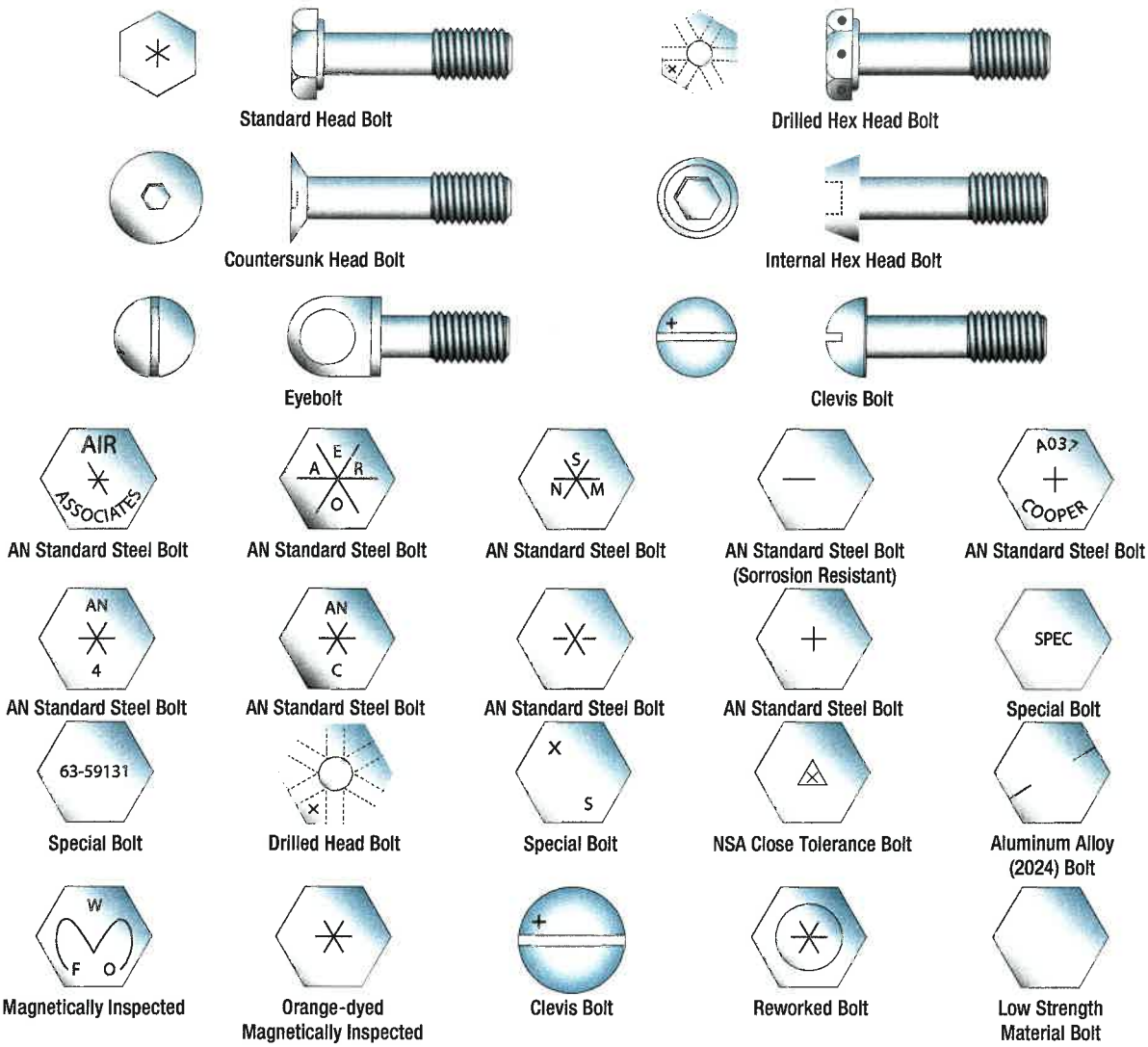


Figure 5-8. Aircraft bolt identification.

INTERNAL WRENCHING BOLTS

These bolts, (MS-20004 through MS-20024 or NAS-495) are fabricated from high-strength steel and are suitable for use in both tension and shear applications. When they are used in steel parts, the bolt hole must be slightly countersunk to seat the large corner radius of the shank at the head. In Dural material, a special heat treated washer must be used to provide an adequate bearing surface for the head. The head of the internal wrenching bolt is recessed to allow the insertion of an internal wrench when installing or removing the bolt. Special high strength nuts are used on these bolts. Replace an internal wrenching bolt with another internal wrenching bolt. Standard AN hex head bolts and washers cannot be substituted for them as they do not have the required strength.

SPECIAL-PURPOSE BOLTS

Bolts designed for a particular application or use are classified as special purpose bolts. Clevis bolts, eyebolts, Jo-Bolts, and lock-bolts are special purpose bolts.

Clevis Bolts

The head of a clevis bolt is round and is either slotted to receive a common screwdriver or recessed to receive a crosspoint screwdriver. This type of bolt is used only where shear loads occur and never in tension. It is often inserted as a mechanical pin in a control system.

Eyebolt

This type of special purpose bolt is used where external tension loads are to be applied. The eyebolt is designed for the attachment of such devices as the fork of a turnbuckle, a clevis, or a cable shackle. The threaded end may or may not be drilled for safetying.

Jo-Bolt

Jo-bolt is a trade name for an internally threaded three-piece blind fastener. The Jo-bolt consists of three parts: a threaded steel-alloy bolt, a threaded steel nut, and an expandable stainless steel sleeve. (***Figure 5-9***)



Figure 5-9. A Jo-Bolt fastener.

The parts are factory preassembled. As the Jo-bolt is installed, the bolt is turned while the nut is held. This causes the sleeve to expand over the end of the nut, forming the blind head and clamping against the work. When driving is complete, a portion of the bolt breaks off. The high shear and tensile strength of the Jo-bolt makes it suitable for use in cases of high stresses where some of the other blind fasteners would not be practical. Jo-bolts are often a part of the permanent structure of late model aircraft. They are used in areas that are not often subjected to replacement or servicing. (Because it is a three-part fastener, it should not be used where any part, in becoming loose, could be drawn into the engine air intake.) Other advantages of using Jo-bolts are their excellent resistance to vibration, weight saving, and fast installation by one person.

Presently, Jo-bolts are available in four diameters:

- 200 series, approximately 3/16 inch in diameter.
- 260 series, approximately 1/4 inch in diameter.
- 312 series, approximately 5/16 inch in diameter.
- 375 series, approximately 3/8 inch in diameter.

Jo-bolts are available in three head styles: F (flush), P (hex head), and FA (flush millable).

Lockbolts

Lockbolts are used to attach two materials permanently. They are lightweight and are equal in strength to standard bolts. Lockbolts are manufactured by several companies and conform to Military Standards. Military Standards specify the size of a lockbolt's head in relation to the shank diameter, plus the alloy used in its construction. The only drawback to lockbolt installations is that they are not easily removable compared to nuts and bolts.

The lockbolt combines the features of a high strength bolt and rivet, but it has advantages over both. The lockbolt is generally used in wing splice fittings, landing gear fittings, fuel cell fittings, longerons, beams, skin splice plates, and other major structural attachments. It is more easily and quickly installed than the conventional rivet or bolt and eliminates the use of lockwashers, cotter pins, and special nuts. Like the rivet, the lockbolt requires a pneumatic hammer or "pull gun" for installation; when installed, it is rigidly and permanently locked in place. Three types of lockbolts are commonly used: the pull type, the stump type, and the blind type. (***Figure 5-10***)

Pull type lockbolts are used mainly in aircraft primary and secondary structures. They are installed very rapidly and have approximately one-half the weight of equivalent AN steel bolts and nuts. A special pneumatic "pull gun" is required to install this type of lockbolt. Installation can be accomplished by one person since bucking is not required.

Stump type lockbolts, although they do not have the extended stem with pull grooves, are companion fasteners to pull type lockbolts. They are used primarily where clearance will not permit installation of the pull type lockbolt. A standard pneumatic riveting hammer (with a hammer set attached for swaging the collar into the pin locking grooves) and a bucking bar are tools necessary for the installation of stump type lockbolts.

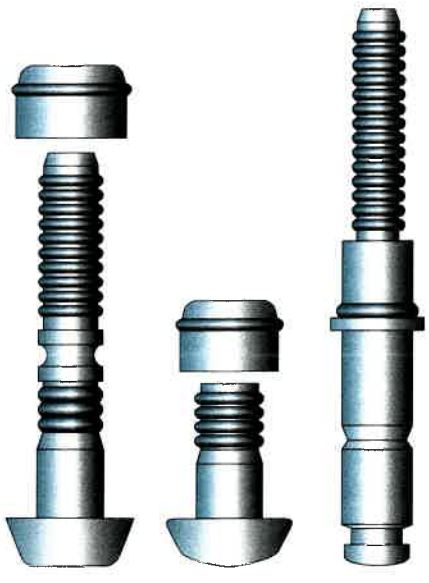


Figure 5-10. Lockbolt types.

Blind type lockbolts come as complete units or assemblies. They have exceptional strength and sheet pull together characteristics. Blind lockbolts are used where only one side of the work is accessible and, generally, where it is difficult to drive a conventional rivet. This type of lockbolt is installed in the same manner as the pull type lockbolt.

Common features of the three types of lockbolts are the annular locking grooves on the pin and the locking collar which is swaged into the pin's lock grooves to lock the pin in tension. The pins of the pull and blind type lockbolts are extended for pull installation. The extension is provided with pulling grooves and a tension breakoff groove.

The pins of pull and stump type lockbolts are made of heat treated alloy steel or high strength aluminum alloy. Companion collars are made of aluminum alloy or mild steel. The blind lockbolt consists of a heat-treated alloy steel pin, blind sleeve and filler sleeve, mild steel collar, and carbon steel washer.

Alloy steel lockbolts may be used to replace steel high-shear rivets, solid steel rivets, or AN bolts of the same diameter and head type. Aluminum alloy lockbolts may be used to replace solid aluminum alloy rivets of the same diameter and head type.

Steel and aluminum alloy lockbolts may also be used to replace steel and 2024T aluminum alloy bolts, respectively, of the same diameter. Blind lockbolts may be used to replace solid aluminum alloy rivets, stainless steel rivets, or all blind rivets of the same diameter. The numbering systems for the various types of lockbolts are explained by the breakouts in ***Figure 5-11***.

To determine the bolt grip range required for any application, measure the thickness of the material with a hook scale inserted through the hole. Once this measurement is determined, select the correct grip range by referring to the charts provided by the rivet manufacturer. Examples of grip range charts are shown in ***Figures 5-12 and 5-13***.

When installed, the lockbolt collar should be swaged substantially throughout the complete length of the collar. The tolerance of the broken end of the pin relative to the top of the collar must be within the dimensions given in ***Figure 5-14***.

When removal of a lockbolt becomes necessary, remove the collar by splitting it axially with a sharp, cold chisel. Be careful not to break out or deform the hole. The use of a backup bar on the opposite side of the collar being split is recommended. The pin may then be driven out with a drift punch.

Pull-type Lockbolt

ALPP H T 8 8

- ALPP — Head type
  - ACT509 = close tolerance AN-509 C-sink head
  - ALPP = pan head
  - ALPB = brazier head
  - ALP509 = standard AN-509 C-sink head
  - ALP426 = standard AN-426 C-sink head
- H — Class Fit
  - H = hole filling (interference fit)
  - N = non-hole filling (clearance fit)
- T — Pin Materials
  - E = 75S-T6 aluminum alloy
  - T = heat-treated alloy steel
- 8 — Body diameter in 32nds of an inch
- 8 — Grip length in 16ths of an inch

Blind-type Lockbolt

BL 8 4

- BL — Blind lockbolt
- 8 — Diameter in 32nds of an inch
- 4 — Grip length in 16ths of an inch, ± 1/32 inch

Lockbolt Collar

LC C C

- LC — Lockbolt collar
- C — Material
  - C = 24ST aluminum alloy (green color). Use with heat-treated alloy lockbolts only.
  - F = 61ST aluminum alloy (plain color). Use with 75ST aluminum alloy lockbolts only.
  - R = mild steel (cadmium plated). Use with heat-treated alloy steel lockbolts for high temperature applications only.
- C — Diameter of a pin in 32nds of an inch

Stump-type Lockbolt

ALSF E 8 8

- ALSF — Head type
  - ASCT509 = close tolerance AN-509 C-sink head
  - ALSF = flathead type
  - ALS509 = standard AN-509 C-sink head
  - ALS426 = standard AN-426 C-sink head
- E — Pin materials
  - E = 75S-T6 aluminum alloy
  - T = heat-treated alloy steel
- 8 — Body diameter in 32nds of an inch
- 8 — Grip length in 16ths of an inch

Figure 5-11. Lockbolt numbering system.

Grip No.	Grip Range Min.	Grip Range Max.	Grip No.	Grip Range Min.	Grip Range Max.
1	.031	.094	17	1.031	1.094
2	.094	.156	18	1.094	1.156
3	.156	.219	19	1.156	1.219
4	.219	.281	20	1.219	1.281
5	.281	.344	21	1.281	1.344
6	.344	.406	22	1.344	1.406
7	.406	.469	23	1.406	1.469
8	.469	.531	24	1.469	1.531
9	.531	.594	25	1.531	1.594
10	.594	.656	26	1.594	1.656
11	.656	.718	27	1.656	1.718
12	.718	.781	28	1.718	1.781
13	.781	.843	29	1.781	1.843
14	.843	.906	30	1.843	1.906
15	.906	.968	31	1.906	1.968
16	.968	1.031	32	1.968	2.031
			33	2.031	2.094

Figure 5-12. Pull and stump-type lockbolt grip ranges.

Grip No.	1/4 Inch Diameter Grip Range Min.	1/4 Inch Diameter Grip Range Max.	Grip No.	5/16 Inch Diameter Grip Range Min.	5/16 Inch Diameter Grip Range Max.
1	.031	.094	2	.094	.156
2	.094	.156	3	.156	.219
3	.156	.219	4	.219	.281
4	.219	.281	5	.281	.344
5	.281	.344	6	.344	.406
6	.344	.406	7	.406	.469
7	.406	.469	8	.469	.531
8	.469	.531	9	.531	.594
9	.531	.594	10	.594	.656
10	.594	.656	11	.656	.718
11	.656	.718	12	.718	.781
12	.718	.781	13	.781	.843
13	.781	.843	14	.843	.906
14	.843	.906	15	.906	.968
15	.906	.968	16	.968	1.031
16	.968	1.031	17	1.031	1.094
17	1.031	1.094	18	1.094	1.156
18	1.094	1.156	19	1.156	1.219
19	1.156	1.219	20	1.219	1.281
20	1.219	1.281	21	1.281	1.343
21	1.281	1.343	22	1.343	1.406
22	1.344	1.406	23	1.406	1.469
23	1.406	1.469	24	1.460	1.531
24	1.469	1.531			
25	1.531	1.594			

Figure 5-13. Blind-type lockbolt grip ranges.

Pin Diameter	Tolerance Below		Above
3/16	0.079	to	0.032
1/4	0.079	to	0.050
5/16	0.079	to	0.050
3/8	0.079	to	0.060

Figure 5-14. Pin tolerance ranges.

NUTS: SELF-LOCKING, ANCHOR, STANDARD TYPES

AIRCRAFT NUTS

Aircraft nuts are made in a variety of shapes and sizes. They are made of cadmium plated carbon steel, stainless steel, or anodized 2024T aluminum alloy, and may be obtained with either right or left hand threads. No identifying marking or lettering appears on nuts. They can be identified only by the characteristic metallic luster or color of the aluminum, brass, or the insert when the nut is of the self-locking type. They can be further identified by their construction.

Aircraft nuts can be divided into two general groups: non-self-locking and self-locking nuts. Non-self-locking nuts are those that must be safetied by external locking devices, such as cotter pins, safety wire, or locknuts. self-locking nuts contain the locking feature as an integral part.

IDENTIFICATION AND CODING

Part numbers designate the type of nut. The common types and their respective part numbers are: Plain, AN315 and AN335; castle AN310; plain check, AN316; light hex, AN340 and AN345; and castellated shear, AN320. The patented self-locking types are assigned part numbers ranging from MS20363 through MS20367. The Boots, the Flexloc, the fiber locknut, the elastic stop nut, and the self-locking nut belong to this group. Part number AN350 is assigned to the wing nut.

Letters and digits following the part number indicate such items as material, size, threads per inch, and whether the thread is right or left hand. The letter "B" following the part number indicates the nut material

to be brass, a "D" indicates 2017-T aluminum alloy, a "DD" indicates 2024-T aluminum alloy, a "C" indicates stainless steel, and a dash in place of a letter indicates cadmium plated carbon steel.

The digit (or two digits) following the dash or the material code letter is the dash number of the nut, and it indicates the size of the shank and threads per inch of the bolt on which the nut will fit. The dash number corresponds to the first figure appearing in the part number coding of general purpose bolts. A dash and the number 3, for example, indicates that the nut will fit an AN3 bolt (10-32); a dash and the number 4 means it will fit an AN4 bolt (¼-28); a dash and the number 5, an AN5 bolt (5⁄16-24); and so on.

The code numbers for self-locking nuts end in three or four digit numbers. The last two digits refer to threads per inch, and the one or two preceding digits stand for the nut size in 16ths of an inch.

Some other common nuts and their code numbers are:
Code Number AN310D5R:
AN310 = aircraft castle nut
D = 2024-T aluminum alloy
5 = inch diameter
R = right-hand thread (usually 24 threads per inch)

Code Number AN320-10:
AN320 = aircraft castellated shear nut, cadmium plated carbon steel
10 = ⅝ inch diameter, 18 threads per inch
(this nut is usually right-hand thread)

Code Number AN350B1032:
AN350 = aircraft wing nut
B = brass
10 = number 10 bolt
32 = threads per inch

NON-SELF-LOCKING NUTS

Most of the familiar types of nuts, including the plain nut, the castle nut, the castellated shear nut, the plain hex nut, the light hex nut, and the plain checknut are the non-self-locking type. (***Figure 5-15***)

The castle nut, AN310, is used with drilled shank AN hex head bolts, clevis bolts, eyebolts, drilled head bolts, or studs. It is fairly rugged and can withstand large tensional loads. Slots (called castellations) in the nut are designed to accommodate a cotter pin or lockwire for safety.

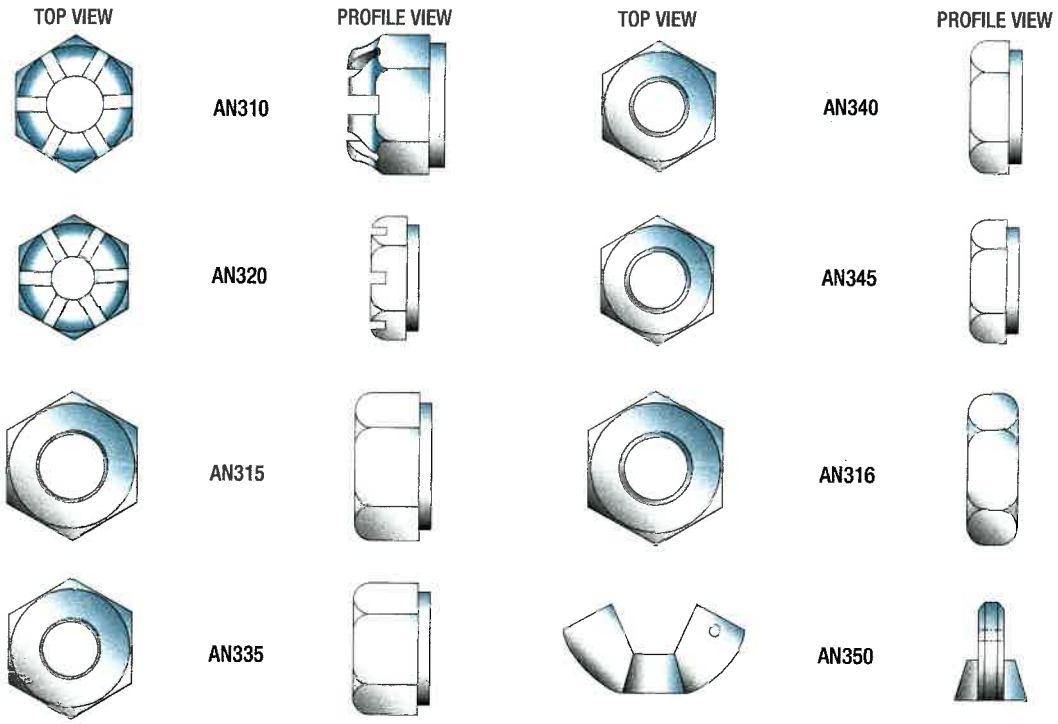


Figure 5-15. Non-self-locking nuts.

The castellated shear nut, AN320, is designed for use with devices (such as drilled clevis bolts and threaded taper pins) which are normally subjected to shearing stress only. Like the castle nut, it is castellated for safetying. Note, however, that the nut is not as deep or as strong as the castle nut; also that the castellations are not as deep as those in the castle nut.

The plain hex nut, AN315 and AN335 (fine and coarse thread), is of rugged construction. This makes it suitable for carrying large tensional loads. However, since it requires an auxiliary locking device, such as a checknut or lockwasher, its use on aircraft structures is somewhat limited. The light hex nut, AN340 and AN345 (fine and coarse thread), is a much lighter nut than the plain hex nut and must be locked by an auxiliary device. It is used for miscellaneous light tension requirements. The plain checknut, AN316, is employed as a locking device for plain nuts, set screws, threaded rod ends, and other devices. The wing nut, AN350, is intended for use where the desired tightness can be obtained with the fingers and where the assembly is frequently removed.

SELF-LOCKING NUTS

As their name implies, self-locking nuts need no auxiliary means of safetying but have a safetying feature included as an integral part of their construction. Many types of self-locking nuts have been designed and their use has become quite widespread. Common applications are: (1) attachment of antifriction bearings and control pulleys; (2) attachment of accessories, anchor nuts around inspection holes and small tank installation openings; and (3) attachment of rocker box covers and exhaust stacks. Self-locking nuts are acceptable for use on certificated aircraft subject to the restrictions of the manufacturer.

Self-locking nuts are used on aircraft to provide tight connections which will not shake loose under severe vibration. Do not use self-locking nuts at joints which subject either the nut or bolt to rotation. They may be used with antifriction bearings and control pulleys, provided the inner race of the bearing is clamped to the supporting structure by the nut and bolt. Plates must be attached to the structure in a positive manner to eliminate rotation or misalignment when tightening the bolts or screws.

The two general types of self-locking nuts currently in use are the all-metal type and the fiber lock type. For the sake of simplicity, only three typical kinds of self-locking nuts are considered in this handbook: the Boots self-locking and the stainless steel self-locking nuts, representing the all metal types; and the elastic stop nut, representing the fiber insert type. (***Figure 5-16***)

Boots Self-Locking Nut

The Boots self-locking nut is of one piece, all metal construction, designed to hold tight in spite of severe vibration. Note in ***Figure 5-16*** that it has two sections and is essentially two nuts in one, a locking nut and a load-carrying nut. The two sections are connected with a spring which is an integral part of the nut.

The spring keeps the locking and load carrying sections such a distance apart that the two sets of threads are out of phase; that is, so spaced that a bolt which has been screwed through the load carrying section must push the locking section outward against the force of the spring to engage the threads of the locking section properly.

Thus, the spring, through the medium of the locking section, exerts a constant locking force on the bolt in the same direction as a force that would tighten the nut. In this nut, the load carrying section has the thread strength of a standard nut of comparable size, while the locking section presses against the threads of the bolt and locks the nut firmly in position. Only a wrench applied to the nut will loosen it. The nut can be removed and reused without impairing its efficiency.

Boots self-locking nuts are made with three different spring styles and in various shapes and sizes. The wing type, which is the most common, ranges in size for No. 6

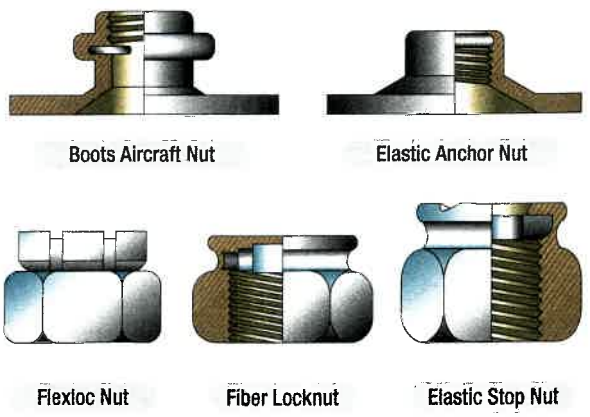


Figure 5-16. Self-locking nuts.

up to ¼ inch, the Rol-top ranges from ¼ inch to ⅛ inch, and the bellows type ranges in size from No. 8 up to ⅜ inch. Wing nuts are made of anodized aluminum alloy, cadmium-plated carbon steel, or stainless steel. The Rol-top nut is cadmium-plated steel, and the bellows type is made of aluminum alloy only.

Stainless Steel Self-Locking Nuts

The stainless steel self-locking nut may be spun on and off with the fingers, as its locking action takes place only when the nut is seated against a solid surface and tightened. The nut consists of two parts: a case with a beveled locking shoulder and key, and a threaded insert with a locking shoulder and slotted keyway. Until the nut is tightened, it spins on the bolt easily because the threaded insert is the proper size for the bolt. However, when the nut is seated against a solid surface and tightened, the locking shoulder of the insert is pulled downward and wedged against the locking shoulder of the case. This action compresses the threaded insert and causes it to clench the bolt tightly. The cross-sectional view in ***Figure 5-17*** shows how the key of the case fits into the slotted keyway of the insert so that when the case is turned, the threaded insert is turned with it. Note that the slot is wider than the key. This permits the slot to be narrowed and the insert to be compressed when the nut is tightened.

Elastic Stop Nuts

The elastic stop nut is a standard nut with the height increased to accommodate a fiber locking collar. This fiber collar is very tough and durable and is unaffected by immersion in hot or cold water or ordinary solvents, such as ether, carbon tetrachloride, oils, and gasoline. It will not damage bolt threads or plating.

As shown in ***Figure 5-18***, the fiber locking collar is not threaded and its inside diameter is smaller than the largest diameter of the threaded portion or the outside diameter of a corresponding bolt. When the nut is screwed onto a bolt, it acts as an ordinary nut until the bolt reaches the fiber collar. When the bolt is screwed into the fiber collar, however, friction (or drag) causes the fiber to be pushed upward. This creates a heavy downward pressure on the load carrying part and automatically throws the load carrying sides of the nut and bolt threads into positive contact. After the bolt has been forced all the way through the fiber collar, the downward pressure remains constant. This pressure locks and holds the nut securely in place even under severe vibration.

Nearly all elastic stop nuts are steel or aluminum alloy. However, such nuts are available in practically any kind of metal. Aluminum alloy elastic stop nuts are supplied with an anodized finish. Steel nuts are cadmium plated.

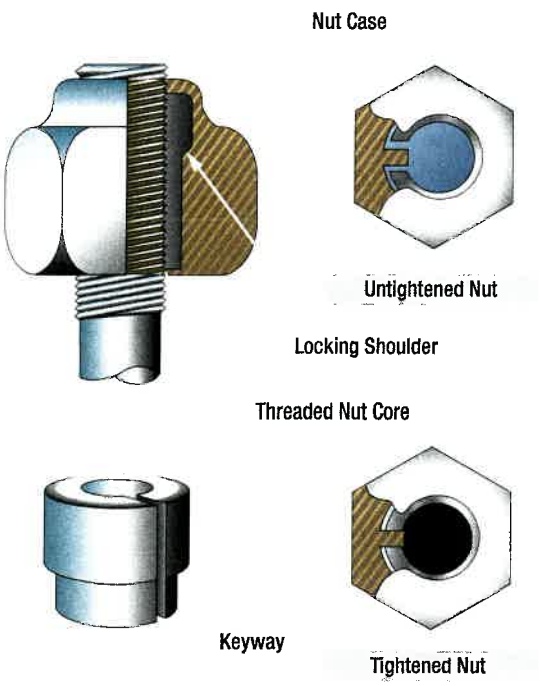


Figure 5-17. Stainless steel self-locking nut.

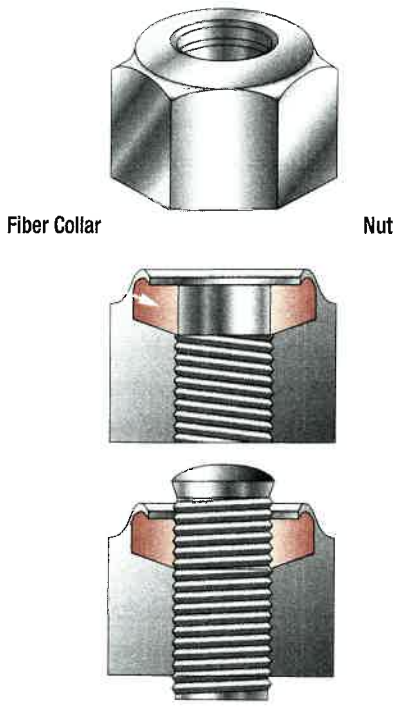


Figure 5-18. Elastic stop nut.

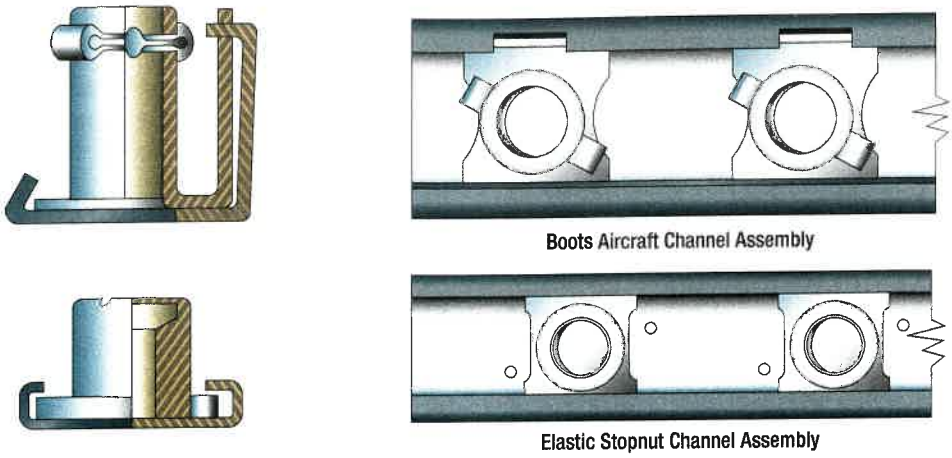


Figure 5-19. Self-locking nut bases.

Normally, elastic stop nuts can be used many times with complete safety and without detriment to their locking efficiency. When reusing elastic stop nuts, be sure the fiber has not lost its locking friction or become brittle. If a nut can be turned with the fingers, replace it.

After the nut has been tightened, make sure the rounded or chamfered end of the bolts, studs, or screws extends at least the full round or chamfer through the nut. Flat end bolts, studs, or screws should extend at least 1/32 inch through the nut. Bolts of 5/16 inch diameter and over with cotter pin holes may be used with self-locking nuts, but only if free from burrs around the holes. Bolts with damaged threads and rough ends are not acceptable. Do not tap the fiber locking insert. The self-locking action of the elastic stop nut is the result of having the bolt threads impress themselves into the untapped fiber.

Do not install elastic stop nuts in places where the temperature is higher than 120°C, because the effectiveness of the self-locking action is reduced beyond this point. Self-locking nuts may be used on aircraft engines and accessories when their use is specified by the engine manufacturer. Self-locking nut bases are made in a number of forms and materials for riveting and welding to aircraft structure or parts. (***Figure 5-19***)

Certain applications require the installation of self-locking nuts in channels, an arrangement which permits the attachment of many nuts with only a few rivets. These channels are track like bases with regularly spaced nuts which are either removable or non-removable. The removable type carries a floating nut, which can be snapped in or out of the channel, thus making possible the easy removal of damaged nuts. Nuts, such as the clinch type and spline type, which depend on friction for their anchorage, are not acceptable for use in aircraft structures.

Pal Nuts

Palnut is a registered trademark of Trans Technology. They are also identified as MS27151 series nuts when they are cadmium plated. A pal nut is a stamped sheet metal nut used in the same manner as a jam nut or a check nut but it is lighter in weight. (***Figure 5-20***)

Sheet Spring Nuts

Sheet spring nuts, such as speed nuts, are used with standard and sheet metal self-tapping screws in nonstructural locations. They find various uses in supporting line clamps, conduit clamps, electrical equipment, access doors, and the like, and are available in several types.



Figure 5-20. A group of Pal nuts.

Speed nuts are made from spring steel and are arched prior to tightening. This arched spring lock prevents the screw from working loose. These nuts should be used only where originally used in the fabrication of the aircraft.

Internal and External Wrenching Nuts

Two commercial types of high strength internal or external wrenching nuts are available; they are the internal and external wrenching elastic stop nut and the Unbrako internal and external wrenching nut. Both are of the self-locking type, are heat treated, and are capable of carrying high strength bolt tension loads.

ANCHOR NUTS

Self-locking nut bases are made in a number of forms and materials for riveting and welding to aircraft structure or parts. These are called anchor nuts. Anchor nuts are commonly used on aircraft since they require the technician only have access to a single side of the work (i.e. the side that a wrench, socket or screwdriver can be used). With the nut anchored in place, installation of a bolt or machine screw can be accomplished without holding the nut. (***Figure 5-21***)

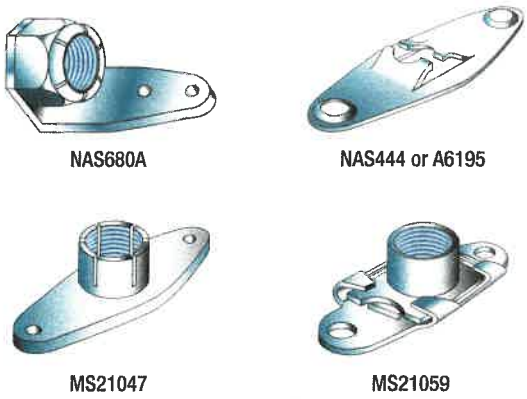


Figure 5-21. Examples of anchor nuts.

Certain applications require the installation of self-locking nuts in channels, an arrangement which permits the attachment of many nuts with only a few rivets. These channels are track-like bases with regularly spaced nuts which are either removable or non-removable. The removable type carries a floating nut, which can be snapped in or out of the channel, thus making possible the easy replacement of damaged nuts. (***Figure 5-19***) Nuts such as the clinch type or the spline type, which depend on friction for their anchorage, are not acceptable for use in aircraft structures.

NUTPLATES

Nuts that are made to be riveted in place in aircraft are called nutplates. Their purpose is to allow bolts and screws to be inserted without having to hold the nut. They are permanently mounted to enable inspection panels and access doors to be easily removed and installed. When many screws are used on a panel, to make installation easier, normally floating anchor nuts are used. The floating anchor nut fits into a small bracket which is riveted to the aircraft skin. The nut is free to move, which makes it much easier to align it with the screw. For production ease, sometimes ganged anchor nuts are used for inspection panels. Ganged anchor nuts allow the nuts to float in a channel, making alignment with the screw easy.

Self-locking nutplates are made under several standards and come in several shapes and sizes. ***Figure 5-22*** shows an MS21078 two lug nutplate with a non-metallic insert, and an MS21047 lightweight, all-metal nutplate. Nutplates can also have three riveting points if the added strength is required.

Sealing Nutplates

When securing nutplates in pressurized aircraft and in fuel cells, a sealing nutplate is used instead of the open ended variety previously described. Care must be taken to use exactly the correct length of bolt or screw. If a bolt or screw is too short, there will not be enough threads to hold the device in place. If the bolt or screw is too long, it will penetrate the back side of the nutplate and compromise the seal. Normally, a sealant is also used to ensure complete sealing of the nutplate. Check the manufacturer's specifications for the acceptable sealant to be used for sealing nutplates.

Riveted and Rivetless Nutplates

When access to the back of a screw or bolt installation is impractical, riveted or rivetless nutplates are used to secure the connection of panels. One example in aircraft this technique is especially useful is to secure the floorboards to the stringers and to each other.

RIVNUTS

This is the trade name of a hollow, blind rivet made of 6053 aluminum alloy, counterbored and threaded on the inside. Rivnuts can be installed by one person using a special tool which heads the rivet on the blind side of the

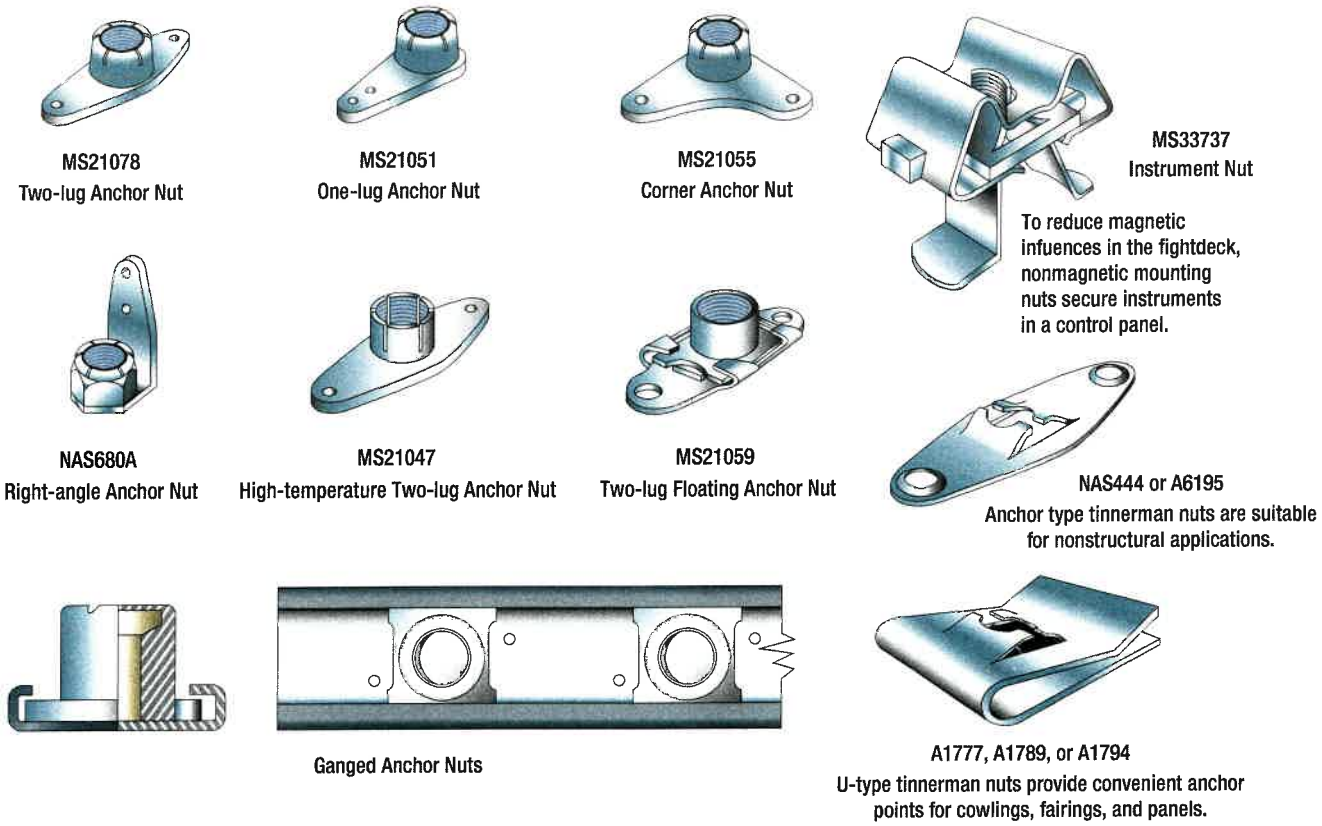


Figure 5-22. Various nutplates.

material. The Rivnut is threaded on the mandrel of the heading tool and inserted in the rivet hole. The heading tool is held at right angles to the material, the handle is squeezed, and the mandrel crank is turned clockwise after each stroke. Continue squeezing the handle and turning the mandrel crank of the heading tool until a solid resistance is felt, which indicates that the rivet is set.

The Rivnut is used primarily as a nut plate and in the attachment of deicer boots to the leading edges of wings. It may be used as a rivet in secondary structures or for the attachment of accessories such as brackets, fairings, instruments, or soundproofing materials.

Rivnuts are manufactured in two head types, each with two ends: the flathead with open or closed end, and the countersunk head with open or closed end. All Rivnuts, except the thin head countersunk type, are available with or without small projections (keys) attached to the head to keep the Rivnut from turning. Keyed Rivnuts are used as a nut plate, while those without keys are used for straight blind riveting repairs where no torque loads are imposed. A keyway cutter is needed when installing Rivnuts which have keys.

The countersunk style Rivnut is made with two different head angles: 100° with 0.048 and 0.063 inch head thickness, and 115° with 0.063 inch head thickness. Each of these head styles is made in three sizes: 6-32, 8-32, and 10-32. These numbers represent the machine screw size of the threads on the inside of the Rivnut. The actual outside diameters of the shanks are 3/16 in for the 6-32 size, 7/32 in for the 8-32 size, and 1/4 in for the 10-32 size.

Open end Rivnuts are the most widely used and are recommended in preference to the closed end type wherever possible. However, closed end Rivnuts must be used in pressurized compartments.

Rivnuts are manufactured in six grip ranges. The minimum grip length is indicated by a plain head, and the next higher grip length by one radial dash mark on the head. Each succeeding grip range is indicated by an additional radial dash mark until five marks indicate the maximum range.

Notice in ***Figure 5-23*** that some part number codes consist of a "6," an "8," or a "10," a "dash," and two or three more numbers. In some, the dash is replaced by the letters "K" or "KB." The first number indicates the machine screw size of the thread, and the last two or three numbers indicate the maximum grip length in thousandths of an inch. A dash between the figures indicates that the Rivnut has an open end and is keyless; a "B" in place of the dash means it has a closed end and is keyless; a "K" means it has an open end and has a key; and a "KB" indicates that it has a closed end and a key. If the last two or three numbers are divisible by five, the Rivnut has a flathead; if they are not divisible by five, the Rivnut has a countersunk head.

An example of a part number code is:
10KB106
10 = Grip length
KB = Closed end and key
106 = Screw and thread size

MACHINE SCREWS; AIRCRAFT SPECIFICATION

SCREWS

Screws are the most commonly used threaded fastening devices on aircraft. They differ from bolts inasmuch as they are generally made of lower strength materials. They can be installed with a loose fitting thread, and the head shapes are made to engage a screwdriver or wrench. Some screws have a clearly defined grip or unthreaded portion while others are threaded along their entire length.

Several types of structural screws differ from the standard structural bolts only in head style. The material is the same, and a definite grip length is provided. The AN525 washer head screw and the NAS220 through NAS227 series are such screws.

Commonly used screws are classified in three groups: (1) structural screws, which have the same strength as equal size bolts; (2) machine screws, which include the majority of types used for general repair; and (3) selftapping screws, which are used for attaching lighter parts. A fourth group, drive screws, are not actually screws but nails. They are driven into metal parts with a mallet or hammer and their heads are not slotted or recessed.



Flat—0.32 Head Thickness		
6-45	6-75	6-100
8-45	8-75	8-100
10-45	10-75	10-100
6B45	6B75	6B100
8B45	8B75	8B100
10B45	10B75	10B100
6K45	6K75	6K100
8K45	8K75	8K100
10K45	10K75	10K100
6KB45	6KB75	6KB100
8KB45	8KB75	8KB100
10KB45	10KB75	10KB100
100°—0.48 Head Thickness		
6-91	6-121	6-146
8-91	8-121	8-146
10-91	10-121	10-146
6B91	6B121	6B146
8B91	8B121	8B146
10B91	10B121	10B146
100°—0.63 Head Thickness		
6-106	6-136	6-161
8-106	8-136	8-161
10-106	10-136	10-161
6B106	6B136	6B161
8B106	8B136	8B161
10B106	10B136	10B161
6K106	6K136	6K161
8K106	8K136	8K161
10K106	10K136	10K161
6KB106	6KB136	6KB161
8KB106	8KB136	8KB161
10KB106	10KB136	10KB161

Figure 5-23. Rivnut data chart.

IDENTIFICATION/CODING FOR SCREWS

The coding system used to identify screws is similar to that used for bolts. There are AN and NAS screws. NAS screws are structural screws. Part numbers 510, 515, 550, and so on, catalog screws into classes, such as roundhead, flathead, washer head, and so forth. Letters and digits indicate their material composition, length, and thickness.

Examples of AN and NAS code numbers follow:

AN501B-416-7
AN = Air Force-Navy standard
501 = fillister head, fine thread
B = brass
416 = 4/16 inch diameter
7 = 7/16 inch length

The letter "D" in place of the "B" would indicate that the material is 2017-T aluminum alloy. The letter "C" would designate corrosion resistant steel. An "A" placed before the material code letter would indicate that the head is drilled for safetying:

NAS144DH-22
NAS = National Aircraft Standard
144 = head style; diameter and thread; ¼ 28 bolt, internal wrenching
DH = drilled head
22 = screw length in 16ths of an inch-1⅜ inches long

The basic NAS number identifies the part. The suffix letters and dash numbers separate different sizes, plating material, drilling specifications, and so forth. The dash numbers and suffix letters do not have standard meanings. It is necessary to refer to a specific NAS page in the Standards book for the legend.

STRUCTURAL SCREWS

Structural screws are made of alloy steel, are properly heat treated, and can be used as structural bolts. These screws are found in the NAS204 through NAS235 and AN509 and AN525 series. They have a definite grip and the same shear strength as a bolt of the same size. Shank tolerances are similar to AN hex head bolts, and the threads are National Fine. Structural screws are available with round, brazier, or countersunk heads. The recessed head screws are driven by either a Phillips or a Reed & Prince screwdriver.

- The AN509 (100°) flathead screw is used in countersunk holes where a flush surface is necessary.
- The AN525 washer head structural screw is used where raised heads are not objectionable. The washer head screw provides a large contact area.

MACHINE SCREWS

Machine screws are usually of the flathead (countersunk), roundhead, or washer head types. These are general purpose screws and are available in low carbon steel, brass, corrosion-resistant steel, and aluminum alloy.

Roundhead screws, AN515 and AN520, have either slotted or recessed heads. The AN515 screw has coarse threads, and the AN520 has fine threads.

Countersunk machine screws are listed as AN505 and AN510 for 82°, and AN507 for 100°. The AN505 and AN510 correspond to the AN515 and AN520 roundhead in material and usage.

The fillister head screw, AN500 through AN503, is a general purpose screw and is used as a capscrew in light mechanisms. This could include attachments of cast aluminum parts such as gearbox cover plates. The AN500 and AN501 screws are available in low carbon steel, corrosion resistant steel, and brass. The AN500 has coarse threads, while the AN501 has fine threads. They have no clearly defined grip length. *Screws larger than No. 6 have a hole drilled through the head for safetying purposes.*

The AN502 and AN503 fillister head screws are made of heat-treated alloy steel, have a small grip, and are available in fine and coarse threads. These screws are used as capscrews where great strength is required. The coarse threaded screws are commonly used as capscrews in tapped aluminum alloy and magnesium castings because of the softness of the metal.

STUDS; TYPES AND USES

Studs are metal rods that are threaded at both ends. In general they are used where it is not possible, or desirable for a bolt to be used. Like many screw types of fastener, most studs are produced in a standard form, with variants used for special purposes. For example, where a standard type is unsuitable, such as when being used in a soft metal, then a stepped stud (which has a greater holding power) would be used. A stepped stud would also be used where a damaged thread had been removed, the hole drilled out and retapped.

It will be appreciated that the security of a stud depends upon the friction between its thread and that of the tapped hole (the metal thread) into which it is inserted. If this friction fails to hold the stud, it will work loose and all precautions to prevent the nut from slackening will be negated.

STANDARD STUDS

By far the most widely used stud is the standard (plain, or parallel) type, in which the diameter of the whole stud, along its length, is constant. Standard studs are classified by the thread type, diameter and overall length. The metal thread is, usually, finished very slightly oversize to give a tight fit into the tapped hole. Other variants of the standard stud are available for use in circumstances that require special consideration. (***Figure 5-24***)



Figure 5-24. Standard stud.

To meet special requirements, the various types of standard studs may also be supplied with non-standard lengths of plain portion and metal end. A simple method of fitting and removing a stud is by running two plain nuts down the nut end of the stud and cinching (locking) them together using two spanners. The stud can then be screwed into or removed from the material. Breaking the cinch then separating and removing the nuts completes the operation.

WAISTED STUDS

Waisted studs are used where reduction of weight, without the loss of strength, is of paramount importance. The diameter of the plain portion of the stud is reduced to the minor diameter of the end threads, thus lightening the stud without impairing its effective strength. (***Figure 5-25***)



Figure 5-25. Waisted stud.



Figure 5-26. Stepped stud.

STEPPED STUDS

This type affords a stronger anchorage than the standard type, if the metal end of the stud has to be housed in soft metal. The thread of the metal end is one size larger than that of the nut end. For example, a % inch BSF stepped stud has a plain portion of % inch thread on the nut end and a 3⁄16 inch thread on the metal end.

Stepped studs are also used as replacements for standard studs when the tapped stud hole has to be redrilled and tapped with a larger thread, due to damage. (***Figure 5-26***)

SHOULDERED STUDS

This type is used where maximum rigidity of assembly is of prime importance. The stud is machined from oversize bar and a projecting shoulder is left between the metal end of the thread and the normal diameter plain portion. This shoulder seats firmly on the surface of the metal and gives additional resistance to sideways stresses. The clearance hole in the second component, through which the nut end and plain portion of the stud passes, must be machined at the inner end to give clearance to the stud shoulder. (***Figure 5-27***)

SELF-TAPPING SCREWS, DOWELS

SELF-TAPPING SCREWS

Machine self-tapping screws are listed as AN504 and AN506. The AN504 screw has a roundhead, and the AN506 is 82° countersunk. These screws are used for attaching removable parts, such as nameplates, to castings and parts in which the screw cuts its own threads.

AN530 and AN531 self-tapping sheet metal screws, such as the Parker-Kalon Z-type sheet metal screw, are blunt on the end. They are used in the temporary attachment of metal for riveting, and in the permanent assembly of nonstructural assemblies. Self-tapping screws should not be used to replace standard screws, nuts, bolts, or rivets.

DRIVE SCREWS

Drive screws, AN535, correspond to the Parker-Kalon U-type. They are plain head self-tapping screws used as capscrews for attaching nameplates in castings and for sealing drain holes in corrosion proofing tubular structures. They are not intended to be removed after installation.

DOWELS

Dowels or dowel pins are used in machinery to precisely locate two adjoining parts or assemblies. They are typically cylindrical and machined from steel but can be made from any material appropriate for a specific application. (***Figure 5-28***)

Most alignment dowels are machined to specific tight tolerances. One end of a dowel is secured in place to a part to be mated with another part. A portion of the dowel protrudes such that it can be inserted into a reamed hole in the mating part. In this fashion, the two parts are held in alignment more exactly than if the parts were simply bolted together. In aviation, sections of turbine engines bolt together axially and often use dowels to achieve the close tolerance alignment required.

Dowel pins provide positive location and alignment of fixed assemblies which otherwise may rotate if not secured. For fixed assemblies, the dowel pins are usually a push fit into reamed holes on the inner faces of the assembly. Examples of the use of dowel pins include the location for electric motors and generators which are



Figure 5-27. Shoulder stud.



Figure 5-28. Machined steel alignment dowels.

secured by V band clamps on tapered flange mountings, location of hydraulic pumps, and location of brake units on landing gear axles which simplifies and saves weight as bolts are not used.

THREAD REPAIR HARDWARE

Installation or replacement of bolts is simple when compared to the installation or replacement of studs. Bolt heads and nuts are out in the open, whereas studs are installed into internal threads in a casting or built up assembly. Damaged threads on bolts or nuts can be seen and only require replacement of the defective part. If internal threads are damaged, two alternatives are apparent: the part may be replaced or the threads repaired or replaced. Correction of the thread problem is usually cheaper and more convenient. Two methods of repairing are by replacement bushings or helicoils.

REPLACEMENT BUSHINGS

Bushings are usually special material (for example, steel or brass spark plug bushings into aluminum cylinder heads). A material that will resist wear is used where removal and replacement is frequent. The external threads on the bushing are usually coarse. The bushing is installed, a thread lock compound may or may not be used, and staked to prevent loosening. Many bushings have left hand threads external and right hand threads



Figure 5-29. Helicoil insert.

internal. With this installation, removal of the bolt or stud (right hand threads) tends to tighten the bushing. Bushings for common installations, such as spark plugs, may be up to 0.040 oversize (in increments of 0.005). Original installation and overhaul shop replacements are shrunk fit: consisting of a heated cylinder head and a frozen bushing.

HELICOILS

Helicoils are precision formed screw thread coils of 18-8 stainless steel wire having a diamond shaped cross section. (***Figure 5-29***)

They form unified coarse or unified fine thread classes when assembled into (helicoil) threaded holes. The assembled insert accommodates UNJ (controlled radius root) male threaded members. Each insert has a driving tang with a notch to facilitate removal of the tang after the insert is screwed into a helicoil tapped hole.

They are used as screw thread bushings. In addition to being used to restore damaged threads, they are used in the original design of missiles, aircraft engines, and all types of mechanical equipment and accessories to protect and strengthen tapped threads in light materials, metals, and plastics, particularly in locations which require frequent assembly and disassembly, and/or where a screw locking action is desired.

Helicoil installation is a 5 or 6 step operation, depending upon how the last step is classed. (***Figure 5-30***)

Step 1: Determine what threads are damaged.

Step 2: (a) New installation of helicoil. Drill out damaged threads to minimum depth specified. (b) Previously installed helicoil. Using proper size extracting tool, place edge of blade in 90° from the edge of the insert. Tap with hammer to seat tool. Turn to left, applying pressure, until insert backs out. Threads are not damaged if insert is properly removed.

Step 3: Tap. Use the tap of required nominal thread size. The tapping procedure is the same as standard thread tapping. Tap length must be equal to or exceed the requirement.

Step 4: Gauge. Threads may be checked with a helicoil thread gauge.

Step 5: Insert assembly. Using proper tool, install insert to a depth that puts end of top coil ¼ to ½ turn below the top surface of the tapped hole.

Step 6: Tang breakoff. Select proper breakoff tool. Tangs should be removed from all drilled through holes. In blind holes, the tangs may be removed when necessary if enough hole depth is provided below the tang of the installed insert.

These are not to be considered specific instructions on helicoil installation. The manufacturer's instruction must be followed when making an installation. Helicoils are available for the following threads: unified coarse, unified fine, metric, spark plug, and national taper pipe threads.

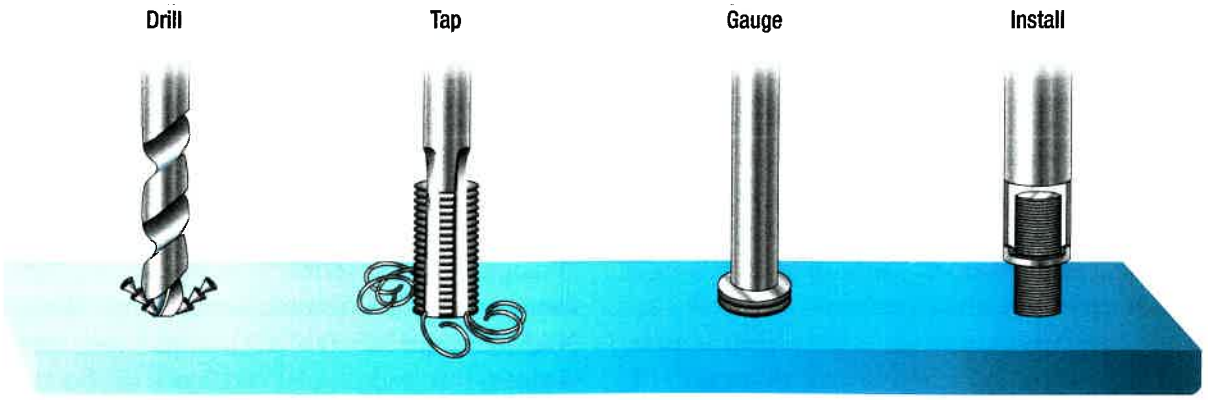


Figure 5-30. Helicoil installation.

6.5.3 - LOCKING DEVICES

To ensure fasteners do not separate from their nuts or holding ends, various safetying methods are used from heavy aircraft to gliders and recreational aircraft.

Safetying is the process of securing all aircraft, bolts, nuts, screws, pins, and other fasteners so that they do not work loose due to vibration. A familiarity with the various methods and means of safetying equipment on an aircraft is necessary in order to perform maintenance and inspection. There are various methods of safetying aircraft parts. The most widely used methods are safety wire, cotter pins, lockwashers, snaprings, and special nuts, such as self-locking nuts, pal nuts, and jamnuts. Some of these nuts and washers have been previously described in this chapter.

AIRCRAFT WASHERS

Aircraft washers used in airframe repair are either plain, lock, or special type washers. (***Figure 5-31***)

PLAIN WASHERS

Plain washers, both the AN960 and AN970, are used under hex nuts. They provide a smooth bearing surface and act as a shim in obtaining correct grip length for a bolt and nut assembly. They are used to adjust the position of castellated nuts in respect to drilled cotter pin holes in bolts. Use plain washers under lockwashers to prevent damage to the surface material.

Aluminum and aluminum alloy washers may be used under bolt heads or nuts on aluminum alloy or magnesium structures where corrosion caused by dissimilar metals is a factor. When used in this manner, any electric current flow will be between the washer and the steel bolt. However, it is common practice to use a cadmium plated steel washer under a nut bearing directly against a structure as this washer will resist the cutting action of a nut better than an aluminum alloy washer.

The AN970 steel washer provides a greater bearing area than the AN960 washer and is used on wooden structures under both the head and the nut of a bolt to prevent crushing the surface.

LOCKWASHERS

Lockwashers, both the AN935 and AN936, are used with machine screws or bolts where the self-locking or castellated type nut is not appropriate. The spring action of the washer (AN935) provides enough friction to prevent loosening of the nut from vibration. (***Figure 5-31***) Lockwashers should never be used under the following conditions:

- With fasteners to primary or secondary structures.
- With fasteners on any part of the aircraft where failure might result in damage or danger to the aircraft or personnel.
- Where failure would permit the opening of a joint to the airflow.
- Where the screw is subject to frequent removal.
- Where the washers are exposed to the airflow.
- Where the washers are subject to corrosive conditions.
- Where the washer is against soft material without a plain washer underneath to prevent gouging the surface.

SHAKEPROOF LOCKWASHERS

Shakeproof lockwashers are round washers designed with tabs or lips that are bent upward across the sides of a hex nut or bolt to lock the nut in place There are various methods of securing the lockwasher to prevent it from turning, such as an external tab bent downward 90° into a small hole in the face of the unit, or an internal tab which fits a keyed bolt.

Shakeproof lockwashers can withstand higher heat than other methods of safetying and can be used under high vibration conditions safely. They should be used only once because the tabs tend to break when bent a second time.

SPECIAL WASHERS

The ball socket and seat washers, AC950 and AC955, are special washers used where a bolt is installed at an angle to a surface, or where perfect alignment with a surface is required. These washers are used together. (***Figure 5-31***) The NAS143 and MS20002 washers are used for internal wrenching bolts of the NAS144 through NAS158 series. This washer is either plain or countersunk. The countersunk washer (designated as NAS143C and MS20002C) is used to seat the bolt head shank radius, and the plain washer is used under the nut.

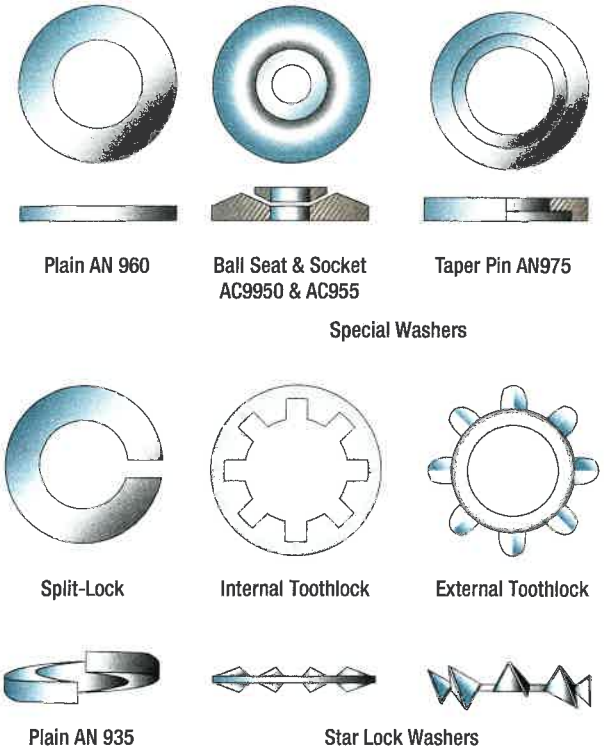


Figure 5-31. Various types of washers.

LOCKPLATES

In certain circumstances, the torque applied, the thread, or the type of nut, being used may not guarantee that the nut would not unwind in use (such as during vibration). Lock plates are used where positive retention of a nut is required.

To install a typical lockplate the nut is torque loaded and then (only if necessary) turned a small amount, (<1/12 revolution) until its flats align with the flats in the lock plate. The plate usually has 12 flats to allow for this adjustment. The plate is then placed over the nut and a small setscrew is fastened into the tapped hole adjacent to the nut. Removal of the nut simply involves removing the setscrew, lifting off the plate and unwinding the nut.

Note: A Tab washer could be used to do the same task. The lock plate is used where the nut is frequently removed. The plate can be used indefinitely providing it retains a good fit with the nut.

PINS

The three main types of pins used in aircraft structures are the taper pin, flathead pin, and cotter pin. Pins are used in shear applications and for safetying. Roll pins are finding increasing uses in aircraft construction. Taper Pins and threaded taper pins (AN385 and AN386) are used in joints which carry shear loads and where absence of play is essential. The plain taper pin is drilled and usually safetied with wire. The threaded taper pin is used with a taper pin washer (AN975) and shear nut (safetied with a cotter pin or safety clip) or self-locking nut.

Flathead Pin commonly called a clevis pin, the flathead pin (MS20392) is used with tie rod terminals and in secondary controls which are not subject to continuous operation. The pin is customarily installed with the head up so that if the cotter pin fails or works out, the pin will remain in place.

COTTER PINS

The AN380 cadmium plated, low carbon steel cotter pin is used for safetying bolts, screws, nuts, other pins, and in various applications where such safetying is necessary. The AN381 corrosion resistant steel cotter pin is used in locations where nonmagnetic material is required, or in locations where resistance to corrosion is desired.

Cotter Pin Hole Line Up

When tightening castellated nuts on bolts, the cotter pin holes may not line up with the slots in the nuts for the range of recommended values. Except in cases of highly stressed engine parts, the nut may not be over torque. Remove hardware and realign the holes. The torque loads specified may be used for all unlubricated cadmium plated steel nuts of the fine or coarse thread series which have approximately equal number of threads and equal face bearing areas. These values do not apply where special torque requirements are specified in the maintenance manual.

If the head end, rather than the nut, must be turned in the tightening operation, maximum torque values may be increased by an amount equal to shank friction, provided the latter is first measured by a torque wrench.

Cotter Pin Safetying

Cotter pin installation is shown in ***Figure 5-32***.

Castellated nuts are used with bolts that have been drilled for cotter pins. The cotter pin should fit neatly into the hole, with very little side play. The following general rules apply to cotter pin safetying:

1. The prong bent over the bolt end should not extend beyond the bolt diameter. (Cut it off if necessary.)

2. The prong bent down should not rest against the surface of the washer. (Again, cut it off if necessary.)
3. If the optional wraparound method is used, the prongs should not extend outward from the sides of the nut.
4. All prongs should be bent over a reasonable radius. Sharp angled bends invite breakage. Tapping lightly with a mallet is the best method of bending the prongs.

ROLL PINS

The roll pin is a pressed fit pin with chamfered ends. It is tubular in shape and is slotted the full length of the tube. The pin is inserted with hand tools and is compressed as it is driven into place. Pressure exerted by the roll pin against the hole walls keeps it in place, until deliberately removed with a drift punch or pin punch.

QUICK RELEASE (CAPTIVE) FASTENERS

Captive fasteners are used for quick removal of engine nacelles, inspection panels, and areas where fast and easy access is important. A captive fastener has the ability to turn in the body in which it is mounted, but which will not drop out when it is unscrewed from the part it is holding.

TURNLOCK FASTENERS

Turnlock fasteners are used to secure inspection plates, doors, and other removable panels on aircraft. Turnlock fasteners are also referred to by such terms as quick opening, quick action, and stressed panel fasteners. The most desirable feature of these fasteners is that they permit quick and easy removal of access panels for inspection and servicing purposes. Turnlock fasteners are manufactured and supplied by a number of manufacturer's under various trade names. Some of the most commonly used are the Dzus, Camloc, and Airloc.

Dzus Fasteners

The Dzus turnlock fastener consists of a stud, grommet, and receptacle. ***Figure 5-33*** illustrates an installed Dzus fastener and the various parts. The grommet is made of aluminum or aluminum alloy material. It acts as a holding device for the stud. Grommets can be fabricated from 1100 aluminum tubing, if none are available from normal sources.

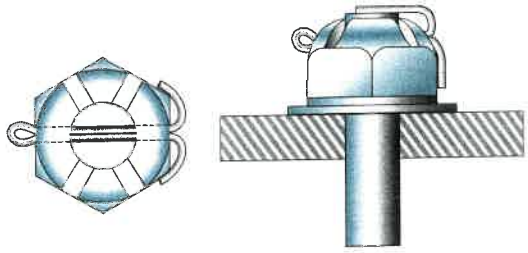


Figure 5-32. Cotter pin installations.

The spring is made of steel, cadmium plated to prevent corrosion. The spring supplies the force that locks or secures the stud in place when two assemblies are joined.

The studs are fabricated from steel and are cadmium plated. They are available in three head styles: wing, flush, and oval. Body diameter, length, and head type may be identified or determined by the markings found on the head of the stud. (***Figure 5-34***)

The diameter is always measured in sixteenths of an inch. Stud length is measured in hundredths of an inch and is the distance from the head of the stud to the bottom of the spring hole.

A quarter of a turn of the stud (clockwise) locks the fastener. The fastener may be unlocked only by turning the stud counterclockwise. A Dzus key or a specially ground screwdriver locks or unlocks the fastener.

Camloc Fasteners

Camloc fasteners are made in a variety of styles and designs. Included among the most commonly used are the 2600, 2700, 40S51, and 4002 series in the regular line, and the stressed panel fastener in the heavy duty line. The latter is used in stressed panels which carry structural loads. The Camloc fastener is used to secure aircraft cowlings and fairings. It consists of three parts: a stud assembly, a grommet, and a receptacle. Two types of receptacles are available: rigid and floating. (***Figure 5-35***)

The stud and grommet are installed in the removable portion; the receptacle is riveted to the structure of the aircraft. The stud and grommet are installed in either a plain, dimpled, countersunk, or counterbored hole, depending upon the location and thickness of the material involved. A quarter turn (clockwise) of the stud locks the fastener. The fastener can be unlocked only by turning the stud counterclockwise.

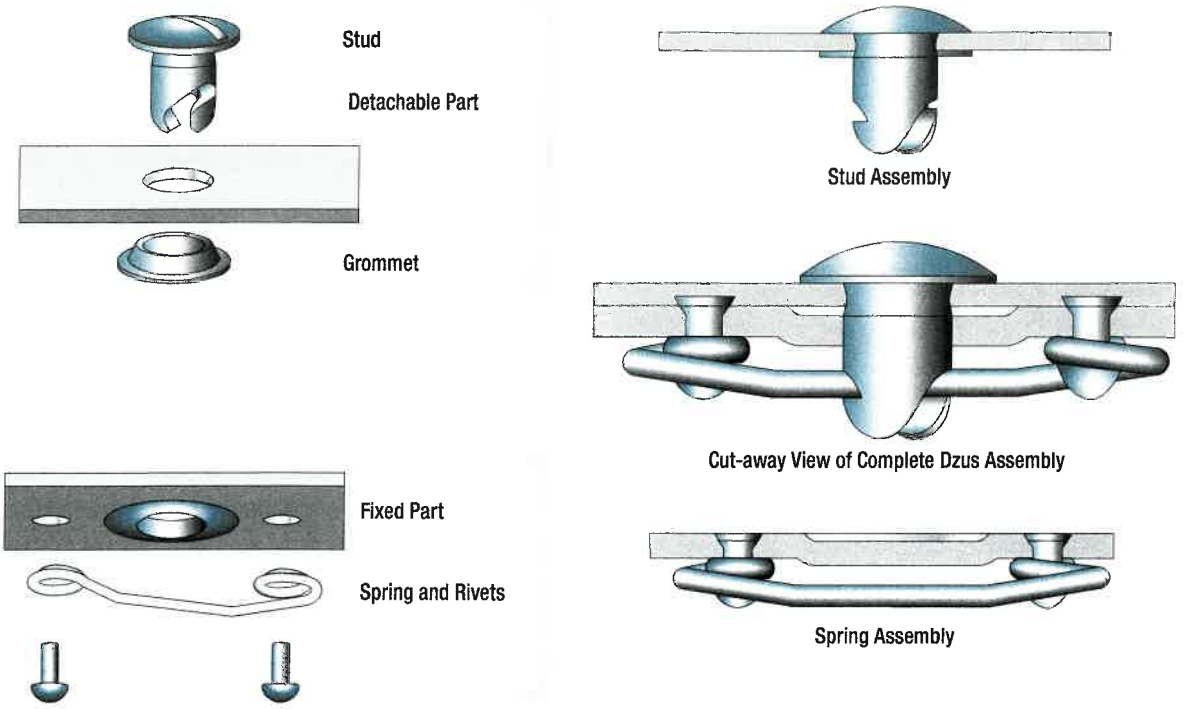


Figure 5-33. Dzus fastener.

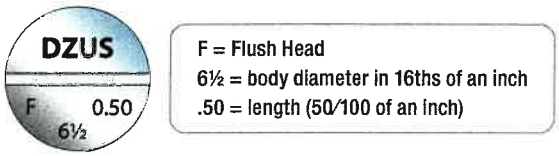


Figure 5-34. Dzus identification.

Airloc Fasteners

The Airloc fastener consists of three parts: a stud, a cross pin, and a stud receptacle. (***Figure 5-36***) The studs are manufactured from steel and casehardened to prevent excessive wear. The stud hole is reamed for a press fit of the cross pin. The total amount of material thickness to be secured with the Airloc fastener must be known before the correct length of stud can be selected for installation. The total thickness of material that each stud will satisfactorily lock together is stamped on the head of the stud in thousandths of an inch (0.040, 0.070, 0.190, and so forth). Studs are manufactured in three head styles: flush, oval, and wing.

The cross pin (***Figure 5-36***) is manufactured from chrome-vanadium steel and heat treated to provide maximum strength, wear, and holding power. It should never be used the second time; once removed from the stud, replace it with a new pin.

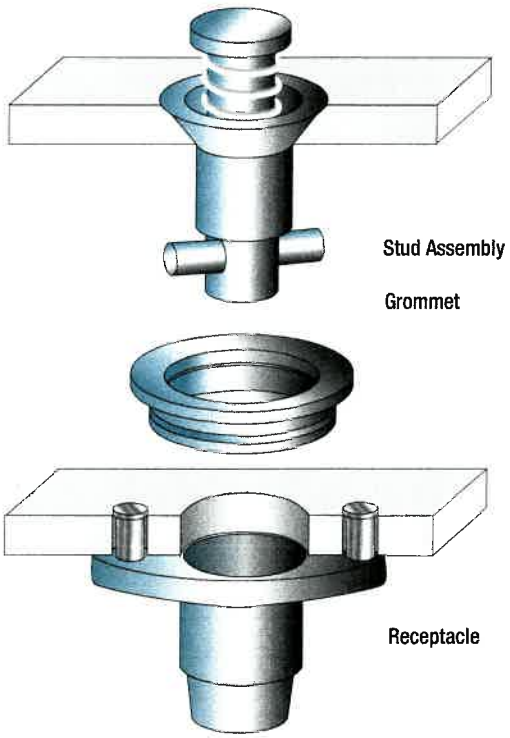


Figure 5-35. Camloc fastener.

Receptacles for Airloc fasteners are manufactured in two types: rigid and floating. Sizes are classified by number—No. 2, No. 5, and No. 7. They are also classified by the center-to-center distance between the rivet holes of the receptacle: No. 2 is ¾ inch; No. 5 is 1 inch; and No. 7 is 1⅜ inch. Receptacles are fabricated from high-carbon,

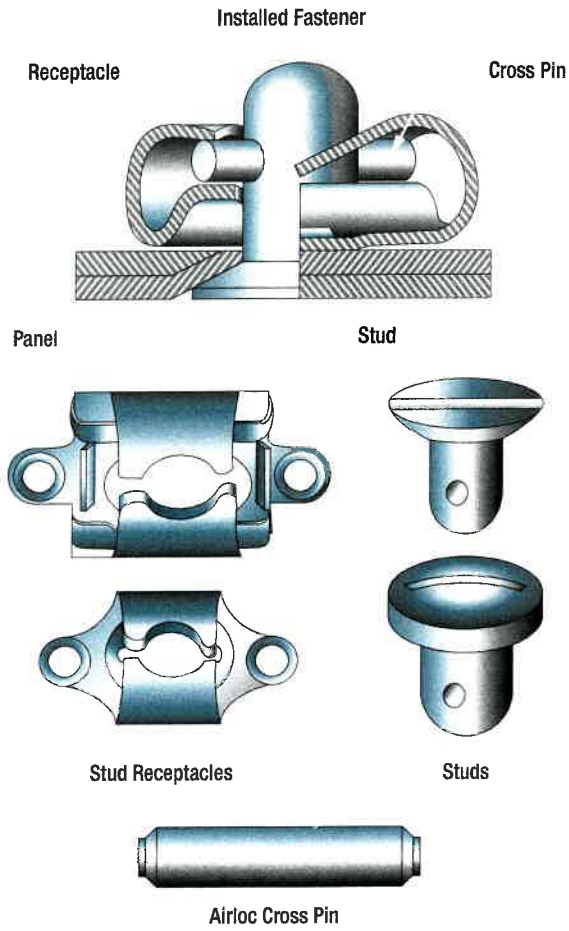


Figure 5-36. Airloc fastener.

heat treated steel. An upper wing assures ejection of the stud when unlocked and enables the cross pin to be held in a locked position between the upper wing, cam, stop, and wing detent, regardless of the tension to which the receptacle is subjected.

KEYS

A keyway is a slot cut in a part of a machine or an electrical connector to ensure correct orientation with another part. The keyway is fitted with a key that not only aligns the parts but also prevents rotation or movement between the parts. Keys are frequently used to mate a part onto a rotating shaft such as on a tapered aircraft propeller shaft. The shaft has machined into it a slot, the keyway, with a rectangular cross section as does the propeller hub. When the propeller is fitted to the shaft, a key inserted into the keyways aligns the assembly and prevents the propeller from rotation on the shaft. Torque is also transmitted from the propeller to the shaft through the key. Gears are often fitted to their drive shafts in the same manner. (***Figure 5-37***)

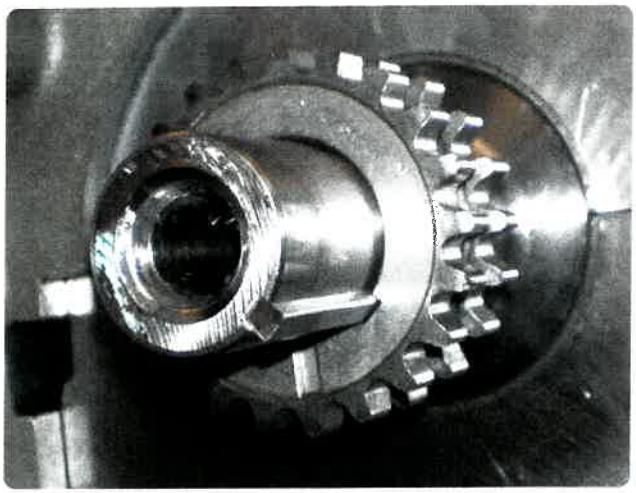


Figure 5-37. A key and keyway in a gear installation.

SNAP RINGS (CIRCLIPS)

A snap ring is a ring of metal, either round or flat in cross section, which is tempered to have springlike action. This springlike action will hold the snap ring firmly seated in a groove. The external types are designed to fit in a groove around the outside of a shaft or cylinder, and may be safety wired. Safety wiring of an external type snap ring is shown in ***Figure 5-38***.

The internal types fit in a groove inside a cylinder, and are never safetied. A special type of pliers is designed to install each type of snap ring. Snap rings can be reused as long as they retain their shape and springlike action.

SAFETY WIRE LOCKING

Safety wiring is the most satisfactory method of safetying capscrews, studs, nuts, bolt heads, and turnbuckle barrels which cannot be safetied by any other practical means. It is a method of wiring together two or more units in such a manner that any tendency of one to loosen is counteracted by the tightening of the wire.

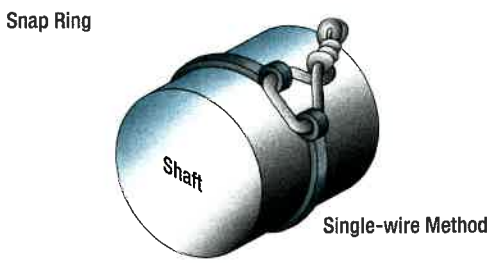


Figure 5-38. External type snapring with safety wire installation.

GENERAL SAFETY WIRING RULES

When using the safety wire method of safetying, the following general rules should be followed:

1. A pigtail of ¼ to ½ inch (three to six twists) should be made at the end of the wiring. This pigtail must be bent back or under to prevent it from becoming a snag.
2. The safety wire must be new upon each application.
3. When castellated nuts are to be secured with safety wire, tighten the nut to the low side of the selected torque range, unless otherwise specified, and if necessary, continue tightening until a slot aligns with the hole.
4. All safety wires must be tight after installation, but not under such tension that normal handling or vibration will break the wire.
5. The wire must be applied so that all pull exerted by the wire tends to tighten the nut.
6. Twists should be tight and even, and the wire between the nuts as taut as possible without over twisting.
7. The safety wire should always be installed and twisted so that the loop around the head stays down and does not tend to come up over the bolt head, causing a slack loop.

NUTS, BOLTS, AND SCREWS

Nuts, bolts, and screws are safety wired by the single wire or double twist method. The double twist method is the most common method of safety wiring. The single wire method may be used on small screws in a closely spaced closed geometrical pattern, on parts in electrical systems, and in places that are extremely difficult to reach. Safety wiring should always be per conventional methods or as required by the manufacturer, especially for Light Sport Aircraft (LSA).

Figure 5-39 is an illustration of various methods which are commonly used in safety wiring nuts, bolts, and screws. Careful study of ***Figure 5-39*** shows that:

- Examples 1, 2, and 5 illustrate the proper method of safety wiring bolts, screws, squarehead plugs, and similar parts when wired in pairs.
- Example 3 illustrates several components wired in series.
- Example 4 illustrates the proper method of wiring castellated nuts and studs. (Note that there is no loop around the nut.)
- Examples 6 and 7 illustrate a single threaded component wired to a housing or lug.
- Example 8 illustrates several components in a closely spaced closed geometrical pattern, using a single wire method.

When drilled head bolts, screws, or other parts are grouped together, they are more conveniently safety wired to each other in a series rather than individually. The number of nuts, bolts, or screws that may be safety wired together is dependent on the application. For instance, when safety wiring widely spaced bolts by the double twist method, a group of three should be the maximum number in a series.

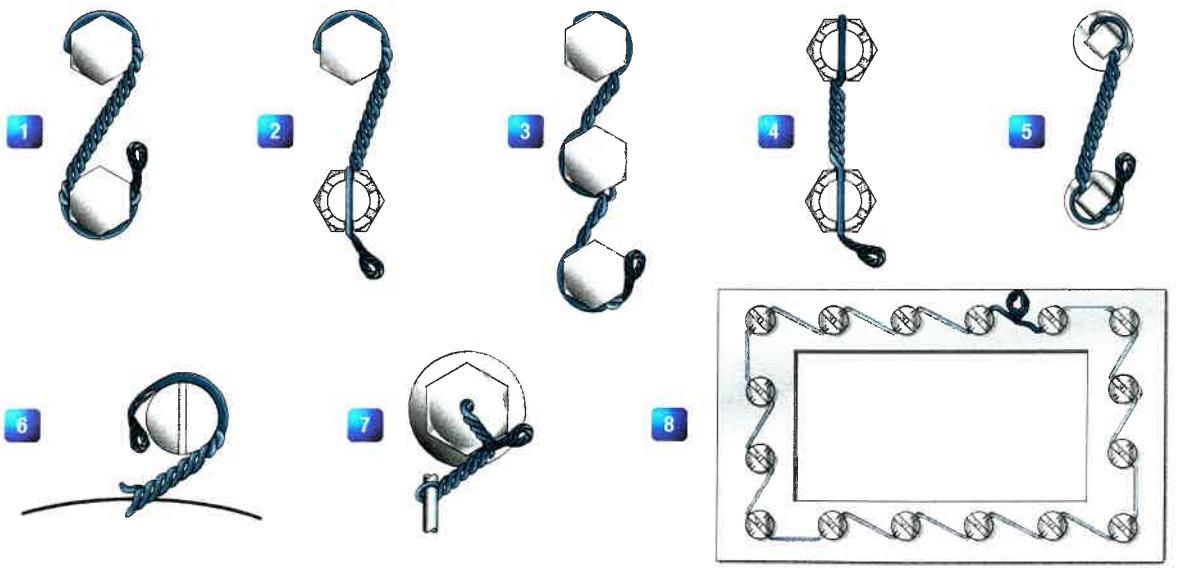


Figure 5-39. Safety wiring methods.

When safety wiring closely spaced bolts, the number that can be safety wired by a 24 inch length of wire is the maximum in a series. The wire is arranged so that if the bolt or screw begins to loosen, the force applied to the wire is in the tightening direction. Parts being safety wired should be torqued to recommend values and the holes aligned before attempting the safetying operation. Never over torque or loosen a torqued nut to align safety wire holes.

OIL CAPS, DRAIN COCKS, AND VALVES

These units are safety wired as shown in ***Figure 5-40***. In the case of the oil cap, the wire is anchored to an adjacent fillister head screw. This system applies to any other unit which must be safety wired individually. Ordinarily, anchorage lips are conveniently located near these individual parts. When such provision is not made, the safety wire is fastened to some adjacent part of the assembly.

ELECTRICAL CONNECTORS

Under conditions of severe vibration, the coupling nut of a connector may vibrate loose, and with sufficient vibration the connector may come apart. When this occurs, the circuit carried by the cable opens. The proper protective measure to prevent this occurrence is by safety wiring as shown in ***Figure 5-41***. The safety wire should be as short as practicable and must be installed in such a manner that the pull on the wire is in the direction which tightens the nut on the plug.

TURNBUCKLES

After a turnbuckle has been properly adjusted, it must be safetied. There are several methods of safetying turnbuckles; however, only two methods will be discussed in this section. These methods are illustrated in ***Figure 5-42***. The clip locking method is used only on the most modern aircraft. The older type aircraft still use the type turnbuckles that require the wire wrapping method.

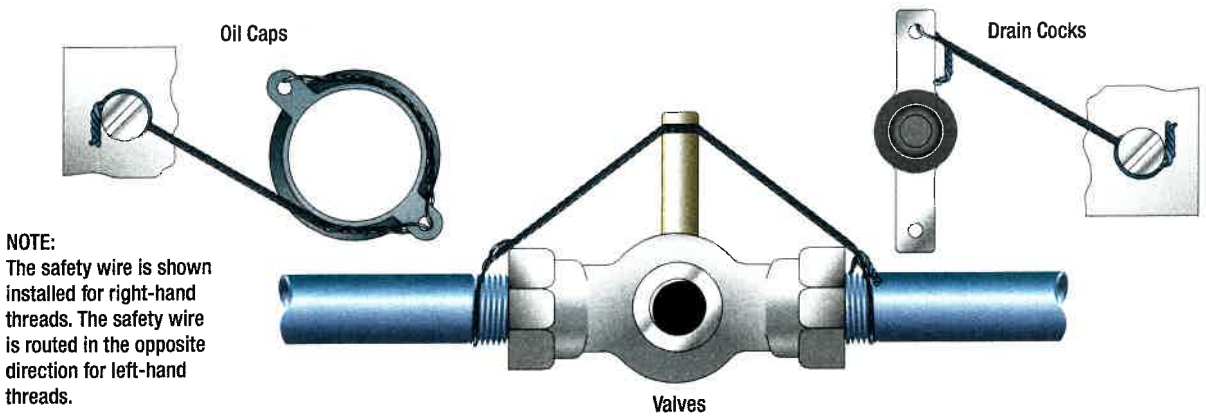


Figure 5-40. Safety wiring attachment for plug connectors.

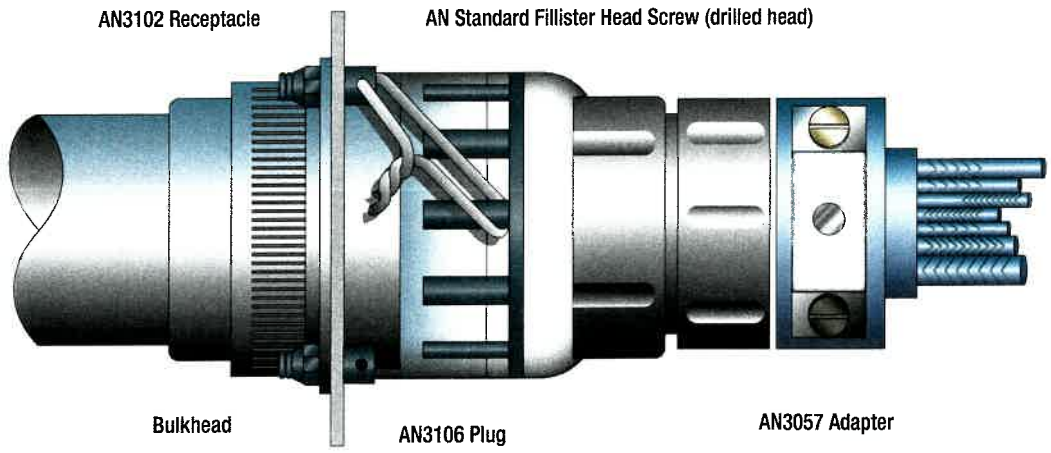
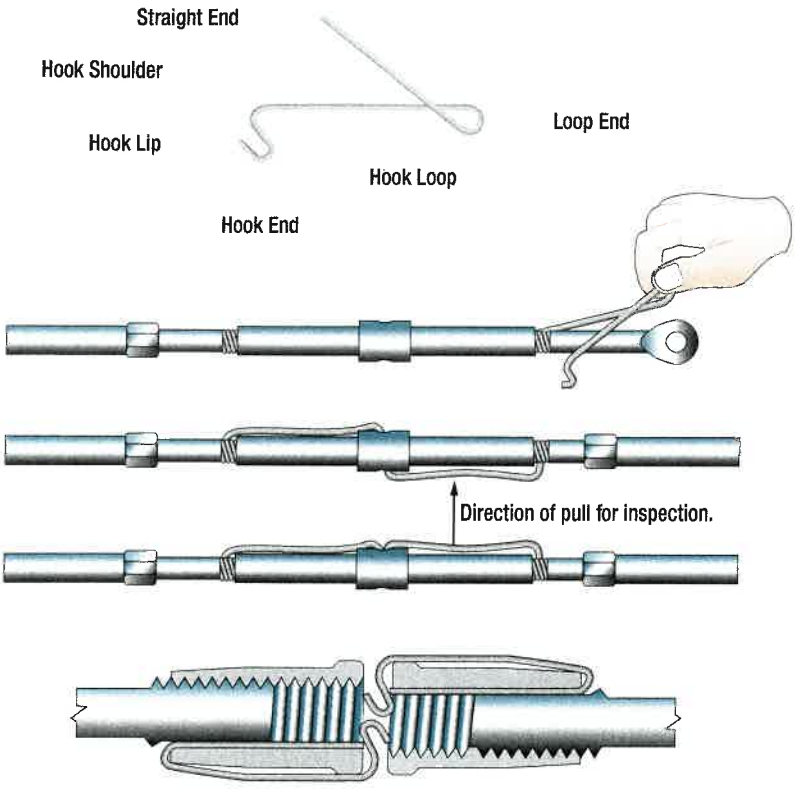
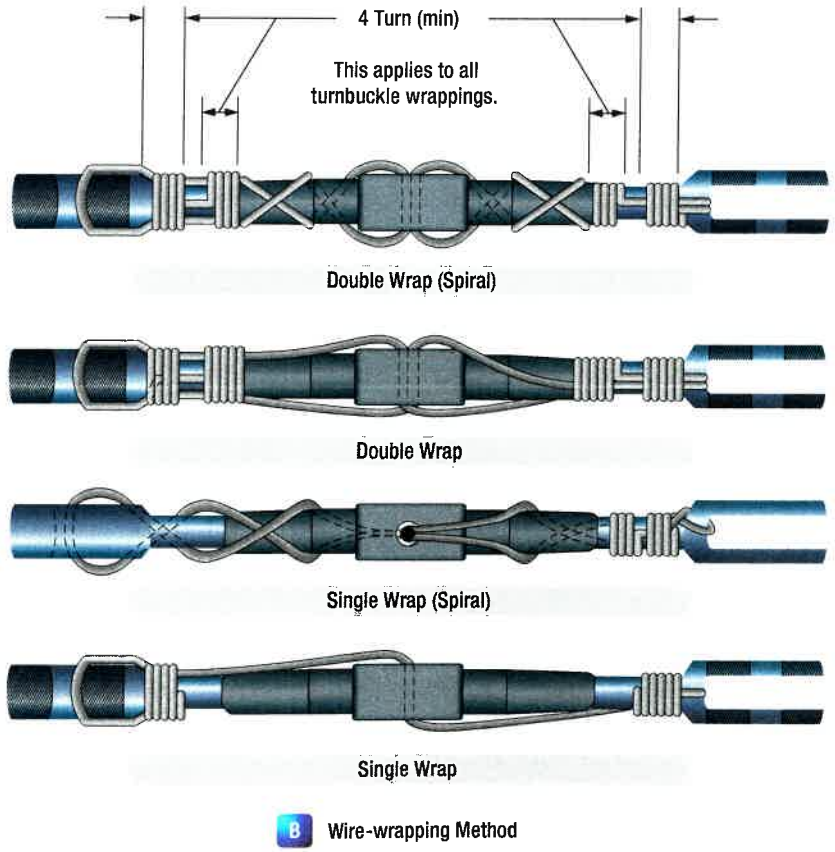


Figure 5-41. Safety wiring attachment for plug connectors.



A Clip-locking Method



B Wire-wrapping Method

Figure 5-42. Safetying turnbuckles: (A) clip-locking method and (B) wire-wrapping method.

Of the methods using safety wire for safetying turnbuckles, the double wrap method is preferred, although the single wrap methods described are satisfactory. The method of double wrap safetying is shown in ***Figure 5-42***. Use two separate lengths of the proper wire as shown in ***Figure 5-43***.

Run one end of the wire through the hole in the barrel of the turnbuckle and bend the ends of the wire toward opposite ends of the turnbuckle. Then pass the second length of the wire into the hole in the barrel and bend the ends along the barrel on the side opposite the first. Then pass the wires at the end of the turnbuckle in opposite directions through the holes in the turnbuckle eyes or between the jaws of the turnbuckle fork, as applicable. Bend the laid wires in place before cutting off the wrapped wire. Wrap the remaining length of safety wire at least four turns around the shank and cut it off. Repeat the procedure at the opposite end of the turnbuckle.

When a swaged terminal is being safetied, pass the ends of both wires, if possible, through the hole provided in the terminal for this purpose and wrap both ends around the shank as described above. If the hole is not large enough to allow passage of both wires, pass the wire through the hole and loop it over the free end of the other wire, and then wrap both ends around the shank as described. The single wrap safetying methods described in the following paragraphs are acceptable but are not the equal of the double wrap methods.

Pass a single length of wire through the cable eye or fork, or through the hole in the swaged terminal at either end of the turnbuckle assembly. Spiral each of the wire ends in opposite directions around the first half of the turnbuckle barrel so that the wires cross each other twice. Thread both wire ends through the hole in the middle of the barrel so that the third crossing of the wire ends is in the hole. Again, spiral the two wire ends in opposite directions around the remaining half of the turnbuckle, crossing them twice. Then, pass one wire end through the cable eye or fork, or through the hole in the swaged terminal. In the manner described above, wrap both wire ends around the shank for at least four turns each, cutting off the excess wire.

An alternate to the above method is to pass one length of wire through the center hole of the turnbuckle and bend the wire ends toward opposite ends of the turnbuckle. Then pass each wire end through the cable eye or fork, or through the hole in the swaged terminal and wrap each wire end around the shank for at least four turns, cutting off the excess wire. After safetying, no more than three threads of the turnbuckle threaded terminal should be exposed.

Cable size (in)	Type of Wrap	Diameter of Safety Wire (in)	Material (Annealed Condition)
1/16	Single	.020	Stainless Steel
3/32	Single	.040	Copper, Brass[1]
1/8	Single	.040	Stainless Steel
1/8	Double	.040	Copper, Brass[1]
1/8	Single	.057 min	Copper, Brass[1]
5/32 and greater	Single	.057	Stainless Steel

[1]Galvanized or tinned steel, or soft iron wires are also acceptable.

Figure 5-43. Turnbuckle safetying guide.

AIRCRAFT RIVETS

Sheets of metal must be fastened together to form the aircraft structure, and this is usually done with solid aluminum alloy rivets. A rivet is a metal pin with a formed head on one end when the rivet is manufactured. The shank of the rivet is inserted into a drilled hole, and its shank is then upset (deformed) by a hand or pneumatic tool. The second head, formed either by hand or by pneumatic equipment, is called a "shop head." The shop head functions in the same manner as a nut on a bolt. In addition to their use for joining aircraft skin sections, rivets are also used for joining spar sections, for holding rib sections in place, for securing fittings to various parts of the aircraft, and for fastening innumerable bracing members and other parts together. The rivet creates a bond that is at least as strong as the material being joined.

Two of the major types of rivets used in aircraft are the common solid shank type, which must be driven using a bucking bar, and the special (blind) rivets, which may be installed where it is impossible to use a bucking bar.

Aircraft rivets are not hardware store rivets. Rivets purchased at a hardware store should never be used as a substitute for aircraft quality rivets. The rivets may be made from very different materials, the strength of the rivets differs greatly, and their shear strength qualities are very different. The countersunk heads on hardware store rivets are 78°, whereas countersunk aircraft rivets have 100° angle heads for more surface contact to hold it in place.

RIVET IDENTIFICATION

Markings on the heads of rivets are used to classify their characteristics. These markings may be either a raised teat, two raised teats, a dimple, a pair of raised dashes, a raised cross, a single triangle, or a raised dash; some other heads have no markings.

The different markings indicate the composition of the rivet stock. As explained previously, the rivets have different colors to identify the manufacturer's' protective surface coating.

Roundhead rivets are used in the interior of the aircraft, except where clearance is required for adjacent members. The roundhead rivet has a deep, rounded top surface. The head is large enough to strengthen the sheet around the hole and, at the same time, offer resistance to tension.

The flathead rivet, like the roundhead rivet, is used on interior structures. It is used where maximum strength is needed and where there isn't sufficient clearance to use a roundhead rivet. It is seldom, if ever, used on external surfaces. The brazier head rivet has a head of large diameter, which makes it particularly adaptable for riveting thin sheet stock (skin). The brazier head rivet offers only slight resistance to the airflow, and because of this factor, it is frequently used for riveting skin on exterior surfaces, especially on aft sections of the fuselage and empennage. It is used for riveting thin sheets exposed to the slipstream. A modified brazier head rivet is also manufactured; it is simply a brazier head of reduced diameter.

The universal head rivet is a combination of the roundhead, flathead, and brazier head. It is used in aircraft construction and repair in both interior and exterior locations. When replacement is necessary for protruding head rivets—roundhead, flathead, or brazier head—they can be replaced by universal head rivets.

The countersunk head rivet is flat topped and beveled toward the shank so that it fits into a countersunk or dimpled hole and is flush with the material's surface. The angle at which the head slopes may vary from 78° to 120°. The 100° rivet is the most commonly used type. These rivets are used to fasten sheets over which other sheets must fit. They are also used on exterior surfaces of the aircraft because they offer only slight resistance to the slipstream and help to minimize turbulent airflow.

The markings on the heads of rivets indicate the material of which they are made and, therefore, their strength. ***Figure 5-44*** identifies the rivet head markings and the materials indicated by them. Although there are three materials indicated by a plain head, it is possible to distinguish their difference by color. The 1100 is aluminum color; the mild steel is a typical steel color; and the copper rivet is a copper color. Any head marking can appear on any head style of the same material.

Material	Head Marking	AN Material Code	AN425 78° Counter-sunk Head	AN426 100° Counter-sunk Head MS20426*	AN427 100° Counter-sunk Head MS20427*	AN430 Round Head MS20470*	AN435 Round Head MS20613* MS20615*	AN441 Flat Head	AN442 Flat Head MS20470*	AN455 Brazier Head MS20470*	AN456 Brazier Head MS20470*	AN470 Universal Head MS20470*	Heat Treat Before Use	Shear Strength psi	Bearing Strength psi
1100	Plain	A	X	X		X			X	X	X	X	No	10 000	25 000
2117T	Recessed Dot	AD	X	X		X			X	X	X	X	No	30 000	100 000
2017T	Raised Dot	D	X	X		X			X	X	X	X	Yes	34 000	113 000
2017T-HD	Raised Dot	D	X	X		X			X	X	X	X	No	38 000	126 000
2024T	Raised Double Dash	DD	X	X		X			X	X	X	X	Yes	41 000	136 000
5056T	Raised Cross	B		X		X			X	X	X	X	No	27 000	90 000
7075-T73	Three Raised Dashes		X	X		X			X	X	X	X	No		
Carbon Steel	Recessed Triangle				X		X MS20613*	X					No	35 000	90 000
Corrosion Resistant Steel	Recessed Dash	F			X		X MS20613*						No	65 000	90 000
Copper	Plain	C			X		X	X					No	23 000	
Monel	Plain	M			X			X					No	49 000	
Monel (Nickel-Copper Alloy)	Recessed Double Dots	C					X MS20615*						No	49 000	
Brass	Plain						X MS20615*						No		
Titanium	Recessed Large and Small Dot			MS20426									No	95 000	

*New specifications are for design purposes.

Figure 5-44. Rivet identification chart.

Each type of rivet is identified by a part number so that the user can select the correct rivet for the job. The type of rivet head is identified by AN or MS standard numbers. The numbers selected are in series and each series represents a particular type of head. (***Figure 5-44***)

The most common numbers and the types of heads they represent are:

AN426 or MS20426—Countersunk head rivets-100°
AN430 or MS20430—Roundhead rivets
AN441—Flathead rivets
AN456—Brazier head rivets
AN470 or MS20470—Universal head rivets

There are also letters and numbers added to a part number. The letters designate alloy content; the numbers designate rivet diameter and length.

The letters in common use for alloy designation are:

A—Aluminum alloy, 1100 or 3003 composition
AD—Aluminum alloy, 2117-T composition
D—Aluminum alloy, 2017-T composition
DD—Aluminum alloy, 2024-T composition
B—Aluminum alloy, 5056 composition
C—Copper
M—Monel

The absence of a letter following the AN standard number indicates a rivet made from mild steel. The first number following the material composition letters expresses the diameter of the rivet shank in 32nds of an inch (Examples: 3 indicates 3/32, 5 indicates 5/32, and so forth). (***Figure 5-45***)

The last number(s), separated by a dash from the preceding number, expresses the length of the rivet shank in 16ths of an inch (Examples: 3 indicates 3/16, 7 indicates 7/16, 11 indicates 11/16, and so forth).

An example of identification marking of a rivet is:

AN470AD3-5—Complete part number
AN—Air Force-Navy standard number
470—Universal head rivet
AD—2117-T Aluminum alloy
3—3/32 in diameter
5—5/16 in length

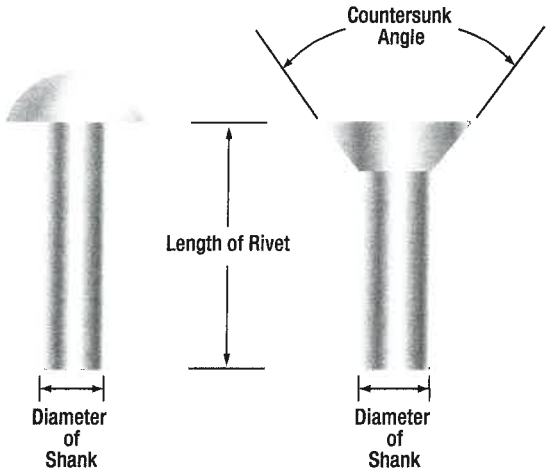


Figure 5-45. Methods of measuring rivets.

STANDARDS AND SPECIFICATIONS

The FAA requires that the structural strength and integrity of type-certificated aircraft conform to all airworthiness requirements. These requirements apply to performance, structural strength, and integrity as well flight characteristics. To meet these requirements, each aircraft must meet the same standards. To accomplish standardization, all materials and hardware must be manufactured to a standard of quality. Specifications and standards for aircraft hardware are usually identified by the organization that originated them.

Some common standardizing organizations include:

AMS—Aeronautical Material Specifications
AN—Air Force-Navy
AND—Air Force-Navy Design
AS—Aeronautical Standard
ASA—American Standards Association
ASTM—American Society for Testing Materials
MS—Military Standard
NAF—Naval Aircraft Factory
NAS—National Aerospace Standard
SAE—Society of Automotive Engineers

When a MS20426-AD4-6 rivet is required, the specifications have already been written for it in the Military Standard (MS) specifications and is available to the aircraft manufacturer's and to the rivet manufacturer's as well as to the mechanic. The specifications designate the material to be used as well as the head type, diameter, and length of the rivet. The use of standardized materials in the production of aircraft makes each aircraft exactly the same as the previous one and makes them less expensive to build.

Aircraft rivets are manufactured to much higher standards and specifications than for general use. When aircraft manufacturer's started building all-metal aircraft in the 1930s, different manufacturer's had different rivet head designs. Brazier heads, modified brazier heads, button heads, mushroom heads, flatheads, and 78° countersunk heads were used. As aircraft standardized, four rivet head designs almost completely replaced all of the others. Rivets exposed to the airflow over the top of the structure are usually either universal head MS20470 or 100° countersunk head MS20426 rivets. For rivets used in internal structures, the roundhead MS20430 and flathead MS20442 are generally used.

SOLID SHANK RIVETS

Solid shank rivets are generally used in repair work. They are identified by the kind of material of which they are made, their head type, size of shank, and their temper condition. The designation of the solid shank rivet head type, such as universal head, roundhead, flathead, countersunk head, and brazier head, depends on the cross-sectional shape of the head. (***Figure 5-44***) The temper designation and strength are indicated by special markings on the head of the rivet.

The material used for the majority of aircraft solid shank rivets is aluminum alloy. The strength and temper conditions of aluminum alloy rivets are identified by digits and letters similar to those adopted for the identification of strength and temper conditions of aluminum and aluminum alloy stock. The 1100, 2017-T, 2024-T, 2117-T, and 5056 rivets are the five grades usually available.

The 1100 rivet, which is composed of 99.45 percent pure aluminum, is very soft. It is for riveting the softer aluminum alloys, such as 1100, 3003, and 5052, which are used for nonstructural parts (all parts where strength is not a factor). The riveting of map cases is a good example of where a rivet of 1100 aluminum alloy may be used.

The 2117-T rivet, known as the field rivet, is used more than any other for riveting aluminum alloy structures. The field rivet is in wide demand because it is ready for use as received and needs no further heat treating or annealing. It also has a high resistance to corrosion.

The 2017-T and 2024-T rivets are used in aluminum alloy structures where more strength is needed than is obtainable with the same size 2117-T rivet. These rivets are known as "ice box rivets," are annealed, and must be kept refrigerated until they are to be driven. The 2017-T rivet should be driven within approximately 1 hour and the 2024-T rivet within 10 to 20 minutes after removal from refrigeration.

The 5056 rivet is used for riveting magnesium alloy structures because of its corrosion-resistant qualities in combination with magnesium. Mild steel rivets are used for riveting steel parts. The corrosion resistant steel rivets are for riveting corrosion- resistant steels in firewalls, exhaust stack brackets, and similar structures.

Monel rivets are used for riveting nickel-steel alloys. They can be substituted for those made of corrosion resistant steel in some cases.

The use of copper rivets in aircraft repair is limited. Copper rivets can be used only on copper alloys or nonmetallic materials such as leather.

HEAT TREATING RIVETS

Metal temper is an important factor in the riveting process, especially with aluminum alloy rivets. Aluminum alloy rivets have the same heat treating characteristics as aluminum alloy stock. They can be hardened and annealed in the same manner as aluminum. The rivet must be soft, or comparatively soft, before a good head can be formed. The 2017-T and 2024-T rivets are annealed before being driven. They harden with age.

The process of heat treating (annealing) rivets is much the same as that for stock. Either an electric air furnace, a salt bath, or a hot oil bath is needed. The heat treating range, depending on the alloy, is 330°C to 510°C. For convenient handling, rivets are heated in a tray or a wire basket. They are quenched in cold water (21°C) immediately after heat treating. The 2017-T and 2024-T rivets, which are heat-treatable rivets, begin to age harden within a few minutes after being exposed to room temperature. Therefore, they must be used immediately after quenching or else be placed in cold storage.

The most commonly used means for holding heat treatable rivets at low temperature (below 0°C) is to keep them in a refrigerator. They are referred to as "icebox" rivets. Under this storage condition, they will remain soft enough for driving for periods up to 2 weeks. Any rivets not used within that time should be removed for reheat treating.

Icebox rivets attain about one-half their maximum strength in approximately 1 hour after driving and full strength in about 4 days. When 2017-T rivets are exposed to room temperature for 1 hour or longer, they must be subject to reheat treatment. This also applies to 2024-T rivets exposed to room temperature for a period exceeding 10 minutes.

Once an icebox rivet has been taken from the refrigerator, it should not be mixed with the rivets still in cold storage. If more rivets are removed from the refrigerator than can be used in 15 minutes, they should be placed in a separate container and stored for reheat treatment. Heat treatment of rivets may be repeated a number of times if done properly. Proper heating times and temperatures are shown in ***Figure 5-46***.

RIVET CORROSION

Most metals, and therefore aircraft rivet stock, are subject to corrosion. Corrosion may be the result of local climatic conditions or the fabrication process used. It is reduced to a minimum by using metals which are highly resistant to corrosion and possess the correct strength to weight ratio.

Heating Time — Air Furnace

Rivet Alloy	Time at Temperature	Heat Treating Temperature
2024	1 hour	488°C–500°C
2017	1 hour	496°C–510°C

Heating Time — Salt Bath

Rivet Alloy	Time at Temperature	Heat Treating Temperature
2024	30 minutes	488°C–500°C
2017	30 minutes	496°C–510°C

Figure 5-46. Rivet heating times and temperatures.

Ferrous metals placed in contact with moist salt air will rust if not properly protected. Nonferrous metals, those without an iron base, do not rust, but a similar process known as corrosion takes place. The salt in moist air (found in the coastal areas) attacks the aluminum alloys. It is a common experience to inspect the rivets of an aircraft which has been operated near salt water and find them badly corroded.

If a copper rivet is inserted into an aluminum alloy structure, two dissimilar metals are brought in contact with each other. Remember, all metals possess a small electrical potential. Dissimilar metals in contact with each other in the presence of moisture cause an electrical current to flow between them and chemical byproducts to be formed. Principally, this results in the deterioration of one of the metals. Certain aluminum alloys react to each other and, therefore, must be thought of as dissimilar metals. The commonly used aluminum alloys may be divided into the two groups shown in ***Figure 5-47***.

Members within either group A or group B can be considered as similar to each other and will not react to others within the same group. A corroding action will take place, however, if any metal of group A comes in contact with a metal in group B in the presence of moisture.

Avoid the use of dissimilar metals whenever possible. Their incompatibility is a factor which was considered when the AN Standards were adopted. To comply with AN Standards, the manufacturer's must put a protective surface coating on the rivets. This may be zinc chromate, metal spray, or an anodized finish.

The protective coating on a rivet is identified by its color. A rivet coated with zinc chromate is yellow, an anodized surface is pearl gray, and the metal sprayed rivet is

Group A	Group B
1100	2117
3003	2017
5052	2124
6053	7075

Figure 5-47. Aluminum groupings.

identified by a silvery gray color. If a situation arises in which a protective coating must be applied on the job, paint the rivet with zinc chromate before it is used and again after it is driven.

BLIND RIVETS

There are many places on an aircraft where access to both sides of a riveted structure or structural part is impossible, or where limited space will not permit the use of a bucking bar. Also, in the attachment of many nonstructural parts, such as aircraft interior furnishings, flooring, deicing boots, and the like, the full strength of solid shank rivets is not necessary.

For use in such places, special rivets have been designed which can be bucked from the front. They are sometimes lighter than solid shank rivets, yet amply strong for their intended use. These rivets are produced by several manufacturer's and have unique characteristics that require special installation tools, special installation procedures and special removal procedures. That is why they are called special rivets. Because these rivets are often inserted in locations where one head (usually the shop head) cannot be seen, they are also called blind rivets.

MECHANICALLY EXPANDED RIVETS

Two classes of mechanically expanded rivets are:

1. Nonstructural.
   (A) Self-plugging (friction lock) rivets.
   (B) Pull thru rivets.
2. Mechanical lock, flush fracturing, and self-plugging rivets.

SELF-PLUGGING RIVETS (FRICTION LOCK)

The self-plugging (friction lock) blind rivets are manufactured by several companies; the same general basic information about their fabrication, composition, uses, selection, installation, inspection, and removal procedures apply to all of them.

Self-plugging (friction lock) rivets are fabricated in two parts: a rivet head with a hollow shank or sleeve, and a stem that extends through the hollow shank. ***Figure 5-48*** illustrates a protruding head and a countersunk head self-plugging rivet produced by one manufacturer.

Several events, in their proper sequence, occur when a pulling force is applied to the stem of the rivet: (1) the stem is pulled into the rivet shank; (2) the mandrel portion of the stem forces the rivet shank to expand; and (3) when friction (or pulling action pressure) becomes great enough, it will cause the stem to snap at a break off groove on the stem. The plug portion (bottom end of the stem) is retained in the shank of the rivet giving the rivet much greater shear strength than could be obtained from a hollow rivet.

Self-plugging (friction lock) rivets are fabricated in two common head styles: (1) a protruding head similar to the MS20470 or universal head, and (2) a 100° countersunk head. Other head styles are available from some manufacturer's. The stem of the self-plugging (friction lock) rivet may have a knot or knob on the upper portion, or it may have a serrated portion. (***Figure 5-48***)

Self-plugging (friction lock) rivets are fabricated from several materials. Rivets are available in the following material combinations: stem 2017 aluminum alloy and sleeve 2117 aluminum alloy; stem 2017 aluminum alloy and sleeve 5056 aluminum alloy; and stem steel and sleeve steel.

Self-plugging (friction lock) rivets are designed so that installation requires only one person; it is not necessary to have the work accessible from both sides. The pulling strength of the rivet stem is such that a uniform job can always be assured. Because it is not necessary to have access to the opposite side of the work, self plugging

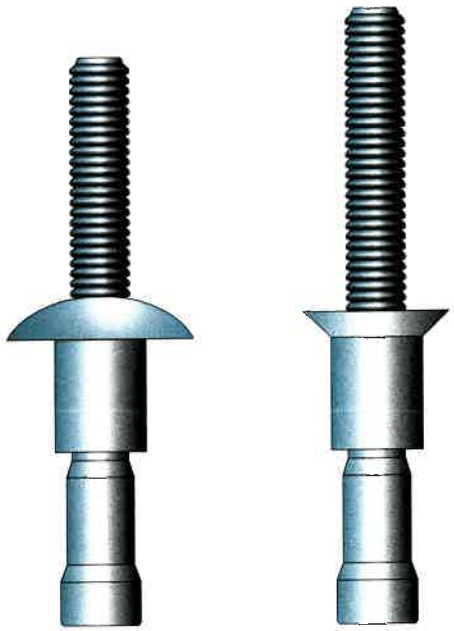


Figure 5-48. Self-plugging (friction lock) rivets.

(friction lock) rivets can be used to attach assemblies to hollow tubes, corrugated sheet, hollow boxes, and so forth. Because a hammering force is not necessary to install the rivet, it can be used to attach assemblies to plywood or plastics.

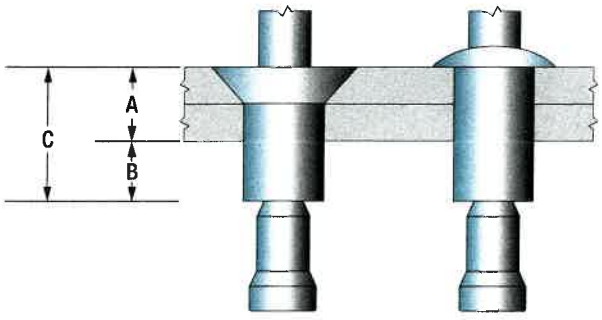
Factors to consider in the selection of the correct rivet for installation are: (1) installation location, (2) composition of the material being riveted, (3) thickness of the material being riveted, and (4) strength desired. If the rivet is to be installed on an aerodynamically smooth surface, or if clearance for an assembly is needed, countersunk head rivets should be selected. In other areas where clearance or smoothness is not a factor, the protruding head type rivet may be utilized.

Material composition of the rivet shank depends upon the type of material being riveted. Aluminum alloy 2117 shank rivets can be used on most aluminum alloys. Aluminum alloy 5056 shank rivets should be used when the material being riveted is magnesium. Steel rivets should always be selected for riveting assemblies fabricated from steel.

The thickness of the material being riveted determines the overall length of the shank of the rivet. As a general rule, the shank of the rivet should extend beyond the material thickness approximately 3/64 inch to 1/8 inch before the stem is pulled. (***Figure 5-49***)

PULL-THRU RIVETS

The pull-thru blind rivets are manufactured by several companies; the same general basic information about their fabrication, composition, uses, selection, installation, inspection, and removal procedures apply to all of them.



A = Thickness of Material (Grip Range)
B = 3/64 - 1/8"
C = Total Rivet Shank Length

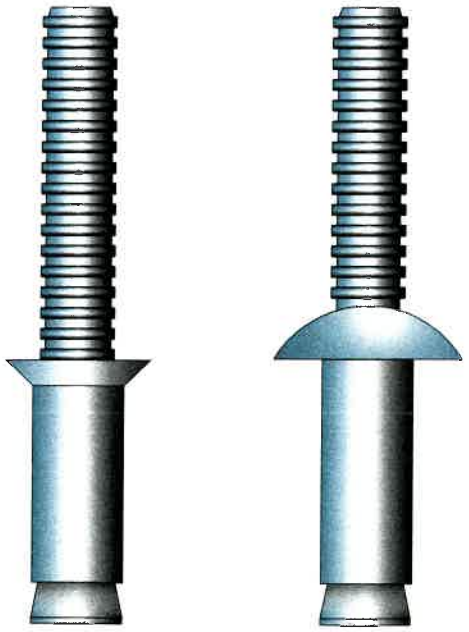
Figure 5-49. Determining length of friction lock rivets.

Pull-thru rivets are fabricated in two parts: a rivet head with a hollow shank or sleeve and a stem that extends through the hollow shank. ***Figure 5-50*** illustrates a protruding head and a countersunk head pull-thru rivet. Several events, in their proper sequence, occur when a pulling force is applied to the stem of the rivet: (1) The stem is pulled through the rivet shank; (2) the mandrel portion of the stem forces the shank to expand forming the blind head and filling the hole.

Pull-thru rivets are fabricated in two common head styles: (1) protruding head similar to the MS20470 or universal head, and (2) a 100° countersunk head. Other head styles are available from some manufacturer's.

Pull-thru rivets are fabricated from several materials. Following are the most commonly used: 2117-T4 aluminum alloy, 5056 aluminum alloy, Monel.

Pull-thru rivets are designed so that installation requires only one person; it is not necessary to have the work accessible from both sides. Factors to consider in the selection of the correct rivet for installation are: (1) installation location, (2) composition of the material being riveted, (3) thickness of the material being riveted, and (4) strength desired.



Countersunk Head Protruding Head

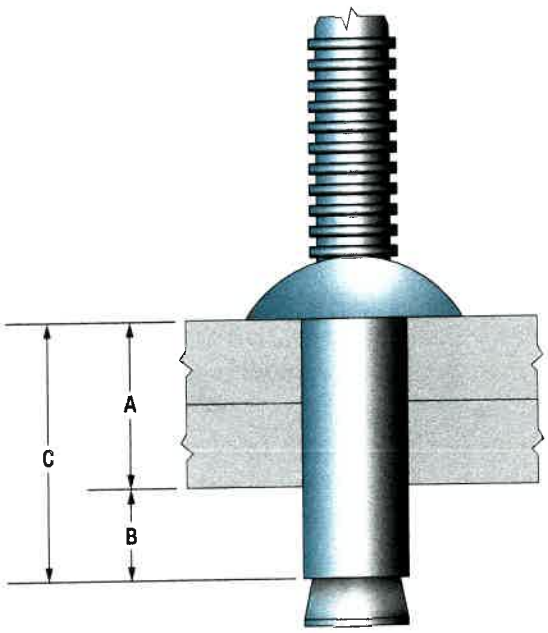
Figure 5-50. Pull-thru rivets.

The thickness of the material being riveted determines the overall length of the shank of the rivet. As a general rule, the shank of the rivet should extend beyond the material thickness approximately 3/64-inch to 1/8-inch before the stem is pulled. (***Figure 5-51***)

Each company that manufactures pull-thru rivets has a code number to help users obtain correct rivet for the grip range of a particular installation. In addition, MS numbers are used for identification purposes. Numbers are similar to those shown on the preceding pages.

SELF-PLUGGING RIVETS (MECHANICAL LOCK)

Self-plugging (mechanical lock) rivets are similar to self-plugging (friction lock) rivets, except for the manner in which the stem is retained in the rivet sleeve. This type of rivet has a positive mechanical locking collar to resist vibrations that cause the friction lock rivets to loosen and possibly fall out. (***Figure 5-52***) Also, the mechanical locking type rivet stem breaks off flush with the head and usually does not require further stem trimming when properly installed. Self-plugging (mechanical lock) rivets display all the strength characteristics of solid shank rivets and in most cases can be substituted rivet for rivet.

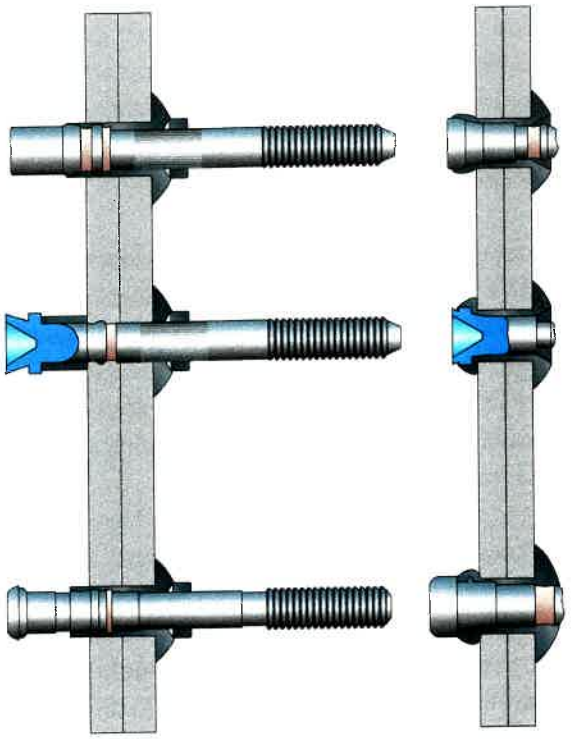


A = Thickness of Material (Grip Range)
B = 3/64 – 1/8"
C = Total Rivet Shank Length

Figure 5-51. Determining length of pull-thru rivets.

BULBED CHERRYLOCK® RIVETS

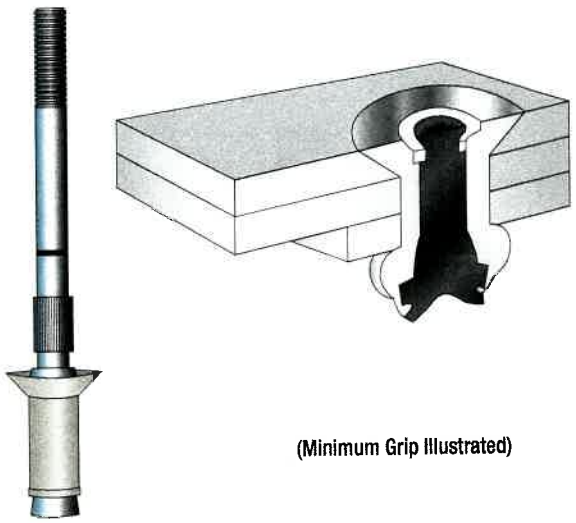
The large blind head of this fastener introduced the word "bulb" to blind rivet terminology. In conjunction with the unique residual preload developed by the high stem break load, its proven fatigue strength makes it the only blind rivet interchangeable structurally with solid rivets. (***Figure 5-53***)



Before Installation

After Installation

Figure 5-52. Self-plugging (mechanical lock) rivets.



(Minimum Grip Illustrated)

Figure 5-53. Bulbed cherrylock rivet.

WIREDRAW CHERRYLOCK® RIVETS

There is a wide range of sizes, materials, and strength levels from which to select. This fastener is especially suited for sealing applications and joints requiring an excessive amount of sheet takeup. (***Figure 5-54***)

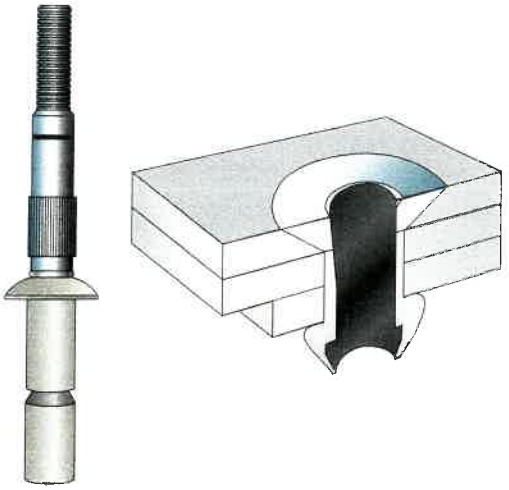


Figure 5-54. Wiredraw cherrylock rivet.

HUCK MECHANICAL LOCKED RIVETS

Self-plugging (mechanical lock) rivets are fabricated in two sections: a head and shank (including a conical recess and locking collar in the head), and a serrated stem that extends through the shank. Unlike the friction lock rivet, the mechanical lock rivet has a locking collar that forms a positive lock for retention of the stem in the shank of the rivet. This collar is seated in position during the installation of the rivet.

BLIND RIVET SPECIFICATIONS

RIVET MATERIAL

Self-plugging (mechanical lock) rivets are fabricated with sleeves (rivet shanks) of 2017 and 5056 aluminum alloys, Monel, or stainless steel. The mechanical lock type of self-plugging rivet can be used in the same applications as the friction lock type of rivet. In addition, because of its greater stem retention characteristic, installation in areas subject to considerable vibration is recommended.

The same general requirements must be met in the selection of the mechanical lock rivet as for the friction lock rivet. Composition of the material being joined together determines the composition of the rivet sleeve; for example, 2017 aluminum alloy rivets for most aluminum alloys and 5056 aluminum rivets for magnesium.

Figure 5-55 depicts the sequences of a typical mechanically locked blind rivet. The form and function may vary slightly between blind rivet styles and specifics should be obtained from manufacturer's.

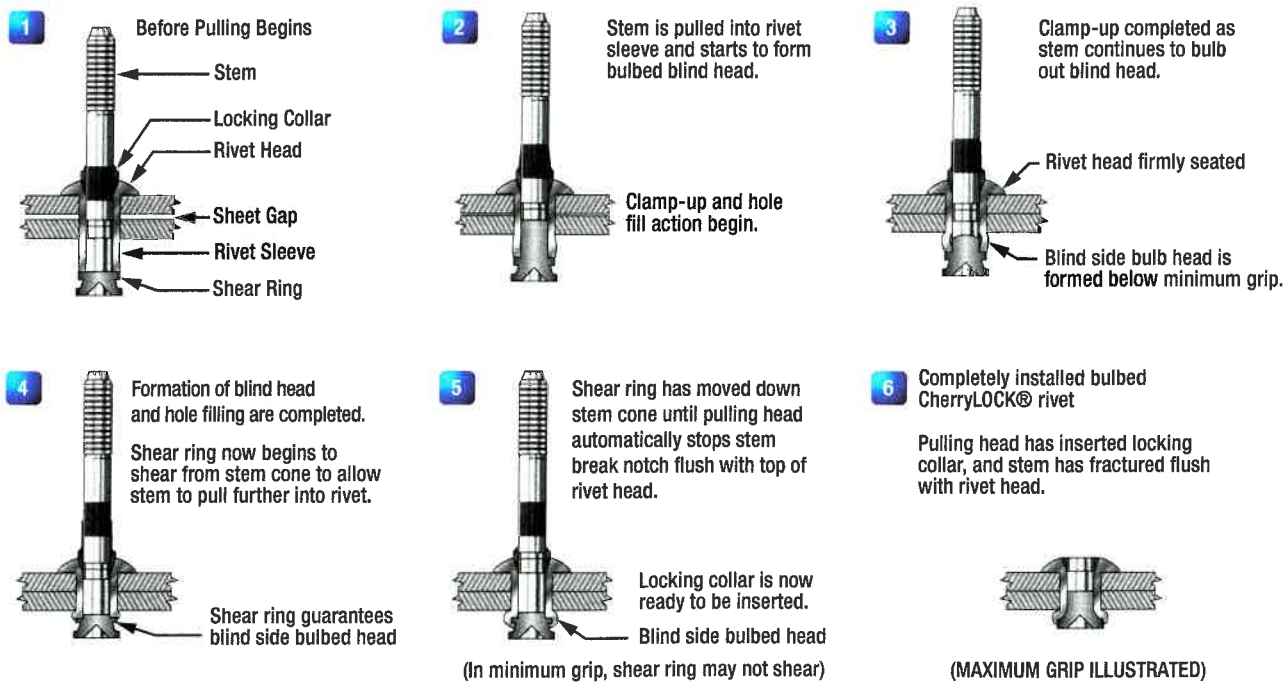


Figure 5-55. Cherrylock® rivet installation.

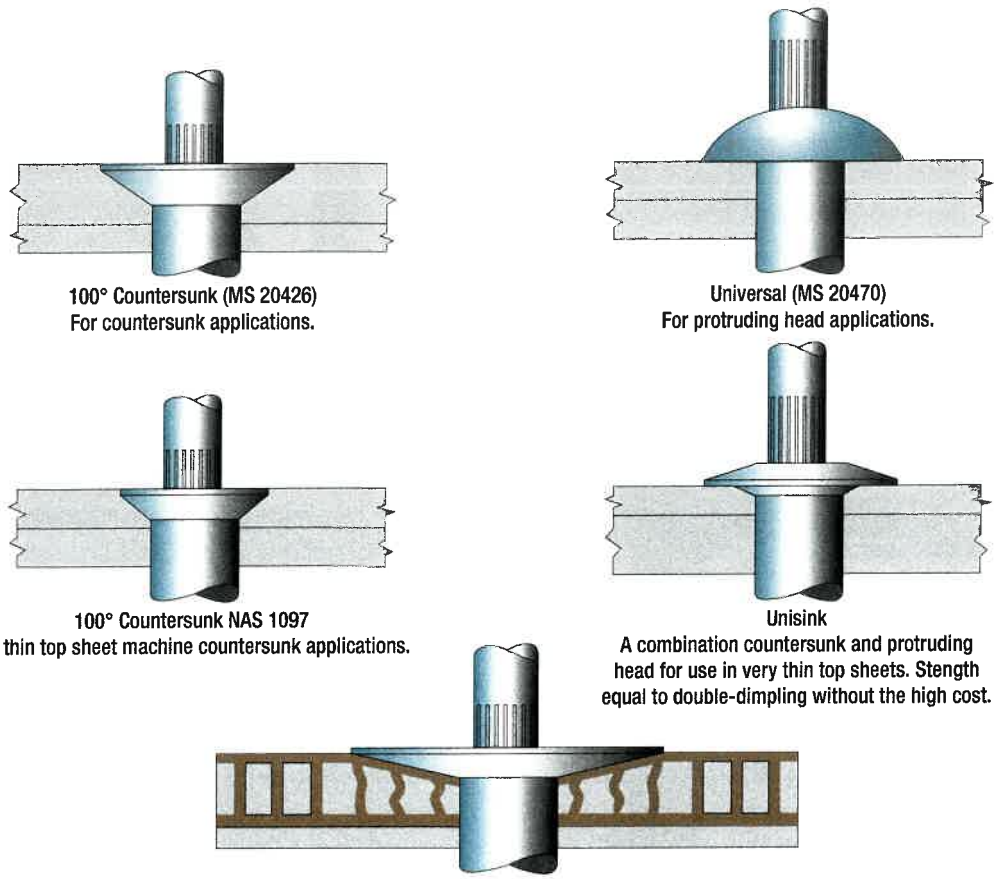


Figure 5-56. Cherrylock® rivet heads.

HEAD STYLES

Self-plugging mechanical locked blind rivets are available in several head styles depending on the installation requirements. (***Figure 5-56***)

DIAMETERS

Shank diameters are measured in 1/32-inch increments and are generally identified by the first dash number: -3 indicates 3/32 inch diameter, -4 indicates 1/8 inch diameter, and so forth. Both nominal and 1/64 inch oversize diameters are available.

GRIP LENGTH

Grip length refers to the maximum total sheet thickness to be riveted and is measured in 1/16 of an inch. This is generally identified by the second dash number. Unless otherwise noted, most blind rivets have their grip lengths (maximum grip) marked on the rivet head and have a total grip range of 1/16 inch. (***Figure 5-57***)

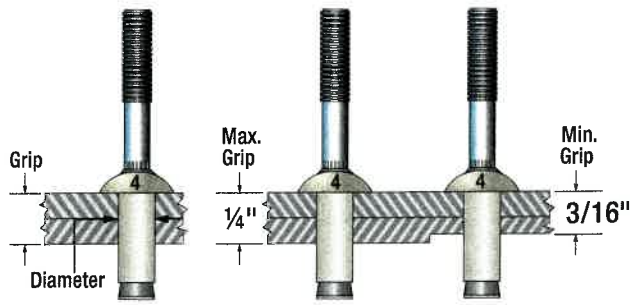


Figure 5-57. Typical grip length.

To determine the proper grip rivet to use, measure the material thickness with a grip selection gauge (available from blind rivet manufacturer's). The proper use of a grip selector gauge is shown in ***Figure 5-58***.

The thickness of the material being riveted determines the overall length of the shank of the rivet. As a general rule, the shank of the rivet should extend beyond the material thickness approximately 3/64 inch to 1/8 inch before the stem is pulled. (***Figure 5-59***)

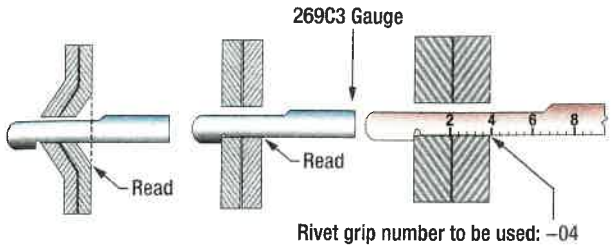


Figure 5-58. Grip gauge use.

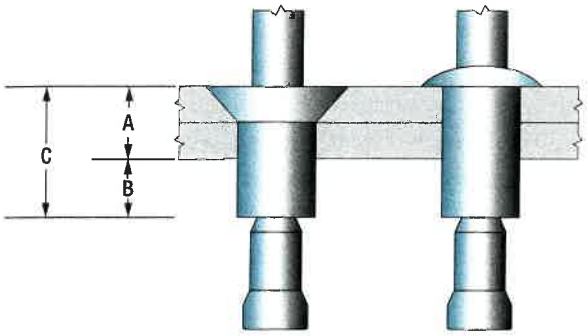


Figure 5-59. Determining rivet length.

BLIND RIVET IDENTIFICATION

Each company that manufactures self-plugging (friction lock) rivets has a code number to help users obtain the correct rivet for the grip range or material thickness of a particular installation. In addition, MS numbers are used for identification purposes. ***Figures 5-60 through 5-63*** contain examples of part numbers for self-plugging (friction lock) rivets that are representative of each.

SPECIAL RIVET TYPES

Many special fasteners produce high strength with light weight and can be used in place of conventional AN bolts and nuts. When AN bolts are tightened with the nut, the bolt stretches, narrowing the diameter and then the bolt is no longer tight in the hole. Special fasteners eliminate this loose fit because they are held in place by a collar that is squeezed into position. These fasteners are not under the same tensile loads as a bolt during installation. Special fasteners are also used extensively for Light Sport Aircraft (LSA). Always follow the aircraft manufacturer's recommendations.

PIN RIVETS

Pin (Hi-Shear) rivets are classified as special rivets but are not of the blind type. Access to both sides of the material is required to install this type of rivet.

Huck Manufacturing Company

9SP-B A 6 3

9SP-B	**Head Style** 9SP-B = brazier or universal head 9SP-100 = 100° countersunk head
A	**Material composition of shank** A = 2017 aluminum alloy B = 5056 aluminum alloy R = mild steel
6	**Shank diameter in 32nds of an inch:** 4 = 1/8" 6 = 3/16" 5 = 5/32" 8 = 1/4"
3	Grip range (material thickness) in 16ths of an inch

Figure 5-60. Huck Manufacturing Company codes.

Olympic Screw and Rivet Corporation

RV 2 0 0 4 2

RV	**Manufacturer** Olympic Screw and Rivet Corporation
2	**Rivet type** 2 = self plugging (friction lock) 5 = hollow pull through
0	**Material composition of shank** 0 = 2017 aluminum alloy 5 = 5056 aluminum alloy 7 = mild steel
0	**Head style** 0 = universal head 1 = 100° countersunk
4	**Shank diameter in 32nds of an inch:** 4 = 1/8" 6 = 3/16" 5 = 5/32" 8 = 1/4"
2	Grip range in 16ths of an inch

Figure 5-61. Olympic Screw and Rivet Corporation codes.

Townsend Company, Cherry Rivet Division

CR 163 6 6

CR	**Cherry rivet**
163	**Series number** Designates rivet material, type of rivet, and head style (163 = 2117 aluminum alloy, self-plugging (friction lock) rivet, protruding head)
6	**Shank diameter in 32nds of an inch:** 4 = 1/8" 6 = 3/16" 5 = 5/32" 8 = 1/4"
6	**Grip range (material thickness):** Knob stem in 32nds of an inch; serrated stem in 16ths of an inch

Figure 5-62. Townsend Company, Cherry Rivet Division codes.

Pin rivets have the same shear strength as bolts of equal diameters, are about 40 percent of the weight of a bolt, and require only about one-fifth as much time

Military Standard Number

MS 20600 B 4 K 2

- **MS** | **Military Standard**
- **20600** | **Type of rivet and head style:**
  20600 = self-plugging (friction lock) protruding head
  20600 = self-plugging (friction lock) 100° countersunk head
- **B** | **Material composition of sleeve:**
  AD = 2117 aluminum alloy
  B = 5056 aluminum alloy
- **4** | **Shank diameter in 32nds of an inch:**
  4 = 1/8" 6 = 3/16"
  5 = 5/32" 8 = 1/4"
- **K** | **Type of stem:**
  K = knot head stem
  W = serrated stem
- **2** | Grip range (material thickness) in 16ths of an inch

Figure 5-63. Military Standard Numbers.

for installation as a bolt, nut, and washer combination. They are approximately three times as strong as solid shank rivets. Pin rivets are essentially threadless bolts. The pin is headed at one end and is grooved about the circumference at the other. A metal collar is swaged onto the grooved end effecting a firm, tight fit. (***Figure 5-64***)

Pin rivets are fabricated in a variety of materials but should be used only in shear applications. They should never be used where the grip length is less than the shank diameter. Part numbers for pin rivets can be interpreted to give the diameter and grip length of the individual rivets.

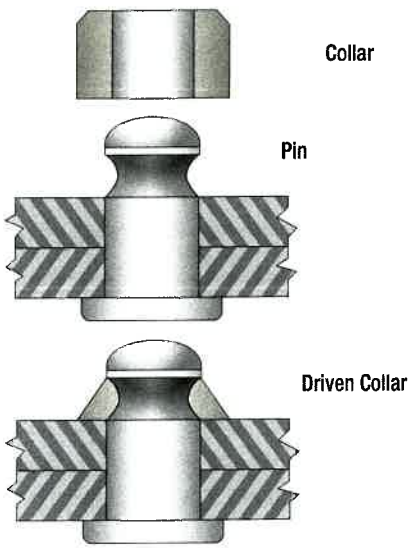


Figure 5-64. Pin (Hi-Shear) rivet.

A typical part number breakdown would be:
NAS177-14-17
NAS = National Aircraft Standard
177 = 100° Countersunk head rivet
OR 178 = Flathead rivet
14 = Nominal diameter in 32nds of an inch
17 = Maximum grip length in 16ths of an inch

TAPER-LOK

Taper-Loks are the strongest special fasteners used in aircraft construction. The Taper-Lok exerts force on the walls of the hole because of its tapered shape. The Taper-Lok is designed to completely fill the hole, but unlike the rivet, it fills the hole without deforming the shank. Instead, the washer head nut squeezes the metal with tremendous force against the tapered walls of the hole. This creates radial compression around the shank and vertical compression lines as the metals are squeezed together. The combination of these forces generates strength unequaled by any other fastener. (***Figure 5-65***)

HI-TIGUE

The Hi-Tigue special fastener has a bead that encircles the bottom of its shank. The bead preloads the hole it fills, resulting in increased joint strength. At installation, the bead presses against the sidewall of the hole, exerting radial force that strengthens the surrounding area. Because it is preloaded, the joint is not subjected to the constant cyclic action that normally causes a joint to become cold worked and eventually fail. Hi-Tigue fasteners are made of aluminum, titanium, and stainless steel alloys. The collars are composed of compatible metal alloys and come in two types: sealing and non-sealing. Just like the Hi-Loks, they can be installed using an Allen wrench and a box-end wrench. (***Figure 5-66***)

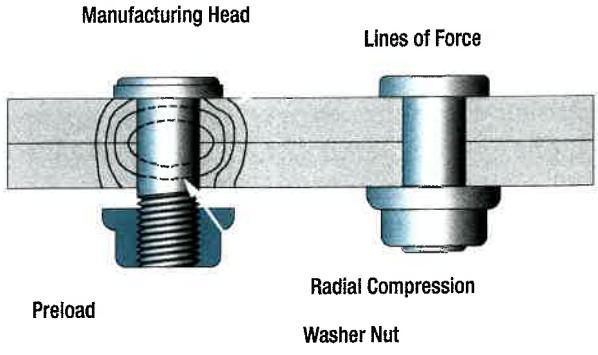


Figure 5-65. Taper-Lok rivets.

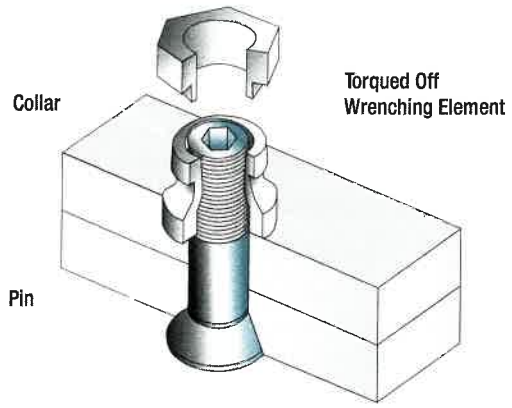


Figure 5-66. Hi-Tigue special fasteners.

DILL LOK-SKRUS AND RIVETS

Dill "Lok-Skru" and "Lok-Rivet" are trade names for internally threaded rivets. They are used for blind attachment of such accessories as fairings, fillets, access door covers, door and window frames, floor panels, and the like. Lok-Skrus and Lok-Rivets are similar to the Rivnut in appearance and application; however, they come in two parts and require more clearance on the blind side than the Rivnut to accommodate the barrel. (***Figure 5-67***)

The Lok-Rivet and the Lok-Skru are alike in construction, except the Lok-Skru is tapped internally for fastening an accessory by using an attaching screw, whereas the Lok-Rivet is not tapped and can be used only as a rivet. Since both Lok-Skrus and Lok-Rivets are installed in the same manner, the following discussions for the Lok-Skru also applies to the Lok-Rivet.

The main parts of a Lok-Skru are the barrel, the head, and an attachment screw. The barrel is made of aluminum alloy and comes in either closed or open ends. The head is either aluminum alloy or steel, and the attachment screw is made of steel. All of the steel parts are cadmium plated, and all of aluminum parts are anodized to resist corrosion. When installed, the barrel screws up over the head and grips the metal on the blind side. The attaching screw is then inserted if needed. There are two head types: the flathead and the countersunk head.

The Lok-Skru is tapped for 7-32, 8-32, 10-32, or 10-24 screws, and the diameters vary from 0.230-inch for 6-32 screws, to 0.292 inch for 10-32 screws. Grip ranges vary from 0.010 inch to 0.225 inch.

DEUTSCH RIVETS

This rivet is a high strength blind rivet used on late model aircraft. It has a minimum shear strength of 75 000 psi, and can be installed by one person. The Deutsch rivet consists of two parts: the stainless steel sleeve and the hardened steel drive pin. (***Figure 5-68***) The pin and sleeve are coated with a lubricant and a corrosion inhibitor.

The Deutsch rivet is available in diameters of 3⁄16, 1⁄4, or 3⁄8 inch. Grip lengths for this rivet range from 3⁄16 to 1 inch. Some variation is allowed in grip length when installing the rivet; for example, a rivet with a grip length of 3⁄16 inch can be used where the total thickness of materials is between 0.198 and 0.228 inch.

When driving a Deutsch rivet, an ordinary hammer or a pneumatic rivet gun and a flathead set are used. The rivet is seated in the previously drilled hole and then the pin is driven into the sleeve. The driving action causes the pin to exert pressure against the sleeve and forces the sides of the sleeve out. This stretching forms a shop head on the end of the rivet and provides positive fastening. The ridge on the top of the rivet head locks the pin into the rivet as the last few blows are struck.

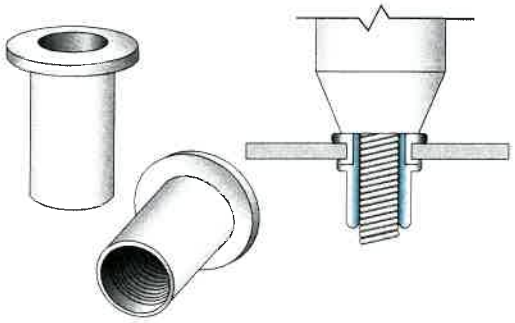


Figure 5-67. Internally threaded rivet (Rivnut).

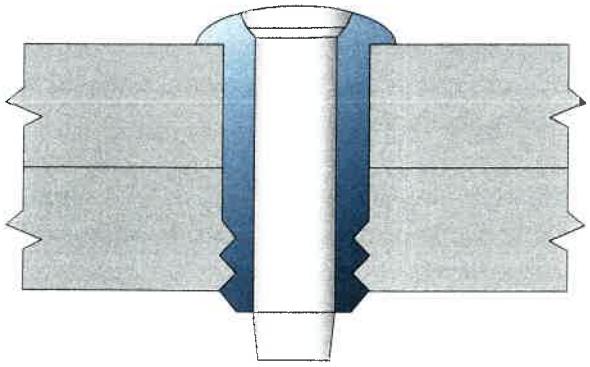


Figure 5-68. Deutsch rivet.

HOLE REPAIR HARDWARE

Many of the blind fasteners are manufactured in oversized diameters to accommodate slightly enlarged holes resulting from drilling out the original fastener. When using rivets or even bolts, care must be taken to ensure the hole is not elongated or slanted.

To reduce the chances of an incorrectly drilled rivet or bolt hole, use a slightly smaller drill bit first, then enlarge to the correct diameter. The last step to prepare the hole for the fastener is to deburr the hole using either a very large drill bit or a special deburring tool. This practice also works well when drilling out a previously attached fastener. If the drill bit does not exactly find the center of the rivet or bolt or screw, the hole can easily be elongated, but when using a smaller drill bit, drill the head only off the fastener, then the ring and stem that is left can be pushed out with a pin punch of the appropriate diameter. If an incorrectly drilled hole is found, the options are to redrill the hole to the next larger diameter for an acceptable fastener or repair the hole using an Acres fastener sleeve.

ACRES FASTENER SLEEVES

Acres fastener sleeves are thin-wall tubular elements with a flared end. The sleeves are installed in holes to accept standard bolts and rivet-type fasteners. The existing fastener holes are drilled 1/64 inch oversize for installation of the sleeves. The sleeves are manufactured in 1 inch increments. Along their length, grooves provide a place to break or cut off excess length to match fastener grip range. The grooves also provide a place to hold adhesive or sealing agents when bonding the sleeve into the hole. The sleeves are used in holes which must be drilled 1/64 inch oversize to clean up corrosion or other damage. The oversize hole with the sleeve installed allows the use of the original diameter fastener in the repaired hole. The sleeves can be used in areas of high galvanic corrosion where the corrosion must be confined to a readily replaceable part. Oversizing of holes reduces the net cross-sectional area of a part and should not be done unless absolutely required.

Consult the manufacturer of the aircraft, engine or aircraft component prior to repair of damaged holes with Acres sleeves.

IDENTIFICATION

The sleeve is identified by a standard code number (***Figure 5-69***) which represents the type and style of sleeve, a material code, the fastener shank diameter, surface finish code letter and grip tang for the sleeve. The type and material of the sleeve is represented by the basic code number. The first dash number represents the diameter of the sleeve for the fastener installed and the second dash represents the grip length of the sleeve. The required length of the sleeve is determined on installation and the excess is broken off of the sleeve. A JK5512A-05N-10 is a 100° low profile head sleeve of aluminum alloy. The diameter is for a 5/32 inch fastener with no surface finish and is 5/8 inch in length.

HOLE PREPARATION

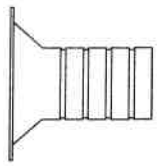
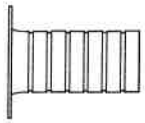
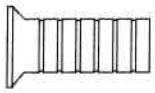
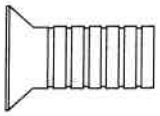
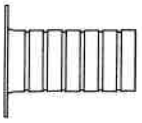
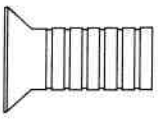
Reference ***Figure 5-70*** for drill number for standard or close fit holes. Inspect hole after drilling to assure all corrosion is removed before installing the sleeve. The hole must also be the correct shape and free from burrs. The countersink must be enlarged to receive the flare of the sleeve so the sleeve is flush with the surrounding surface.

INSTALLATION

After selecting the correct type and diameter sleeve, use the 6501 sleeve breakoff tool for final installation length. Reference ***Figure 5-70*** for the sleeve breakoff procedure. The sleeve may be installed with or without being bonded in the hole. When bonding the sleeve in a hole, use MIL-S-8802A1/2 sealant. Reinstall original size fastener and torque as required.

SLEEVE REMOVAL

Sleeves not bonded in the hole may be removed by either driving them out with a drift pin of the same diameter as the outside diameter of the sleeve or they may be deformed and removed with a pointed tool. Bonded sleeves may be removed by this method, but care should be used not to damage the structure hole. If this method cannot be used, drill the sleeves out with a drill 0.004 to 0.008 inch smaller than the installation drill size. The remaining portion of the sleeve after drilling can be removed using a pointed tool and applying an adhesive solvent to the sealant.

Acres Sleeve	Type	Basic Part Number	Sleeve Part No.	Bolt Size	[2] Sleeve Length
	100° Tension head plus flange (509 type)	JK5610	JK5511( )04( )( ) JK5512( )04( )( ) JK5516( )04( )( ) JK5517( )04( )( )	1/8	8
			JK5511( )45( )( ) JK5512 JK5516( )45( )( ) JK5517( )45( )( )	#6	8
	Protruding head (shear)	JK5511	JK5511( )05( )( ) JK5512( )05( )( ) JK5516( )05( )( ) JK5517( )05( )( )	5/32	10
	100° Low profile head	JK5512			
	100° Standard profile head (509 type)	JK5516	JK5511( )55( )( ) JK5512( )55( )( ) JK5516( )55( )( ) JK5517( )55( )( ) JK5610( )55( )( )	#8	10
	Protruding head (tension)	JK5517	JK5511( )06( )( ) JK5512( )06( )( ) JK5516( )06( )( ) JK5517( )06( )( ) JK5610( )06( )( )	#10	12
	100° Oversize tension head (1/64 oversize bolt)	JK5533	JK5511( )08( )( ) JK5512( )08( )( ) JK5516( )08( )( ) JK5517( )08( )( ) JK5610( )08( )( )	1/4	16
			JK5511( )10( )( ) JK5512( )10( )( ) JK5516( )10( )( ) JK5517( )10( )( ) JK5610( )10( )( )	5/16	16
			JK5511( )12( )( ) JK5512( )12( )( ) JK5516( )12( )( ) JK5517( )12( )( ) JK5610( )12( )( )	3/8	16

Part Number Breakdown

JK5511 A 04 N 08L

- JK5511 | Basic part number
- A | Material Code [1]
- 04 | Fastener shank diameter in 32nds
- N | Surface finish
N = No finish
C = Chemical film per MIL-C-554
- 08 | Length in sixteenth-inch increments
(required installation length by breaking of at proper groove)
- L | "L" at end of part number indicates cetyl alcohol

Material	Material Code
5052 Aluminium alloy ½ hard	A
6061 Aluminium alloy (T6 condition)	B
A286 Stainless steel (passivate)	C

Acres Sleeve for 1/64 Oversize Bolt

[1] Sleeve Part No.	Bolt Size	[2] Sleeve Length
JK5533( )06( )( )	13/64	12
JK5533( )08( )( )	17/64	16
JK5533( )10( )( )	21/64	16
JK5533( )12( )( )	25/64	16

Notes:

[1] Acres sleeve, JK5533 1/64 oversize available in A286 steel only

[2] Acres sleeve length in sixteenth-inch increments

Figure 5-69. Acres sleeve identification.

Hole Preparation for 1/64 Oversize Bolt

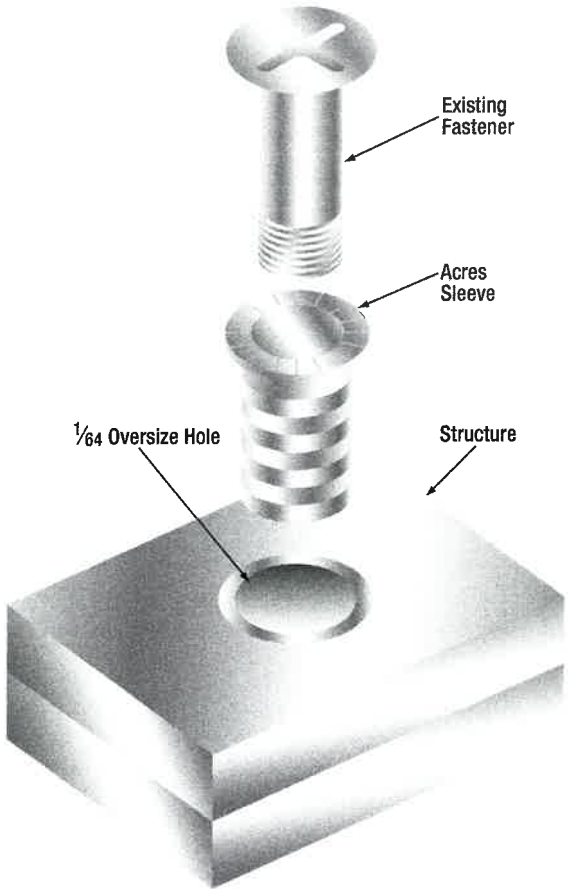
Bolt Size	Drill No.	Drill Dia.
13/64	7/32	0.2187
17/64	9/32	0.2812
21/64	11/32	0.3437
25/64	13/32	0.4062

Hole Preparation

Bolt Size	Standard Fit		Close Fit	
	Drill No.	Drill Dia.	Drill No.	Drill Dia.
1/8	9/64	0.1406	28	0.1405
#6	23	0.1540	24	0.1520
5/32	11/64	0.1719	18	0.1695
#8	15	0.1800	16	0.1770
#10	5	0.2055	6	0.2040
1/4	14	0.2660	17/64	0.2656
5/16	21/64	0.3281		
3/8	25/64	0.3908		

Installation Procedure

A. Drill out corrosion or damage to existing hole to 1/64 oversize.
B. Select proper type and length acres sleeve for existing fastener.
C. Bond sleeve in structure hole with MIL-S-8802 class A ½ sealant.



Acres Sleeve Installation

Figure 5-70. Acres sleeve identification, installation, and breakoff procedure.

Question: 5-1
The grip or grip length of a bolt is the area on the shank _____ threads.

Question: 5-2
_____ bolts are used only where shear loads occur and never in tension.

Question: 5-3
_____ nuts contain a locking feature as an integral part their construction.

Question: 5-4
If a fiber lock nut can be threaded onto a fastener with just finger strength, it must be _____.

Question: 5-5
What prevents a lock washer from loosening from vibration.

Question: 5-6
Which type of stud would used when weight is a major concern?

Question: 5-7
The second head of a rivet, formed either by hand or by pneumatic equipment, is called a _____.

Question: 5-8
What are two advantages of a Deutsch Rivet?

ANSWERS

Answer: 5-1
without.

Answer: 5-2
Clevis.

Answer: 5-3
Self-locking.

Answer: 5-4
replaced.

Answer: 5-5
The spring action of the washer.

Answer: 5-6
Waisted stud.

Answer: 5-7
Shop Head.

Answer: 5-8
Very high strength blind river; can be installed by a single person.

Question: 5-9
A _____ rivet is used where access to both sides of the rivet is impossible.

Question: 5-10
Self-plugging (mechanical lock) rivets display all the strength characteristics of _____ and in most cases can be substituted rivet for rivet.

Question: 5-11
Special fasteners are typically held in place with a _____ that is squeezed into position.

Question: 5-12
A _____ screw differs only from a standard structural bolt by the head style.

Question: 5-13
What are the head angles of countersunk Rivnuts?

Question: 5-14
_____ fastener sleeves are used in areas of high galvanic corrosion where it is better to have a readily replaceable part corrode that the component being fastened.

Question: 5-15
When safety wiring two or more bolts together, the wire is strung so that any tendency one bolt to loosen is counteracted by the _____ of the wire and the other bolt(s).

Question: 5-16
The prong bent over the bolt should not extend beyond the bolt diameter and the prong bent down should not rest against the surface of the washer. These are instructions for installing _____.

ANSWERS

Answer: 5-9
blind.

Answer: 5-10
solid shank rivets.

Answer: 5-11
collar.

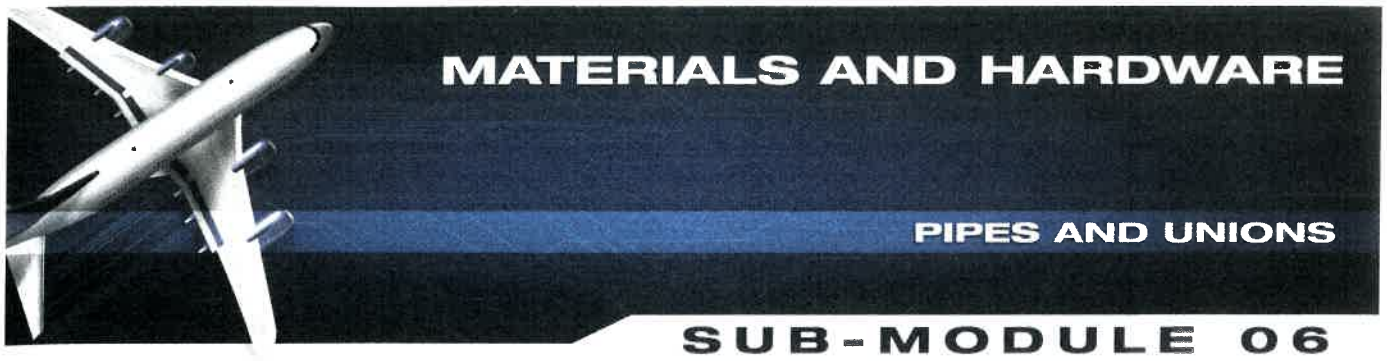
Answer: 5-12
structural.

Answer: 5-13
100°.
115°.

Answer: 5-14
Acres.

Answer: 5-15
tightening.

Answer: 5-16
a cotter pin.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY → B1

Sub-Module 06

PIPES AND UNIONS

Knowledge Requirements

6.6 - Pipes and Unions

(a)	Identification of, and types of rigid and flexible pipes and their connectors used in aircraft;	2
(b)	Standard unions for aircraft hydraulic, fuel, oil, pneumatic and air system pipes.	2

6.6 - PIPES AND UNIONS

PIPES AND UNIONS

Aircraft fluid lines are usually made of metal tubing or flexible hose. Metal tubing (also called rigid fluid line) is used in stationary applications and where long, relatively straight runs are possible. They are widely used in aircraft for fuel, oil, coolant, oxygen, instrument, and hydraulic lines. Flexible hose is generally used with moving parts or where the hose is subject to considerable vibration.

6.6(A) - TYPES OF RIGID AND FLEXIBLE PIPES

RIGID FLUID LINES

TUBING MATERIAL

Copper Tubing

In the early days of aviation, copper tubing was used extensively in aviation fluid applications. In modern aircraft, aluminum alloy, corrosion resistant steel or titanium tubing have generally replaced copper tubing.

Aluminum Alloy Tubing

Tubing made from 1100 H14 (½ hard) or 3003 H14 (½ hard) is used for general purpose lines of low or negligible fluid pressures, such as instrument lines and ventilating conduits. Tubing made from 2024-T3, 5052-O, and 6061-T6 aluminum alloy materials is used in general purpose systems of low and medium pressures, such as hydraulic and pneumatic 1 000 to 1 500 psi systems, and fuel and oil lines.

Steel Tubing

Corrosion resistant steel tubing, either annealed CRES 304, CRES 321 or CRES 304-⅛ hard, is used extensively in high pressure hydraulic systems (3 000 psi or more) for the operation of landing gear, flaps, brakes, and in fire zones. Its higher tensile strength permits the use of tubing with thinner walls; consequently, the final installation weight is not much greater than that of the thicker wall aluminum alloy tubing. Steel lines are used where there is a risk of Foreign Object Damage (FOD); i.e., the landing gear and wheel well areas. Although identification markings for steel tubing differ, each usually includes the manufacturer's name or trademark, the Society of Automotive Engineers (SAE) number, and the physical condition of the metal.

Titanium Tubing 3AL-2.5V

This type of tubing and fitting is used extensively in transport category and high performance aircraft hydraulic systems for pressures above 1 500 psi. Titanium is 30 percent stronger than steel and 50 percent lighter than steel. Cryofit fittings or swaged fittings are used with titanium tubing. Do not use titanium tubing and fittings in any oxygen system assembly. Titanium and titanium alloys are oxygen reactive. If a freshly formed titanium surface is exposed in gaseous oxygen, spontaneous combustion could occur at low pressures.

MATERIAL IDENTIFICATION

Before making repairs to any aircraft tubing, it is important to make accurate identification of tubing materials. Aluminum alloy, steel, or titanium tubing can be identified readily by sight where it is used as the basic tubing material. However, it is difficult to determine whether a material is carbon steel or stainless steel, or whether it is 1100, 3003, 5052-O, 6061-T6 or 2024-T3 aluminum alloy. To positively identify the material used in the original installation, compare code markings of the replacement tubing with the original markings on the tubing being replaced.

On large aluminum alloy tubing, the alloy designation is stamped on the surface. On small aluminum tubing, the designation may be stamped on the surface; but more often it is shown by a color code, not more than 4" in width, painted at the two ends and approximately midway between the ends of some tubing. When the band consists of two colors, one half the width is used for each color. (***Figure 6-1***)

If the code markings are hard or impossible to read, it may be necessary to test samples of the material for hardness by hardness testing.

TUBING SIZES

Metal tubing is sized by outside diameter (OD), which is measured fractionally in sixteenths of an inch. For example, number 6 tubing is 6/16" (or ⅜") and number 8 tubing is 8/16" (or ½") and so forth. The tube diameter is printed on all rigid tubing. In addition to other

Aluminium Alloy Number	Color of Band
1100	White
3003	Green
2014	Gray
2024	Red
5052	Purple
6053	Black
6061	Blue and Yellow
7075	Brown and Yellow

Figure 6-1. Painted color codes used to identify aluminum alloy tubing.

classifications or means of identification, tubing is manufactured in various wall thicknesses. Thus, it is important when installing tubing to know not only the material and outside diameter, but also the thickness of the wall. The wall thickness is printed on the tubing in thousandths of an inch. To determine the inside diameter (ID) of the tube, subtract twice the wall thickness from the outside diameter.

For example, a number 10 piece of tubing with a wall thickness of 0.063" has an inside diameter of 0.625"-2(0.063") = 0.499".

FLEXIBLE HOSE FLUID LINES

Flexible hose is used in aircraft fluid systems to connect moving parts with stationary parts in locations subject to vibration or where a great amount of flexibility is needed. It can also serve as a connector in metal tubing systems.

RUBBER HOSE MATERIALS AND CONSTRUCTION

Pure rubber is never used in the construction of flexible fluid lines. To meet the requirements of strength, durability, and workability, among other factors, synthetics are used in place of pure rubber. Synthetic materials most commonly used in the manufacture of flexible hose are Buna-N, Neoprene, Butyl, Ethylene Propylene Diene Rubber (EPDM) and Teflon™. While Teflon™ is in a category of its own, the others are synthetic rubber.

Flexible rubber hose consists of a seamless synthetic rubber inner tube covered with layers of cotton braid and wire braid and an outer layer of rubber-impregnated cotton braid. This type of hose is suitable for use in fuel, oil, coolant, and hydraulic systems. The types of hose are normally classified by the amount of pressure they are designed to withstand under normal operating conditions.

Buna-N

Buna-N is a synthetic rubber compound which has excellent resistance to petroleum products. Do not confuse with Buna-S. Do not use for phosphate ester base hydraulic fluid (Skydrol®).

Neoprene

Neoprene is a synthetic rubber compound which has an acetylene base. Its resistance to petroleum products is not as good as Buna-N, but it has better abrasive resistance. Do not use for phosphate ester base hydraulic fluid (Skydrol®).

Butyl

Butyl is a synthetic rubber compound made from petroleum raw materials. It is an excellent material to use with phosphate ester base hydraulic fluid (Skydrol®). Do not use with petroleum products.

LOW, MEDIUM, AND HIGH PRESSURE HOSES

- Low pressure — below 250 psi.
  Fabric braid reinforcement.
- Medium pressure — up to 3 000 psi.
  One wire braid reinforcement.
  Smaller sizes carry up to 3 000 psi.
  Larger sizes carry pressure up to 1 500 psi.
- High pressure — all sizes up to 3 000 psi operating pressures.

TEFLON™ HOSE

Teflon™ is the DuPont trade name for tetrafluoroethylene resin. It has a broad operating temperature range (-54°C to 232°C). It is compatible with nearly every substance or agent used. It offers little resistance to flow; sticky, viscous materials will not adhere to it. It has less volumetric expansion than rubber, and the shelf and service life is practically limitless. Teflon™ hose is flexible and designed to meet the requirements of higher operating temperatures and pressures in present aircraft systems.

Generally, it may be used in the same manner as rubber hose. Teflon™ hose is processed and extruded into tube shape to a desired size. It is covered with stainless steel wire, which is braided over the tube for strength and protection. Teflon™ hose is unaffected by any known fuel, petroleum, or synthetic base oils, alcohol, coolants, or solvents commonly used in aircraft. Teflon™ hose has the distinct advantages of a practically unlimited storage time, greater operating temperature range, and broad usage (hydraulic, fuel, oil, coolant, water, alcohol, and pneumatic systems). Medium-pressure Teflon™ hose assemblies are sometimes preformed to clear obstructions and to make connections using the shortest possible hose length. Since preforming permits tighter bends that eliminate the need for special elbows, preformed hose assemblies save space and weight. Never straighten a preformed hose assembly. Use a support wire if the hose is to be removed for maintenance. (***Figure 6-2***)

HOSE IDENTIFICATION

Lay lines and identification markings consisting of lines, letters, and numbers are printed on the hose. Most hydraulic hose is marked to identify its type, the quarter and year of manufacture, and a 5 digit code identifying the manufacturer. These markings are in contrasting colored letters and numerals which indicate the natural lay (no twist) of the hose and are repeated at intervals of not more than 9 inches along the length of the hose. Code markings assist in replacing a hose with one of the same specifications or a recommended substitute. Hose suitable for use with phosphate ester base hydraulic fluid will be marked Skydrol® use. In some instances, several types of hose may be suitable for the same use. Therefore, to make the correct hose selection, always refer to the applicable aircraft maintenance or parts manual.

Size Designations

Hose is also designated by a dash number according to its size. The dash number is stenciled on the side of the hose and indicates the size tubing with which the hose is compatible. It does not denote inside or outside diameter. When the dash number of the hose corresponds with the dash number of the tubing, the proper size hose is being used. (***Figure 6-3***)

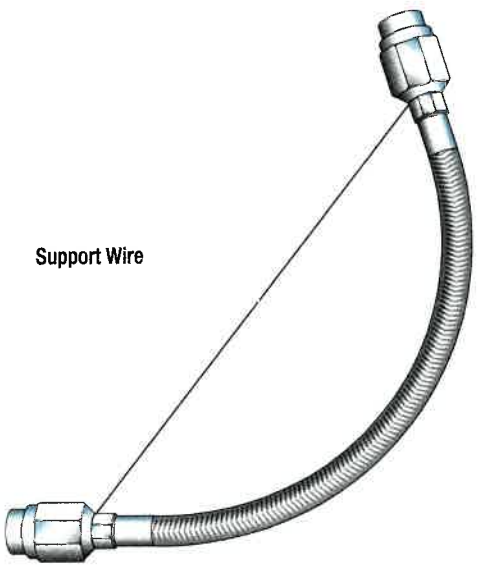


Figure 6-2. Suggested handling of preformed hose.

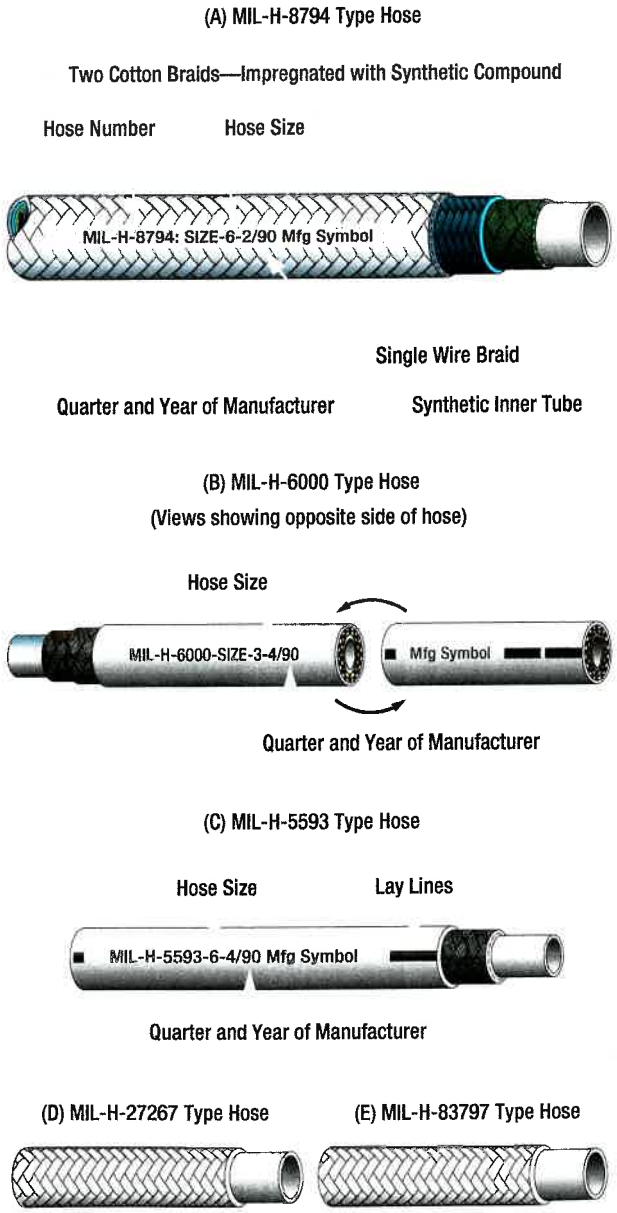


Figure 6-3. Hose identification markings.

FLUID LINE IDENTIFICATION

Fluid lines in aircraft are often identified by markers made up of color codes, words, and geometric symbols. These markers identify each line's function, content, and primary hazard. ***Figure 6-4*** illustrates the various color codes and symbols used to designate the type of system and its contents.

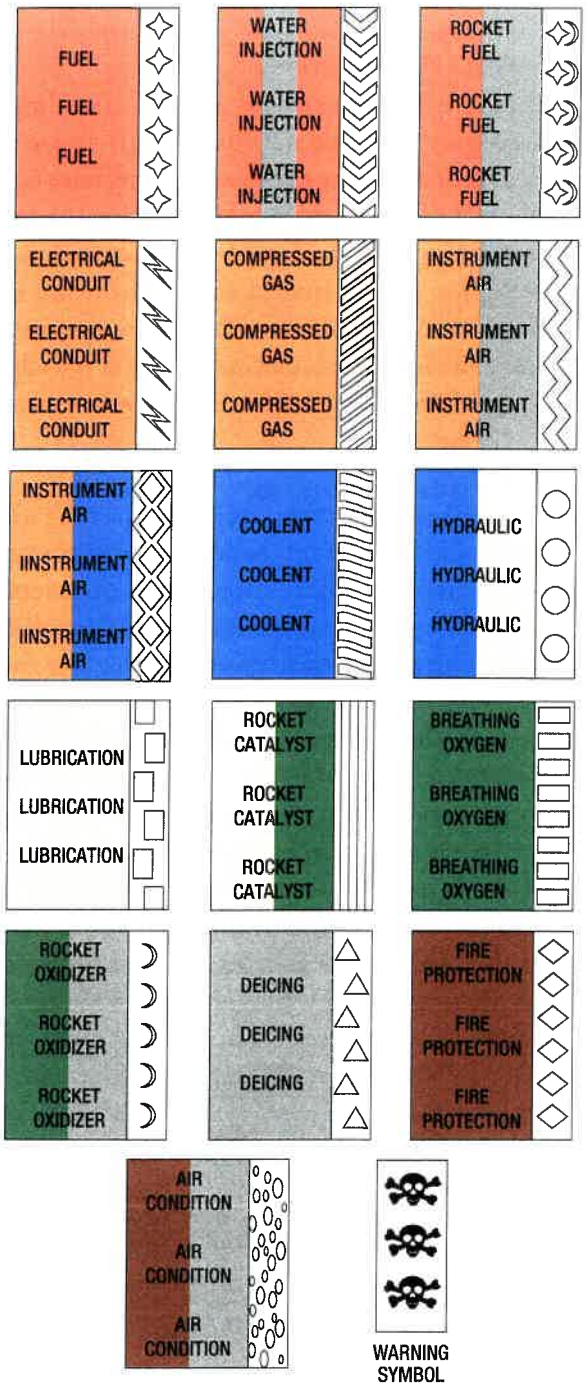


Figure 6-4. Identification of aircraft fluid lines.

Fluid lines are marked, in most instances with 1" tape or decals, as shown in ***Figure 6-5(A)***. On lines 4" in diameter (or larger), lines in oily environment, hot lines, and on some cold lines, steel tags may be used in place of tape or decals, as shown in ***Figure 6-5(B)***. Paint is used on lines in engine compartments, where there is the possibility of tapes, decals, or tags being drawn into the engine induction system.

In addition to the above mentioned markings, certain lines may be further identified regarding specific function within a system; for example, drain, vent, pressure, or return. Lines conveying fuel may be marked FLAM; lines containing toxic materials are marked TOXIC in place of FLAM. Lines containing physically dangerous materials, such as oxygen, nitrogen, or Freon™, may be marked PHDAN.

Aircraft and engine manufacturer's are responsible for the original installation of identification markers, but the aviation mechanic is responsible for their replacement when it becomes necessary. Tapes and decals are generally placed on both ends of a line and at least once in each compartment through which the line runs. In addition, identification markers are placed immediately adjacent to each valve, regulator, filter, or other accessory within a line. Where paint or tags are used, location requirements are the same as for tapes and decals.

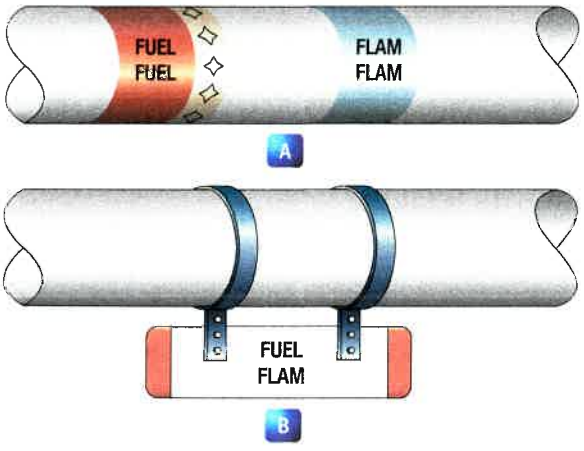


Figure 6-5. Fluid line identification using: (A) tape and decals and (B) metal tags.

6.6(B) - UNIONS FOR HYDRAULIC, FUEL, OIL, AND PNEUMATIC SYSTEMS

FLUID LINE END FITTINGS

Depending on the type and use, fittings will have either pipe threads or machine threads. Pipe threads are similar to those used in ordinary plumbing and are tapered, both internal and external. External threads are referred to as male threads and internal threads are female threads.

When two fittings are joined, a male into a female, the thread taper forms a seal. Some form of pipe thread lubricant approved for the particular fluid application should be used when joining pipe threads to prevent seizing and high pressure leakage. Use care when applying thread lubricant so that the lubricant will not enter and contaminate the system. Do not use lubricants on oxygen lines. Oxygen will react with petroleum products and can ignite (special lubricants are available for oxygen systems).

Machine threads have no sealing capability and are similar to those used on common nuts and bolts. This type of fitting is used only to draw connections together or for attachment through bulkheads. A flared tube connection, a crush washer, or a synthetic seal is used to make the connection fluid tight. Machine threads have no taper and will not form a fluid tight seal. The size of these fittings is given in dash numbers, which equal the nominal outside diameter in sixteenths of an inch.

UNIVERSAL BULKHEAD FITTINGS

When a fluid line passes through a bulkhead, and it is desired to secure the line to the bulkhead, a bulkhead fitting should be used. The end of the fitting that passes through the bulkhead is longer than the other end(s), which allows a locknut to be installed, securing the fitting to the bulkhead.

Fittings attach one piece of tubing to another, or to system units. There are four types: (1) bead and clamp, (2) flared fittings, (3) flareless fittings, and (4) permanent fittings (Permaswage™, Permalite™, and Cyrofit™).

The amount of pressure that the system carries and the material used are usually the deciding factors in selecting a connector.

The beaded type of fitting, which requires a bead and a section of hose and hose clamps, is used only in low or medium pressure systems, such as vacuum and coolant systems. The flared, flareless, or permanent type fittings may be used as connectors in all systems, regardless of the pressure.

AN FLARED FITTINGS

A flared tube fitting consists of a sleeve and a nut, as shown in ***Figure 6-6***. The nut fits over the sleeve and, when tightened, draws the sleeve and tubing flare tightly against a male fitting to form a seal. Tubing used with this type of fitting must be flared before installation. The male fitting has a cone shaped surface with the same angle as the inside of the flare. The sleeve supports the tube so that vibration does not concentrate at the edge of the flare, and distributes the shearing action over a wider area for added strength.

Fitting combinations composed of different alloys should be avoided to prevent dissimilar metal corrosion. As with all fitting combinations, ease of assembly, alignment, and proper lubrication should be assured when tightening fittings during installation.

Standard AN fittings are identified by their black or blue color. All AN steel fittings are colored black, AN aluminum fittings are colored blue, and aluminum bronze fittings are cadmium plated and natural in appearance. ***Figure 6-7*** contains information on sizes, torques, and bend radii. A sampling of AN fittings are shown in ***Figure 6-8*** and ***Figure 6-9***. ***Figure 6-10*** illustrates this process.

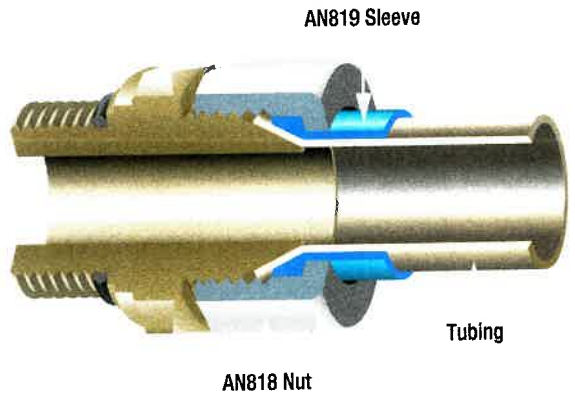


Figure 6-6. AN Flared tube fitting.

Tubing o.d. (inches)	Fitting Bolt or Nut Size	Aluminium Alloy Tubing Bolt Fitting or Nut Torque (in–lb)	Steel Tubing Bolt Fitting or Nut Torque (in–lb)	Hose End Fittings and Hose Assemblies MS28740 or Equivalent End Fitting		Minimum Bend Radii (inches)	
				Minimum	Maximum	Alum. alloy 1100-H14 5052-0	Steel
1/8	–2	20–30				3/8	
3/16	–3	30–40	90–100	70	120	7/16	21/32
1/4	–4	40–65	135–150	100	250	9/16	7/8
5/16	–5	60–85	180–200	210	420	3/4	1 1/8
3/8	–6	75–125	270–300	300	480	15/16	1 5/16
1/2	–8	150–250	450–500	500	850	1 1/4	1 3/4
5/8	–10	200–350	650–700	700	1 150	1 1/2	2 3/16
3/4	–12	300–500	900–1 000			1 3/4	2 5/8
7/8	–14	500–600	1 000–1 100				
1	–16	500–700	1 200–1 400			3	3 1/2
1 1/4	–20	600–900	1 200–1 400			3 3/4	4 3/8
1 1/2	–24	600–900	1 500–1 800			5	5 1/4
1 3/4	–28	850–1 050				7	6 1/8
2	–32	950–1 150				8	7

Figure 6-7. Flared fitting torque data.

FLARELESS FITTINGS

Flareless fittings are designed primarily for high pressure (3 000 psi) hydraulic systems that may be subjected to severe vibration or fluctuating pressure. Using this type of fitting eliminates all tube flaring, yet provides a safe and strong, dependable tube connection. (***Figure 6-9***) The fitting consists of three parts: a body, a sleeve, and a nut. (***Figure 6-10***) The internal design of the body causes the sleeve to cut into the outside of the tube when the body and nut are joined. The counterbore shoulder within the body is designed with a reverse angle of 15° for steel connectors and 45° for aluminum fittings. This reverse angle prevents inward collapse of the tubing when tightened and provides a partial sealing force to be exerted against the periphery of the body counterbore.

Although the use of flareless tube fittings eliminates all tube flaring, another operation, referred to as presetting, is necessary prior to installation of a new flareless tube assembly. Flareless tube assemblies should be preset with the proper size presetting tool or operation. The presetting operation, which is performed as follows: Cut the tube to the correct length, with the ends perfectly square. Debur the inside and outside of the tube. Slip the nut, then the sleeve, over the tube, lubricate the threads of the fitting and nut with hydraulic fluid. Place the fitting in a vise, and hold the tubing firmly and squarely on the seat in the fitting. (The tube must bottom firmly in the fitting.) Tighten the nut until the cutting edge of the sleeve grips the tube. To determine this point, slowly turn the tube back and forth while tightening the nut. When the tube no longer turns, the nut is ready for tightening. Final tightening depends upon the tubing. For aluminum alloy tubing up to and including ½" outside diameter, tighten the nut from 1 to 1⅙ turns. For steel tubing and aluminum alloy tubing over ½" outside diameter, tighten from 1⅙ to 1½ turns.

After presetting the sleeve, disconnect the tubing from the fitting and check the following points: The tube should extend 3/32" to ⅛" beyond the sleeve pilot; otherwise, blowoff may occur. The sleeve pilot should contact the tube or have a maximum clearance of 0.005" for aluminum alloy tubing or 0.015" for steel tubing. A slight collapse of the tube at the sleeve cut is permissible. No movement of the sleeve pilot, except rotation, is permissible.

AN744 to AN932

Material:
Aluminum Alloy.....(code D)
Steel .....(code, absence of letter)
Brass .....(code B)
Aluminum Bronze .....(code Z–for AN819 sleeve)

Size:
The dash number following the AN number indicates the size of the tubing (or hose) for which the fitting is made in 16ths of an inch. This size measures the outer diameter of tubing and the inner diameter of hose. Fittings that have pipe threads are coded by a dash number, indicating the pipe size in 8ths of an inch. The material code letter, as noted above, follows the dash number.

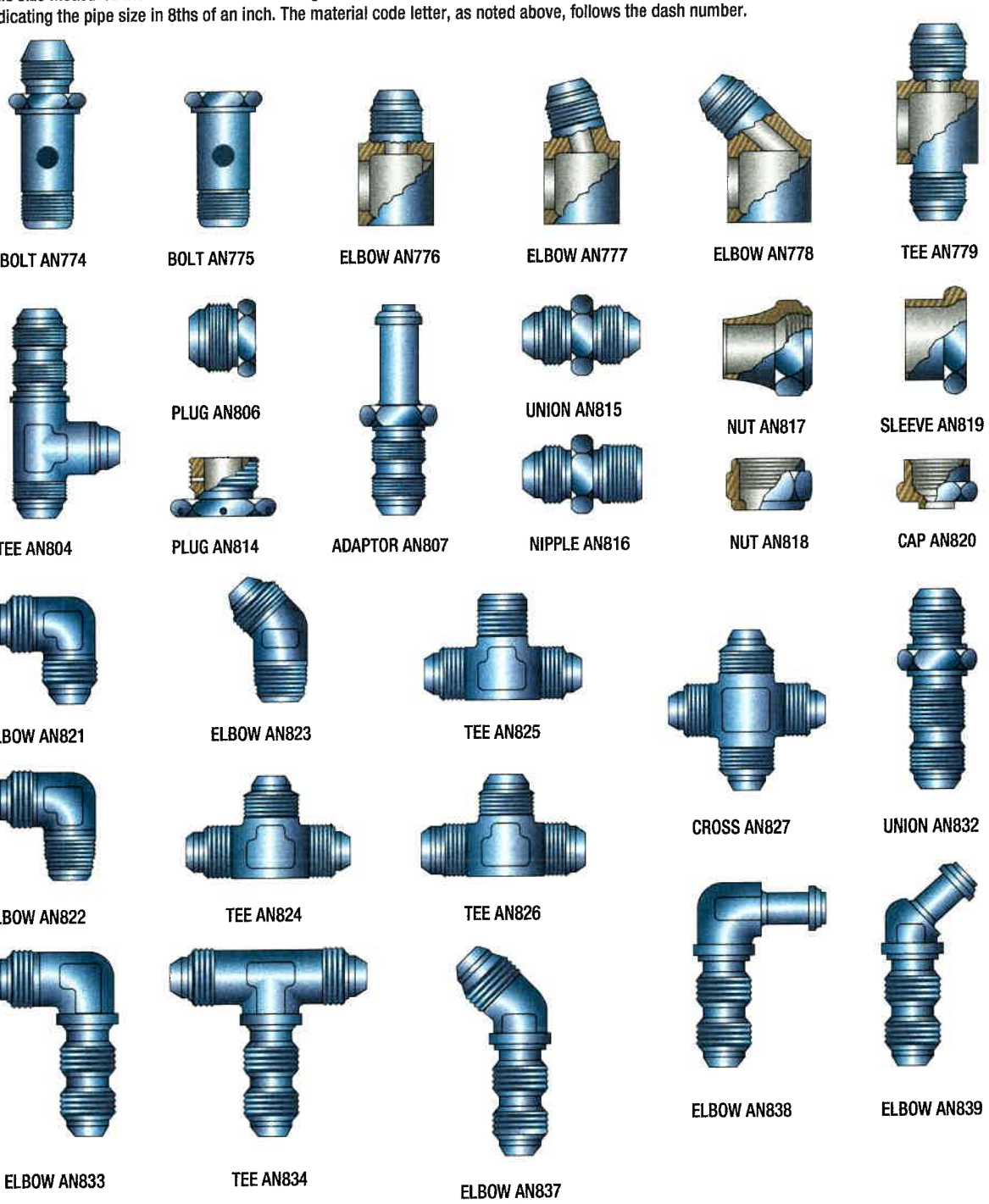


Figure 6-8. AN standard fittings.

AIRCRAFT TECHNICAL BOOK COMPANY

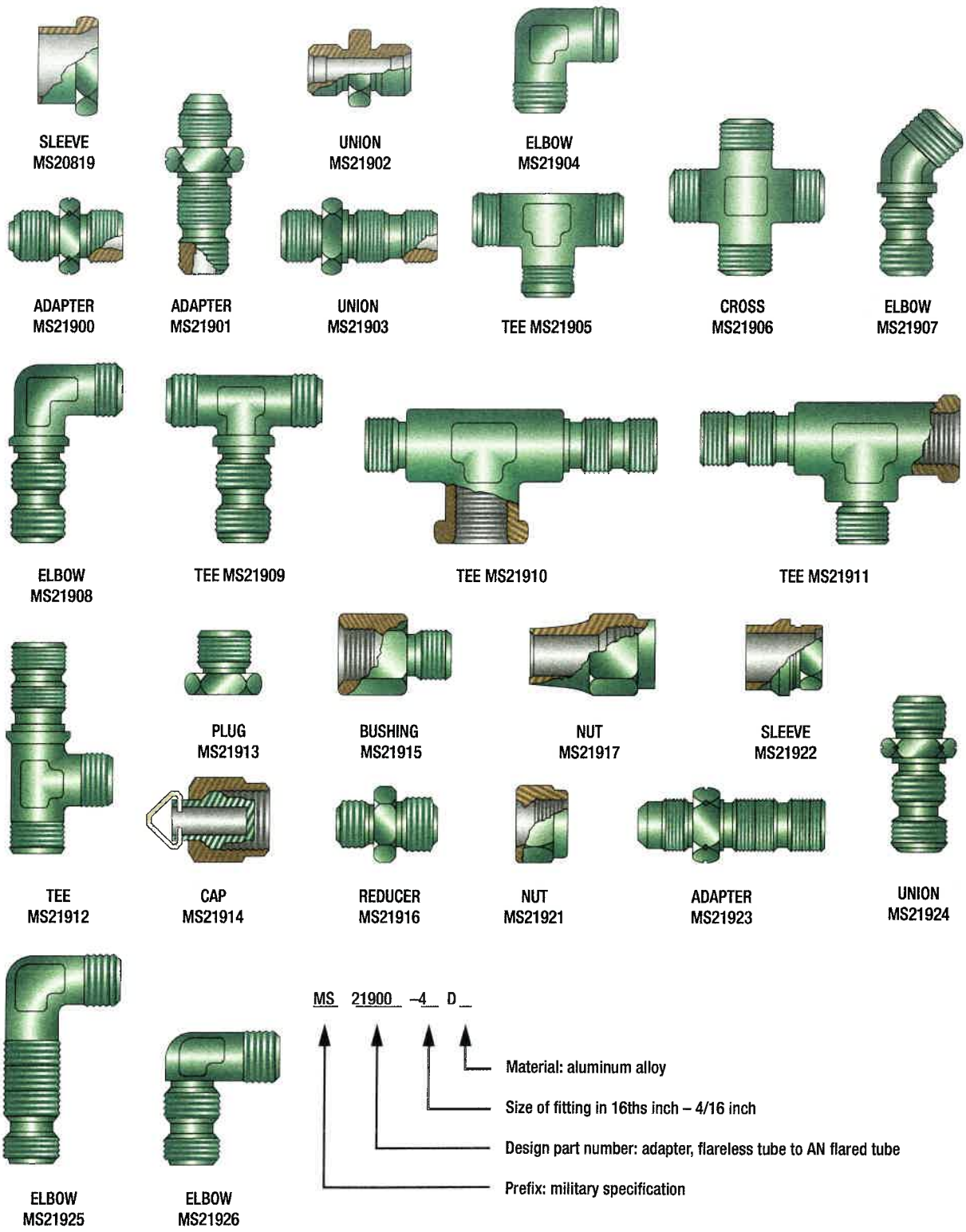
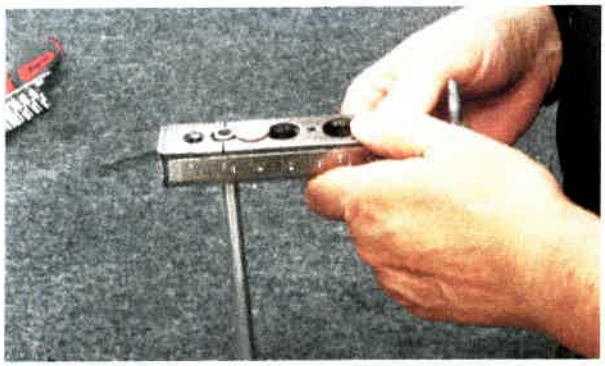


Figure 6-9. Typical MS flareless tube fittings.



Deburring inside of edge.



Setting tube in die (dime's thickness over top).

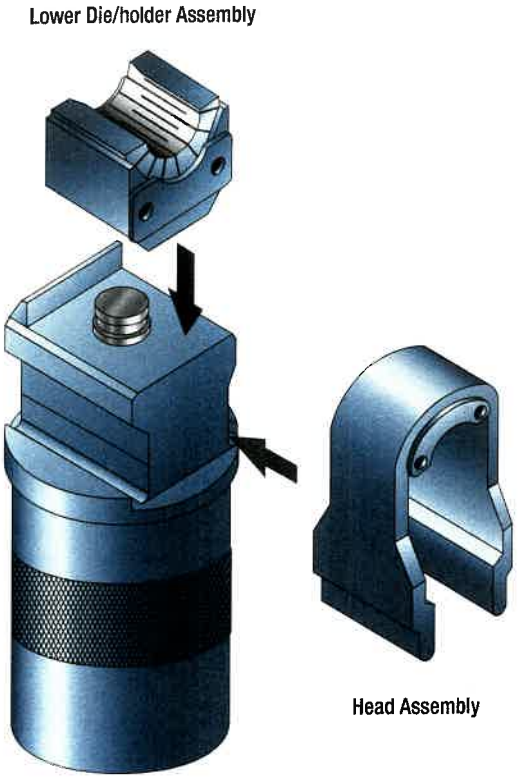


Ready to advance flaring cone.



Flared tube assembled with sleeve and nut.

Figure 6-10. Steps for flaring a tube.





6-11. Swaged fitting tooling.



Figure 6-12. Permaswage™ fitting.



Figure 6-13. Permalite™ fitting.

SWAGED FITTINGS

A popular repair system for connecting and repairing hydraulic lines on transport category aircraft is the use of Permaswage™ fittings. Swaged fittings create a permanent connection that is virtually maintenance free. Swaged fittings are used to join hydraulic lines in areas where routine disconnections are not required and are often used with titanium and corrosion resistant steel tubing. The fittings are installed with portable hydraulically powered tooling, which is compact enough to be used in tight spaces. (***Figure 6-11***) If the fittings need to be disconnected, cut the tubing with a tube cutter. Special installation tooling is available in portable kits. Always use the manufacturer's instructions to install swaged fittings. Typical Permaswage™ fittings are shown in ***Figure 6-12***.

One of the latest developments is the Permalite™ fitting. Permalite™ is a tube fitting that is mechanically attached to the tube by axial swaging. Permalite™ works by deforming the fitting into the tube being joined by moving a ring, a component of the Permalite™ fitting, axially along the fitting length using a Permaswage Axial swage tool. Typical Permalite™ fittings are shown in ***Figure 6-13***.

CRYOFIT™ FITTINGS

Many transport category aircraft use Cryofit™ fittings to join hydraulic lines in areas where routine disconnections are not required. Cryofit™ fittings are standard fittings with a cryogenic sleeve. The sleeve is made of a shape memory alloy, Tinel™. The sleeve is manufactured 3 percent smaller, frozen in liquid nitrogen, and expanded to 5 percent larger than the line. During installation, the fitting is removed from the liquid nitrogen and inserted onto the tube. During a 10 to 15 second warming up period, the fitting contracts to its original size (3 percent smaller), biting down on the tube, forming a permanent seal. Cryofit™ fittings can only be removed by cutting the tube at the sleeve, this leaves enough room to replace it with a swaged fitting without replacing the hydraulic line. It is frequently used with titanium tubing. The shape memory technology is also used for end, flared, and flareless fittings. (***Figure 6-14***)

FLEXIBLE HOSE FITTINGS

Flexible hose may be equipped with either swaged fittings or detachable fittings, or they may be used with beads and hose clamps. Hoses equipped with swaged fittings are ordered by correct length from the manufacturer and ordinarily cannot be assembled by the mechanic. They are swaged and tested at the factory and are equipped with standard fittings. The detachable fittings used on flexible hoses may be detached and reused if they are not damaged; otherwise, new fittings must be used. (***Figure 6-15***)

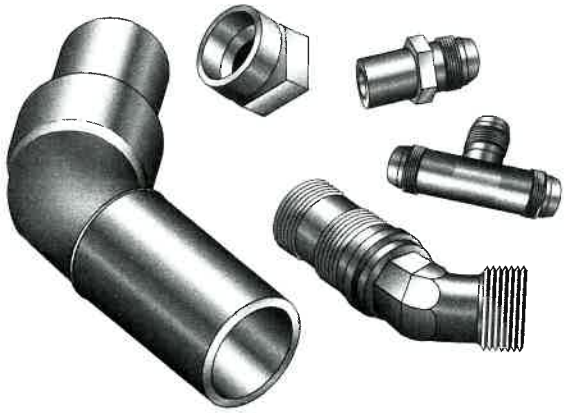


Figure 6-14. Cryofit fittings.

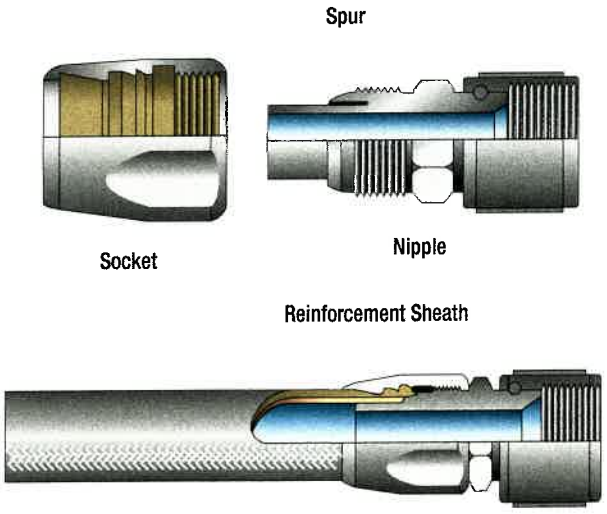


Figure 6-15. Reusable fittings for medium-pressure hose.

Question: 6-1
Rigid fluid lines constructed of _____ are for low to medium pressure (<1 500 psi) applications.

Question: 6-2
_____ are designed primarily for high pressure (3 000 psi) hydraulic systems that may be subjected to severe vibration or fluctuating pressure.

Question: 6-3
A fitting that is frozen and then allowed to shrink onto the fluid line while warming is a _____ fitting.

Question: 6-4
_____ are generally placed on both ends of a line and at least once in each compartment through which a line runs.

Question: 6-5
How would you identify a fluid line designated to carry fuel?

Question: 6-6
Name 4 advantages of Teflon flexible fluid lines versus rubber lines.

ANSWERS

Answer: 6-1
aluminum.

Answer: 6-2
Flareless fittings.

Answer: 6-3
cryogenic.

Answer: 6-4
Tapes and decals.

Answer: 6-5
Red markers showing including a 4 sided star; plus additional markers labeled FLAM designating flammable content.

Answer: 6-6
1. A larger range of acceptable operating temperatures;
2. May be used with most fluids;
3. Less internal resistance to viscous fluids;
4. Less expansion due to temperature and pressure variation.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY → B1

Sub-Module 07

SPRINGS

Knowledge Requirements

6.7 - Springs

Types of springs, materials, characteristics and applications. 2

6.7 - SPRINGS

TYPES OF SPRINGS AND APPLICATIONS

Springs are made in various shapes and sizes to meet the demand for the many applications in which they are used. Descriptions of common spring types and their applications follow.

FLAT SPRINGS

Flat Springs were originally developed from flat rectangular pieces of spring steel. Their shapes may vary but they are usually found as a single flat spring. (***Figure 7-1***) The contact breaker points spring in an aircraft magneto is an example of a flat spring.

LEAF SPRINGS

Leaf Springs are formed by layers of flat springs. Very early aircraft utilized leaf spring in their landing gear, however today, leaf spring are most common on automobiles and trains. (***Figure 7-2***)

SPIRAL SPRINGS

Spiral springs are formed by spirally winding a flat spring or a wire. This type of spring is sometimes known as a motor spring or a power spring. Spiral springs are found in analog instruments and are the springs used to power a wind-up clock or watch. (***Figure 7-3***)

HELICAL COMPRESSION AND TENSION SPRINGS

The most common type of spring is the helical wound spring. (***Figure 7-4***) These springs can be used for compression or tension loads. They come is a very wide variety of sizes and materials and are used in numerous applications. Helical springs used for tension are the same as those used for compression. The difference is that the tension spring typically has a hook or a loop on each end so that it can be attached and pulled. Reciprocating engine valve springs are an example of helical compression springs. When they are used in pairs, the compression springs, situated one inside of the other, are wound in opposite directions.

HELICAL TORSION SPRINGS

Helical torsion springs are similarly wound as helical tension and compression springs. They have specially shaped ends that engage a mechanism so that the spring can absorb a torque force when applied. (***Figure 7-5***)



Figure 7-1. A flat spring.



Figure 7-2. Example of leaf springs.



Figure 7-3. Spiral springs.



Figure 7-4. Helical compression springs and a tension spring.

BELLEVILLE (CONE DISC) SPRINGS

Belleville washers or disc springs are shown in ***Figure 7-6***. They can be used individually or stacked as shown for more deflection. The Belleville spring is typically centered axially by a shaft and can be produced to have non-linear load absorbing characteristics.

TORSION BAR SPRINGS

A rotary spring is a spring that absorbs torsion and angular deflection. The simplest rotary spring is the torsion bar. Torsion bar springs are basically strait bars of metal with each end fashioned to engage a mechanism so that a torsion load is transmitted to the bar. (***Figure 7-7***) Torsion bars are stiff when compared to other springs. They have many applications the most common of which is in automobile suspension.

SPRING MATERIALS

Springs are made from a wide variety of materials. Most are metallic but plastic, elastomer and composite springs are also common. Typical high carbon and alloy steel stock used to cold wind manufactured springs are:

1. Hard drawn spring wire (carbon steel stock) is an inexpensive spring wire used in applications with low stress and not where fatigue loading is exerted.
2. Oil tempered spring wire has a higher fatigue life and sometimes is used for valve springs.
3. Music wire is a high quality, carbon steel stock suitable for small helical springs used in applications involving high fatigue stresses.
4. Chrome vanadium steel wire is suitable for high temperature and high stress conditions. This type is used to make valve springs for reciprocating engines.
5. Chrome silicon steel wire is also used for valve springs but has a higher fatigue life.
6. Stainless steel spring wire has excellent corrosion resistant characteristics and low creep at high temperatures.

Springs can also be hot wound. This is especially done with large diameter stock after which various stress relieving processes can be administered. The same types of steels are used when hot-winding springs as are used to cold wind springs except necessary adjustments in the carbon, manganese, chromium, molybdenum, nickel, silicon, or vanadium alloy percentages are made.



Figure 7-5. Helical torsion springs.



Figure 7-6. Belleville springs stacked on a shaft.



Figure 7-7. A torsion bar spring.

Steels used for cold rolled, flat springs vary in composition, depending on their location. They are commonly carbon and manganese alloys and may be formed from oil tempered steels (thin sections, clock type springs) or from annealed steels which are subsequently heat treated.

Nonferrous metals are also used to construct springs. Copper based alloys are used where electrical conductivity is required with an added benefit of being corrosion resistant. Nickel alloys are used where the ability to work at elevated temperatures is desired. Some of the alloys are:

1. Spring brass - which is comparatively inexpensive, has good electrical conductivity, but is unsuitable for high-stress applications.
2. Nickel silver (also called German Silver) - has better characteristics than brass. It is made from different percentages of copper, zinc and nickel.

3. Phosphor bronze - has a minimum percentage of 90% copper content. It has excellent electrical conductivity and is suitable for applications of higher stress levels than those of brass.
4. Silicon bronze - has similar characteristics to those of phosphor bronze but is less expensive to produce.
5. Beryllium copper - has similar conductivity and corrosion resistance qualities to those of copper with the addition 2 to 2.5% beryllium. This increases hardness and fortifies other mechanical properties.
6. High nickel alloys - are commonly found in aero-engine applications. Monel, 'K' Monel (3% aluminum), Permanickel, Inconel and Inconel X (2.5% titanium) are all used to make springs with high temperature and high corrosion resistant characteristics.

Composite materials are also used to make components with spring characteristics. Some composite springs involve the joining of certain metals with elastomers to form the antivibration mountings such as those found in aero-engine and auxiliary power units (APUs).

The bungee chord is also a device that acts like a spring. It is a combination of rubber strands encased within a sheath of braided material. These shock cords are used on some older light aircraft landing gear.

SPRING DIMENSIONS

The close-coiled, round-wire, helical compression spring is the type of spring most frequently encountered. It is illustrated in ***Figure 7-8***. Also illustrated in this figure are the terms used to describe a spring and to dimension it. Note that the free length of a spring includes a number of coil turns but does not include any end treatment such as an eye or hook used as an attach point. Typical coiled springs are finished on each end in a few basic ways. ***Figure 7-9*** illustrates these finishes.

SPRING CHARACTERISTICS

There are many characteristics of springs that must be considered by design engineers when specifying a spring for a particular installation. The type of spring is critical. It must be the type that is capable of absorbing the type of load present in the mechanism. The size of the spring, in material cross sectional dimension, coil diameter and coil pitch affects the load carrying characteristics of a

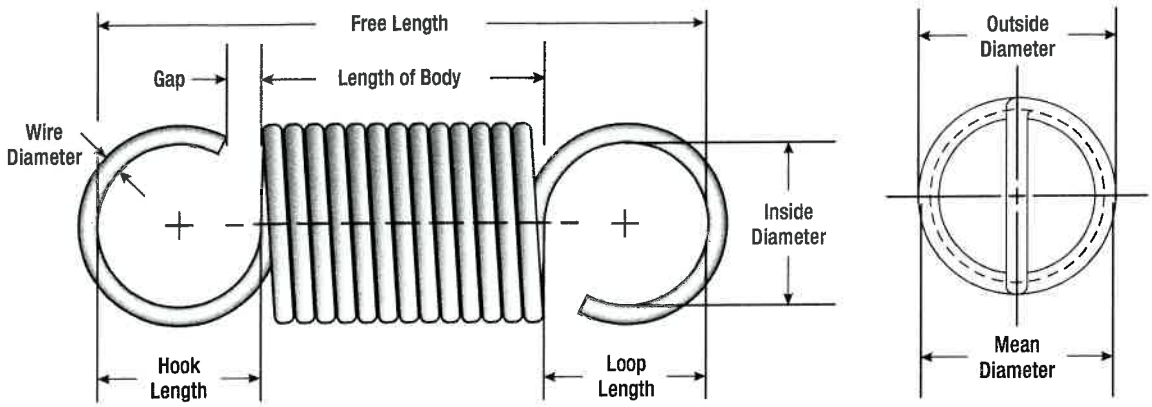


Figure 7-8. A helical spring and its various measuring points.

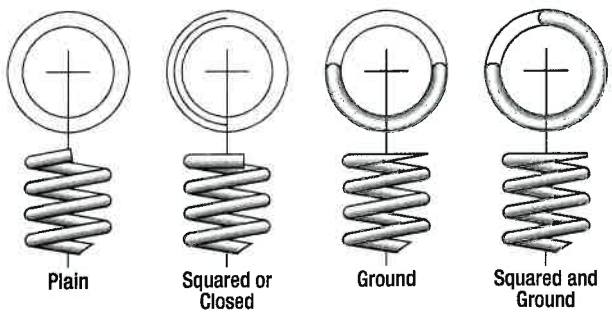


Figure 7-9. End finishes on a coiled spring.

spring as well as the ease of manufacture. The material from which the spring is made also affects the load carrying capability as do the treatments to the spring material before and after it is wound.

The environment in which the spring operates impinges on characteristics considered when selecting a spring. For example, in a corrosion prone area, a spring made of stainless steel may be employed. High temperature applications might use springs made of Inconel. When all of the factors in the operating environment of the spring are known, a spring can be constructed to service the application.

A spring is a device that stores energy. The basic ways to store energy with a spring are through compression, extension (tension) as well as torsion. The primary considerations for spring design usually are diameter, length, and the rate of energy absorption. However, listed below are the myriad of characteristics that are considered when selecting a spring for a given application.

- Working Envelope
- Material
- Elastic Modulus
- Wire Hardness
- Coiling Direction
- Stress
- Cycles
- Initial Tension
- Solid Height
- Hysteresis
- Buckling
- Magnetic Characteristics
- Corrosion
- Temperature
- Heat Treatment

When all of the various characteristics of a spring are considered and the proper spring for the application is selected or constructed, the durability and reliability of the spring is maximized.

FORCES EXERTED ON, AND APPLIED BY SPRINGS

The three basic forces, which may be exerted on, and applied by springs are:

- Compression
- Tension
- Torsion

LAW OBEYED BY SPRINGS

Springs are fundamental mechanical components which form the basis of many mechanical systems. A spring can be defined as an elastic member, which exerts a resisting force when its shape is changed. Most springs are assumed linear and obey Hooke's Law: $F = -\Delta k$.

Where:

- F = is the resisting force
- Δ = is the displacement
- k = is the spring constant which is dependent of material properties of the spring.

Springs are elastic bodies (generally metal) that can be twisted, pulled, or stretched by some force. They can return to their original shape when the force is released. Springs used in aircraft are made of metal; usually steel though some are made of phosphor bronze, brass, or other alloys. A part that is subject to constant thrust or pressure is said to be spring loaded. (Some components that appear to be spring loaded are actually under hydraulic or pneumatic pressure or are moved by weights.)

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Question: 7-1
Reciprocating engine valve springs are an example of what type of spring?

Question: 7-2
A spring is a device that _____ energy.

Question: 7-3
What are the two primary conditions you must know when selecting a springs material?

Question: 7-4
Which type of spring are all other springs a variation of?

Question: 7-5
Name four types of spring material which is suitable for high temperature applications.

Question: 7-6
What are two advantages of using composite materials for springs.

ANSWERS

Answer: 7-1
Helical compression springs.

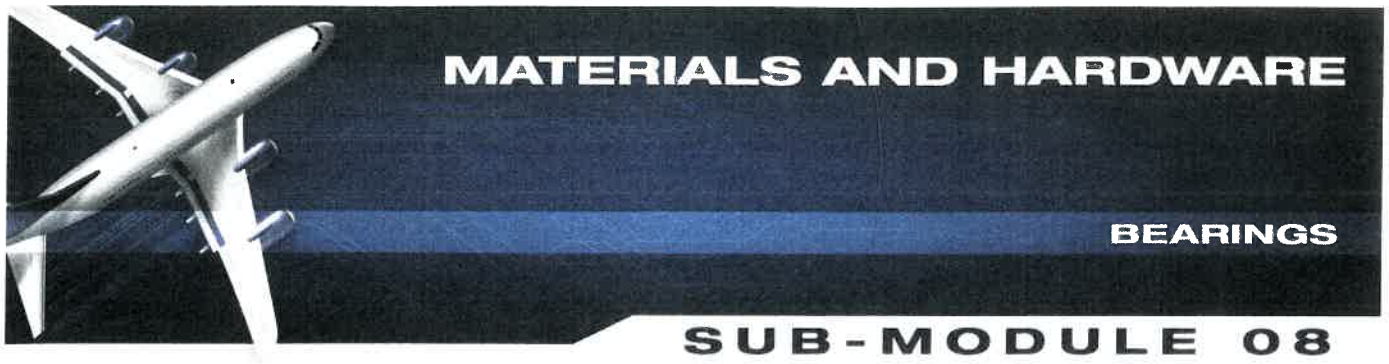
Answer: 7-2
stores.

Answer: 7-3
load, environment conditions.

Answer: 7-4
flat spring.

Answer: 7-5
Stainless Steel, Chromium Vandium Steel, Nickel, Inconel.

Answer: 7-6
Corrosion resistant; Non-conductive.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY → B1

Sub-Module 08

BEARINGS

Knowledge Requirements

6.8 - Bearings

Purpose of bearings, loads, material, construction;	2
Types of bearings and their application.	

6.8 - BEARINGS

PURPOSE OF BEARINGS

A bearing is any surface which supports, or is supported by, another surface. A good bearing must be composed of material that is strong enough to withstand the pressure imposed on it and should permit the other surface to move with a minimum of friction and wear. The parts must be held in position within very close tolerances to provide efficient and quiet operation, and yet allow freedom of motion. To accomplish this, and at the same time reduce friction of moving parts so that power loss is not excessive, lubricated bearings of many types are used. Bearings are required to take radial loads, thrust loads, or a combination of the two. An example of a radial load would be a rotating shaft being held or contained in one position on a radial plane. Thrust load would be the rotating shaft being contained from moving axially along the shafts axis. These radial and thrust loads are illustrated in ***Figure 8-1.***

There are two ways in which bearing surfaces move in relation to each other. One is by the sliding movement of one metal against the other (sliding friction), and the second is for one surface to roll over the other (rolling friction). The three different types of bearings in general use are plain, roller, and ball.

Ball and roller bearings are, broadly, classified by the type of rolling element used in their construction. Ball bearings employ steel balls, which rotate in grooved raceways, whilst Roller bearings utilize cylindrical, tapered and spherical rollers running in suitably shaped raceways. (***Figure 8-2***)

BEARING LOADS

Ball bearings and tapered roller bearings accept both radial and axial loads, whilst the other types of roller bearings may accept only radial loads. Those bearings, which are contained in cages are in general used for engine and gearbox applications with rotational speeds in excess of approximately 100 rpm. Most other bearings on an aircraft or an engine are intended for oscillating or slow rotation conditions and do not have a cage. They are generally sealed and prepacked with grease, although some have external lubrication facilities.

BEARINGS MATERIAL

Bearings are made from low carbon steel containing nickel, molybdenum and chromium. After machining, the bearing components are carburized to provide a hard, wear resistant surface. The bearing cages or separators are made from sheet steel, brass and various types of plastic. Rolling ball and roller bearings are made to high standards of accuracy. Balls and rollers are normally held to diametrical tolerances of 0.000 1 inch (0.002 54mm) or less.

CONSTRUCTION OF BALL BEARINGS

Ball bearings are manufactured with its balls in either single or double rows. They are normally, rigid for fixed applications, or self-aligning for positions where accurate alignment cannot be maintained. Bearings may also be provided with metal shields or synthetic rubber seals to prevent the ingestion of foreign debris and to retain its lubricant. The balls are often retained in a cage, but in some cases, slots in the inner and outer rings permit individual insertion of the balls, thus allowing a larger number of balls to be used and giving the bearing a greater radial load capacity. However, axial loads are limited due to the presence of the raceway interruptions.

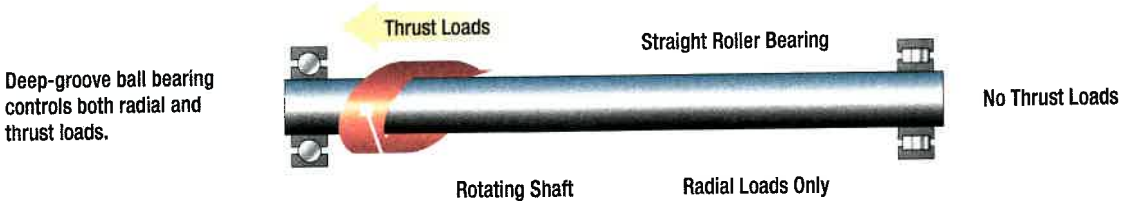


Figure 8-1. Radial and thrust loads.



Figure 8-2. Variety of plain bearings.

TYPES OF BEARINGS

PLAIN BEARINGS

Plain bearings are generally used for the crankshaft, cam ring, camshaft, connecting rods, and the accessory drive shaft bearings. Such bearings are usually subjected to radial loads only, although some have been designed to take thrust loads.

Plain bearings are usually made of nonferrous (having no iron) metals, such as silver, bronze, aluminum, and various alloys of copper, tin, or lead. Master rod or crankpin bearings in some engines are thin shells of steel, plated with silver on both the inside and the outside surfaces and with lead-tin plated over the silver on the inside surface only. Smaller bearings, such as those used to support various shafts in the accessory section, are called bushings. Porous Oilite bushings are widely used in this instance.

Plain bearings support a shaft or housing to permit their free motion about an axis of rotation. Loads can be applied to bearings in either of two basic directions. Radial loads act at right angles to the shaft's axis of rotation. Axial (thrust) bearings acts parallel to the axis of rotation.

BALL BEARINGS

A ball bearing assembly consists of grooved inner and outer races, one or more sets of balls, in bearings designed for disassembly, and a bearing retainer. They are used for shaft bearings and rocker arm bearings in some reciprocating engines. Special deep groove ball bearings are used to transmit propeller thrust and radial loads to the engine nose section of radial engines. Since this type of bearing can accept both radial and thrust loads, it is used in gas turbine engines to support one end of a shaft (radial loads) and to keep the shaft from moving axially (thrust loads). (***Figure 8-3***)

ROLLER BEARINGS

Roller bearings are made in many types and shapes, but the two types generally used in the aircraft engine are the straight roller and the tapered roller bearings. Straight roller bearings are used where the bearing is subjected to radial loads only. In tapered roller bearings, the inner and outer race bearing surfaces are cone shaped. Such bearings withstand both radial and thrust loads. Straight roller bearings are used in high power reciprocating aircraft engines for the crankshaft main bearings. They are also used in gas turbine applications where radial loads are high. Generally, a rotating shaft in a gas turbine engine is supported by a deep groove ball bearing (radial and thrust loads) on one end and a straight roller bearing (radial loads only) on the other end. (***Figure 8-4***)



Figure 8-3. Internal and external type ball bearings.



Figure 8-4. Internal and external type roller bearings.

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Question: 8-1
Bearings are required to take which two kinds of loads?

Question: 8-2
What type of bearing is used to withstand both types of loads?

Question: 8-3
What type of bearing would be found on a crankshaft or camshaft of a reciprocating engine?

Question: 8-4
What type of bearing would be found on a crankshaft or camshaft of a reciprocating engine?

Question: 8-5
Name one factor which would determine whether the balls or rollers of a bearing must be contained in a cage.

Question: 8-6
What main factor would serve to increase the strength of a ball or roller bearing?

ANSWERS

Answer: 8-1
radial loads.
thrust loads.

Answer: 8-2
Tapered roller bearing or a deep-grooved ball bearing.

Answer: 8-3
Radial.

Answer: 8-4
Plain bearings.

Answer: 8-5
The rotational speed of the bearing.

Answer: 8-6
The number of balls or rollers within the bearing.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY → B1

Sub-Module 09

TRANSMISSIONS

Knowledge Requirements

6.9 - Transmissions

Gear types and their application;	2
Gear ratios, reduction and multiplication gear systems, driven and driving gears, idler gears, mesh patterns;	
Belts and pulleys, chains and sprockets.	

GEAR TYPES AND APPLICATIONS

DRIVING AND DRIVEN GEARS

Two gears with teeth on their outer edges, as shown in ***Figure 9-1***, act like a first class lever when one gear drives the other. The gear with the input force is called the drive gear, and the other is called the driven gear. The effort arm is the diameter of the driven gear, and the resistance arm is the diameter of the drive gear.

Notice that the two gears turn in opposite directions (the bottom one clockwise and the top one counterclockwise). The gear on top (yellow) is 9 inches in diameter and has 45 teeth, and the gear on the bottom (blue) is 12 inches in diameter and has 60 teeth. Imagine that the blue gear is driving the yellow one (blue is the drive, yellow is the driven). The mechanical advantage in terms of force would be the effort arm divided by the resistance arm, or 9 ÷ 12, which is 0.75. This would actually be called a fractional disadvantage, because there would be less force out than force in. The mechanical advantage in terms of distance (rpm in this case), would be 12 ÷ 9, or 1.33.

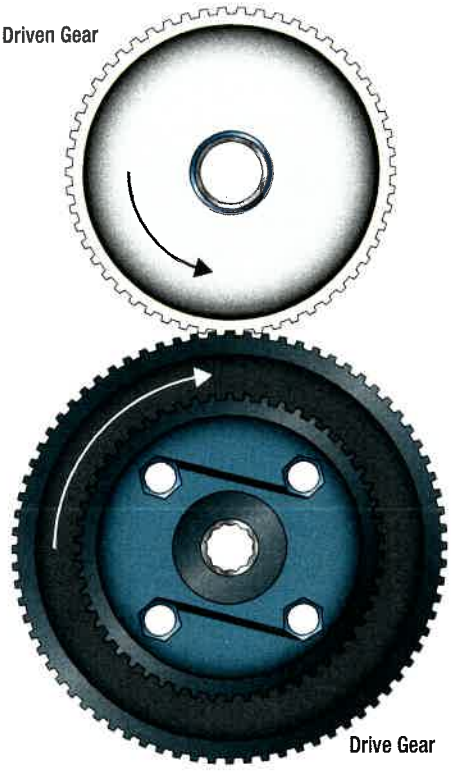


Figure 9-1. Spur gears.

This analysis tells us that when a large gear drives a small one, the small one turns faster and has less available force. In order to be a force gaining machine, the small gear needs to turn the large one. When the terminology reduction gearbox is used, such as a propeller reduction gearbox, it means that there is more rpm going in than is coming out. The end result is an increase in force, and ultimately torque.

BEVEL GEARS

Bevel gears are used to change the plane of rotation, so that a shaft turning horizontally can make a vertical shaft rotate. The size of the gears and their number of teeth determine the mechanical advantage, and whether force is being increased or rpm is being increased. If each gear has the same number of teeth, there would be no change in force or rpm. (***Figure 9-2***)

Bevel gears are an excellent choice for intersecting shaft systems. However, they are typically not used to join parallel shafts because they can become noisy at high speeds.

WORM GEARS

The worm gear has an extremely high mechanical advantage. The input force goes into the spiral worm gear, which drives the spur gear. One complete revolution of the worm gear only makes the spur gear turn an amount

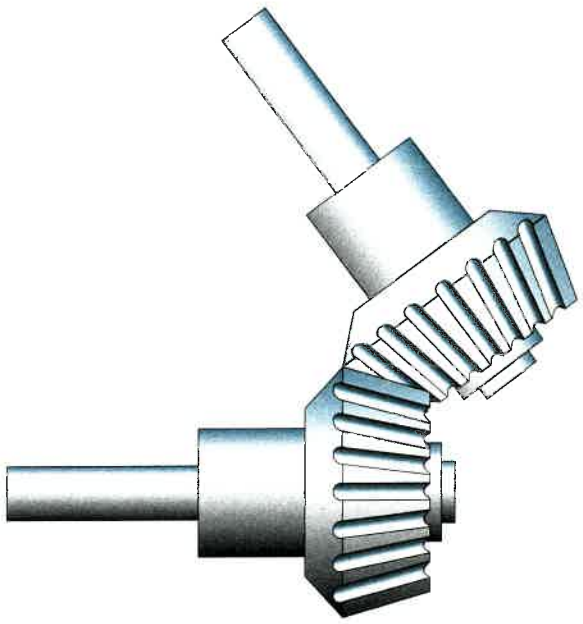


Figure 9-2. Bevel gears.

equal to one tooth. The mechanical advantage is equal to the number of teeth on the spur gear, which in this case is 25. This is a force gaining machine, to the tune of 25 times more output force. (***Figure 9-3***)

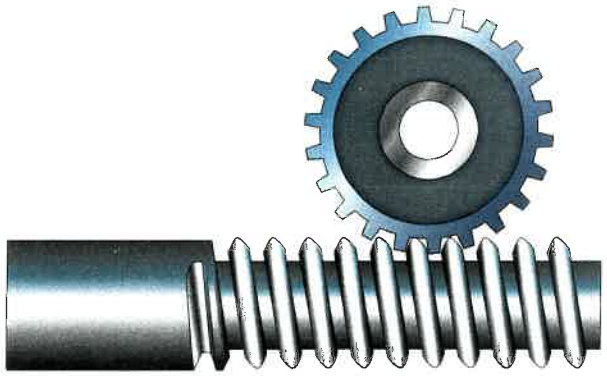


Figure 9-3. Worm gear.

HELICAL WORM GEARS

Helical gears are like worm gears except that their teeth have been twisted at an angle from the gear centerline. (***Figure 9-4***) These gears get their name from being cut in a helix, which is a form of curve. This curve is more difficult to machine, but it is used because it reduces gear noise. Contact between two teeth starts first at the leading ends and moves progressively across the gear faces until the trailing ends are in contact. This kind of meshing action keeps the gears in constant contact with one another. Therefore, less lost motion and smoother, quieter action is possible. Because this angular contact tends to cause side thrusts which a bearing must absorb, it is necessary to put a special thrust bearing at the end of the shaft to counteract this thrust. However, helical worm gears are quieter in operation and have greater strength and durability than straight worm gears, simply because the contacting teeth are longer.



Figure 9-4. An example of a Helical Worm Gear.

Helical gears can be used on non-parallel and even perpendicular shafts, and can carry higher loads than worm gears, because the length of the teeth is more. However, helical gears have the disadvantage of being expensive. Helical gears are also slightly less efficient than a worm gear of the same size as they create an additional thrust loads during operation.

PLANETARY GEARS

The planetary sun gear system is typical of what would be found in a propeller reduction gearbox. The power output shaft of the engine would drive the sun gear in the middle, which rotates the planetary gears and ultimately the ring gear. In this example, the sun gear has 28 teeth, each planet gear has 22 teeth, and the ring gear has 82 teeth. To figure how much gear reduction is taking place, the number of teeth on the ring gear is divided by the number of teeth on the sun gear. In this case, the gear reduction is 2.93, meaning the engine has an rpm 2.93 times greater than the propeller. (***Figure 9-5***)

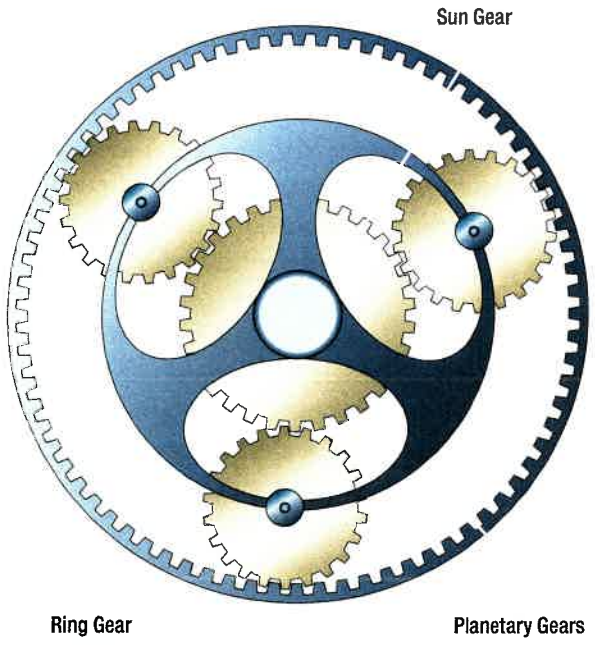


Figure 9-5. Planetary sun gear.

RACK AND PINION GEARS

The rack and pinion gear (***Figure 9-6***) is a type of spur gear. The rack is a piece cut from a gear with an extremely large radius. The rack and pinion arrangement is useful in changing rotary motion into linear motion or the reverse. While this arrangement has limited usefulness, its advantage within this purpose is greater better precision than most other options.



Figure 9-6. Rack and pinion.

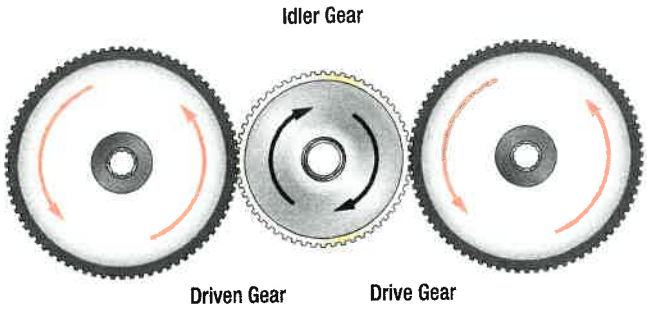


Figure 9-7. Idler gear.

GEAR SYSTEMS

GEAR TERMS

Backlash (or lash)
The terms used to describe the clearance which must exist between gear teeth at point of mesh, essential with all forms of gearing to allow for expansion and lubrication.

Idler Gear
A gear which is interposed between the driving and driven gear, its function is to connect the drive between two shafts. A spur idler gear is used between two parallel shafts to maintain the direction of rotation and does not affect the ratio of the gears. A bevel idler may be used where two shafts intersect and/or are co-axial. (***Figure 9-7***)

Intermediate Gear
A gear which is positioned between the driving gear and one or more driven gears in a gear train. It may function as an idler gear or transmit drive through its own shaft.

Compound Gear
This is a gear wheel which has more than one driving face. These faces may be formed integrally on one casting or forging, or it may comprise two or more gears bolted or splined together to transmit drive to a number of shafts.

Pinion Gear
The pinion is a round gear, usually the smaller of two meshed gears, used in several applications.

Lay-shaft
A shaft which supports an idler gear or intermediate gear, it may be integral with the gear and be supported by bearings, or may be fixed and provide a bearing surface for the rotating gear.

Step-Up Drive
A drive through a gear train in which the speed of rotation of the output (driven) shaft is increased. Example: Used in aero-engines in a generator drive. It ensures that the generator has sufficient rev/mm to remain "on charge" at engine idling rev/mm.

Step-Down Drive
A reduction gear in which the rev/mm of the output shaft is reduced while the torque is increased. Example: Used between the engine and propeller in order to allow the engine to develop its power by running at high RPM while maintaining high propeller efficiency by avoiding the tips speeds reaching Mach 1.

GEAR RATIO

Gear ratio is the number of teeth each gear represents when two gears are used in an aircraft component. In ***Figure 9-8***, the pinion gear has 8 teeth and the spur gear has 28 teeth. The gear ration is 8:28 or 2:7.

When two gears are used in an aircraft component, the rotational speed of each gear is represented as a speed ratio. As the number of teeth in a gear decreases, the rotational speed of that gear increases, and vice-versa. Therefore, the speed ratio of two gears is the inverse (or opposite) of the gear ratio. If two gears have a gear ratio of 2:9, then their speed ratio is 9:2.

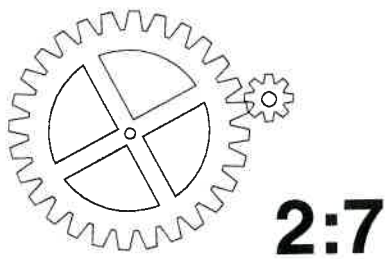


Figure 9-8. Gear ratio.

Example: A pinion gear with 10 teeth is driving a spur gear with 40 teeth. The spur gear is rotating at 160 revolutions per minute (rpm). Determine the speed of the pinion gear.

$$\frac{\text{Teeth in Pinion Gear}}{\text{Teeth in Spur Gear}} = \frac{\text{Speed of Spur Gear}}{\text{Speed of Pinion Gear}}$$

$$\frac{10 \text{ teeth}}{40 \text{ teeth}} = \frac{160 \text{ rpm}}{S_P \text{ (speed of pinion gear)}}$$

To solve for S_P , multiply 40 × 160, then divide by 10. The speed of the pinion gear is 640 rpm.

Example: If the cruising speed of an airplane is 400 km/h and its maximum speed is 500 km/h, what is the ratio of cruising speed to maximum speed? First, express the cruising speed as the numerator of a fraction whose denominator is the maximum speed.

Next, reduce the resulting fraction to its lowest terms.

$$Ratio = \frac{400}{500}$$

Therefore, the ratio of cruising speed to maximum speed is 4:5.

$$Ratio = \frac{400}{500} = \frac{4}{5}$$

Another common use of ratios is to convert any given ratio to an equivalent ratio with a denominator of 1.
Example: Express the ratio 9:5 as a ratio with a denominator of 1.

$$R = \frac{9}{5} = \frac{?}{1} \quad \textit{Since } 9 \div 5 = 1.8, \textit{ then } \frac{9}{5} = \frac{1.8}{1}$$

Therefore, 9:5 is the same ratio as 1.8:1. In other words, 9 to 5 is the same ratio as 1.8 to 1.

PROPELLER REDUCTION GEARING

The increased brake horsepower delivered by a high horsepower engine results partly from increased crankshaft rpm. It is therefore necessary to provide reduction gears to limit the propeller rotation speed to a value at which efficient operation is obtained. Whenever the speed of the blade tips approaches the speed of sound, the efficiency of the propeller decreases rapidly. Reduction gearing for engines allows the engine to operate at a higher rpm, developing more power while slowing down the propeller rpm. This prevents the propeller efficiency from decreasing. Since reduction gearing must withstand extremely high stresses, the gears are machined from steel forgings. Many types of reduction gearing systems are in use. The three types most commonly used are spur planetary, bevel planetary, and spur and pinion. (***Figure 9-9***)

The spur planetary reduction gearing consists of a large driving gear or sun gear splined (and sometimes shrunk) to the crankshaft, a large stationary gear, called a bell gear, and a set of small spur planetary pinion gears mounted on a carrier ring. The ring is fastened to the propeller shaft and the planetary gears mesh with both the sun gear and the stationary bell or ring gear. The stationary gear is bolted or splined to the front section housing. When the engine is operating, the sun gear rotates. Because the planetary gears are meshed with this ring, they also must rotate. Since they also mesh with the stationary gear, they walk or roll around it as they rotate, and the ring in which they are mounted rotates the propeller shaft in the same direction as the crankshaft but at a reduced speed.

In some engines, the bell gear is mounted on the propeller shaft, and the planetary pinion gear cage is held stationary. The sun gear is splined to the crankshaft and acts as a driving gear. In such an arrangement, the propeller travels at a reduced speed but in opposite direction to the crankshaft.

In the bevel planetary reduction gearing system, the driving gear is machined with beveled external teeth and is attached to the crankshaft. A set of mating bevel pinion gears is mounted in a cage attached to the end of the propeller shaft. The pinion gears are driven by the drive gear and walk around the stationary gear, which is bolted or splined to the front section housing. The thrust of the bevel pinion gears is absorbed by a thrust ball bearing of special design. The drive and the fixed gears are generally supported by heavy duty ball bearings. This type of planetary reduction assembly is more compact than the other one described and, therefore, can be used where a smaller propeller gear step-down is desired. In the case of gas turbine turboprop engines, more than one stage of reduction gearing is used do to the high

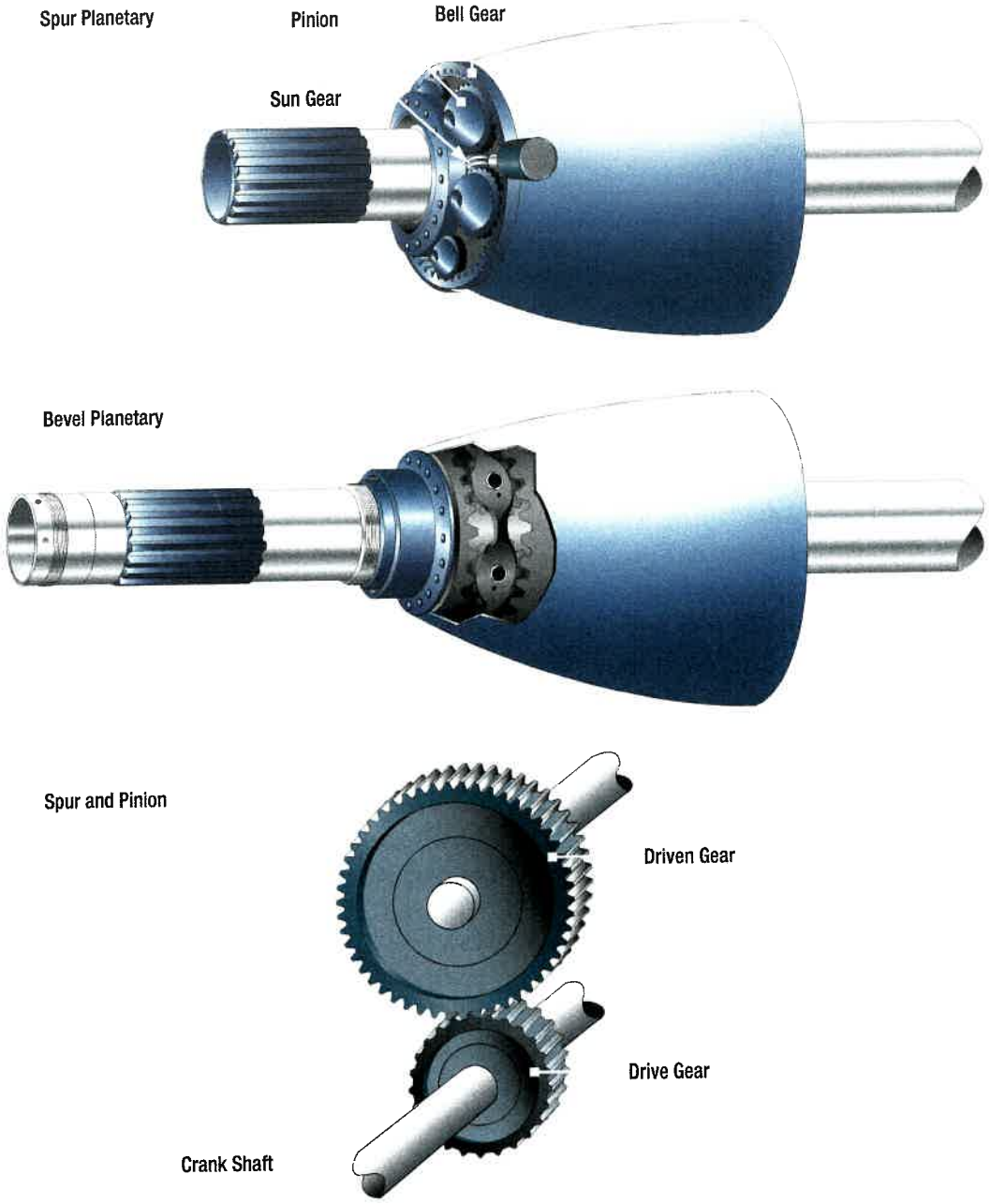


Figure 9-9. Reduction gears.

output speeds of the engine. Several types of lower powered engines can use the spur and pinion reduction gear arrangement.

GEAR LASH AND PATTERN

The lash and pattern of a gear is determined by how the teeth of one gear mate with the teeth of another gear. If the teeth of one gear are set too tightly into the teeth of another, there will be no lash and the gears will not be properly lubricated because a film of oil must be present between the teeth of the gears as they mesh. If the gears are meshed too high in relation to the teeth, the load will be transmitted to the smallest portion of the tooth, breaking the teeth because of the load area. The ideal placement of the teeth is in the middle area. (***Figure 9-10***)

The pattern of two gears is also very important and closely related to the lash. The lashes and patterns are generally most critical where directional changes are made with gears, because they are adjustable by moving the gears inward or outward in order to obtain the correct relationship. (***Figure 9-11***)

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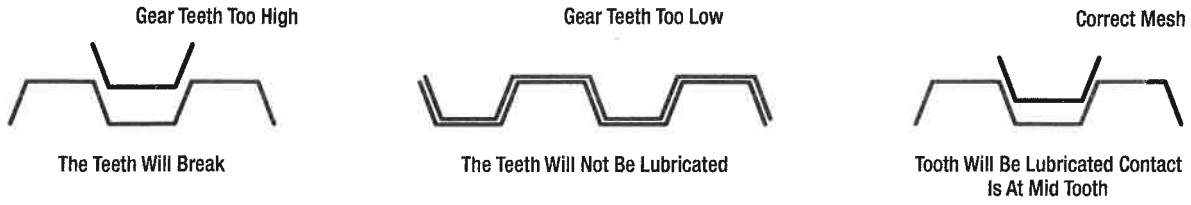


Figure 9-10. Various gear tooth positions with mating gears.

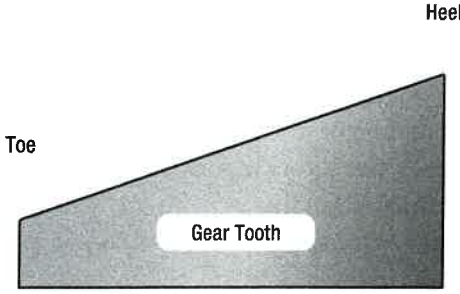


Figure 9-11. Gear tooth nomenclature used with gear patterns.

The pattern is simply the print that one gear leaves on another gear with which it mates. Each gear tooth of a bevelled gear has a heel and a toe, The heel is located at the largest diameter of the bevel gear, and the toe is located at the smallest diameter of the gear. If the gears mate in such a manner that the teeth ride too close to the heel or toe, the load will not be distributed evenly and breakage of the teeth will occur. The diagram below shows the pattern set too close to the toe and to the heel. If it is a straight tooth bevel gear, the proper pattern should be at the middle of the tooth. However, most gears are not straight tooth gears, but are helical in design. By being helical more strength may be developed in a smaller gear. With this type of gear a different pattern is developed in a no-load condition than under a load. With this type of gear the ideal no-load pattern is shown in ***Figure 9-12***.

The patterns of these gears is taken by using Prussian Blue on the gears to leave the imprint of the other gear on the teeth. The gears are moved in and out by shims. The thickness of these shims determine the position of the gear and change the lash and pattern.

Often these gears on which lashes and patterns must be obtained are matched sets and are replaced as matched sets. Some of the older gears of this type even have match marks as to what tooth meshes with what tooth with X's and O's scribed in the tooth. This is so that the high and low spots do not come in contact with each other at the same time. Today this system is almost forgotten because more modern methods of manufacturing have eliminated the problem. (***Figure 9-13***)

Opinions vary considerably as to whether lash or pattern should be determined first. Most technicians prefer to obtain lash first and then check the pattern. The reverse of finding pattern and then determining lash will also work. If either lash or pattern cannot be properly adjusted, it is a good indication of excessive wear in the gear teeth and the gears must be replaced. In such situations it is most probable that if pattern is obtained, the lash will be out. (***Figure 9-14***)

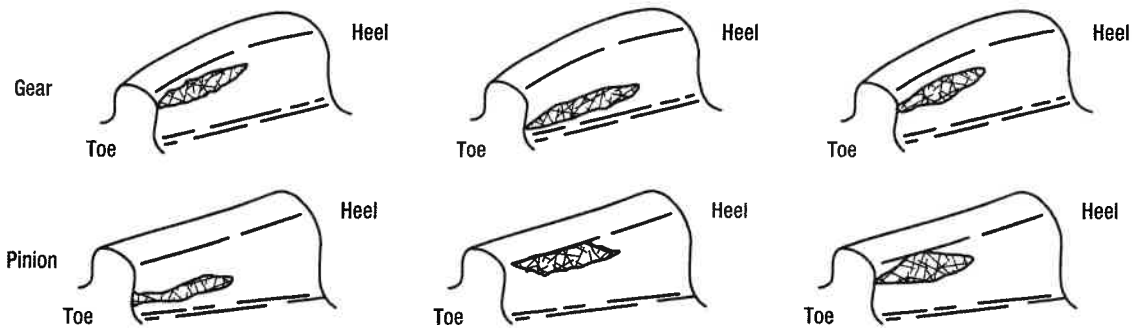


Figure 9-12. Ideal no-load pattern.

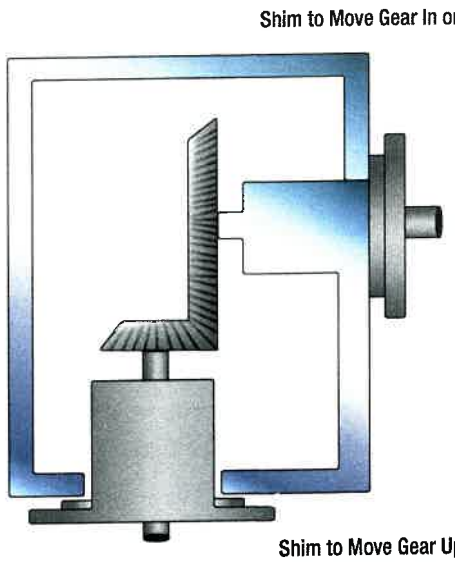


Figure 9-13. Typical shim method of moving gear positions.

NON-GEAR TRANSMISSIONS

BELTS AND PULLEYS

A belt and pulley system is composed of two or more pulleys joined by a belt. This allows for mechanical power, torque, and speed to be transmitted across axles. If the pulleys are of different diameters, a mechanical advantage is realized. (***Figure 9-15***)

One application of belt drives is on certain piston powered helicopters which use a belt to connect the output pulley on the end of the crankshaft to the transmission and rotor. Tension pulleys, which bear onto the belt, keep it at the correct tension for normal use. When starting up, the tension can be totally released, allowing the engine to be started without the load of the rotors and transmission. In an emergency the released

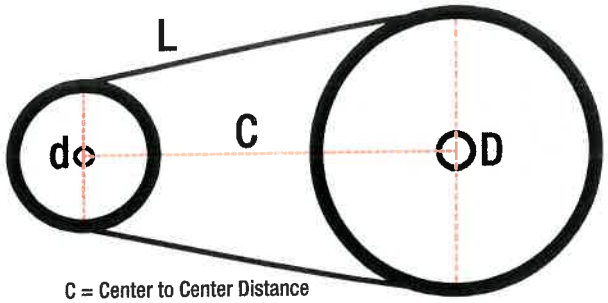


Figure 9-15. Belts and pulleys.

tension allows the rotors to free wheel (autorotate) and thus enables a safe landing.

Types of Belt Drives

In a two pulley system, depending on the direction the belt drives the pulley, belt drives are divided into two types; open belt drive and crossed belt drive.

Open Belt Drives

An open belt drive is used to rotate the driven pulley in the same direction as the driving pulley. In the motion of belt drive, power transmission results make one side of pulley more tightened compared to the other side. In horizontal drives, the tightened side is always kept on the lower side of pulleys because the sag of the upper side slightly increases the angle of wrapping of the belt on the two pulleys. (***Figure 9-16***)

Crossed Belt Drives

A crossed belt drive is used to rotate the driven pulley in the opposite direction of the driving pulley. The higher amount of wrap enables that more power can

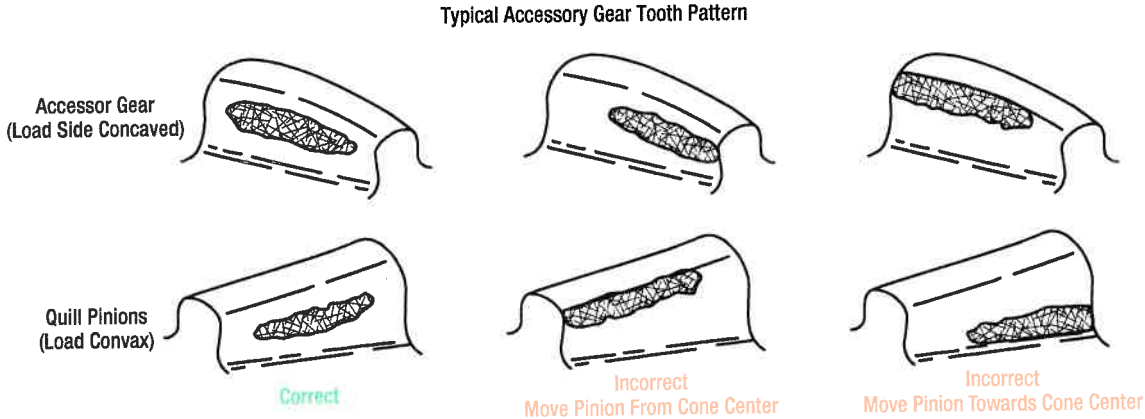
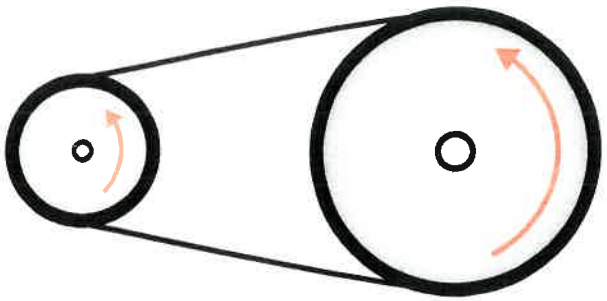
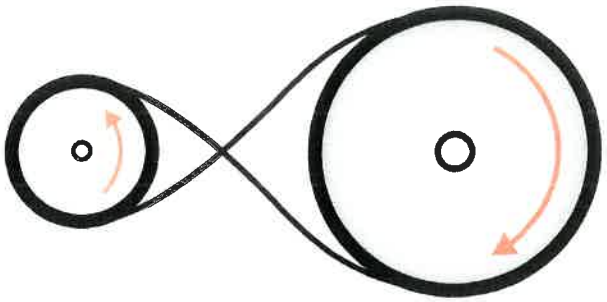


Figure 9-14. Patterns may be changed by moving the gears.



Open Belt Drive

Figure 9-16. Open belt drives.



Crossed Belt Drive

Figure 9-17. Crossed belt drives.

be transmitted than an open belt drive. However, bending and wear of the belt are important concerns. (***Figure 9-17***)

Advantages and Disadvantages of Belt Drives

Some advantages of belt drives are:

- They are simple are economical.
- They don't need parallel shafts.
- Slippage in the system can provide overload and jam protection.
- Noise and vibration are damped out. Machinery life is increased because load fluctuations are shock absorbed.
- They are lubrication free.
- Belt drives are highly efficient in use (typically 95%).
- They are very economical when the distance between shafts is very large.

Some disadvantages of belt drives are:

- Angular velocity ratio is not always constant to the ratio of pulley diameters, because of slipping and stretching.
- Heat buildup occurs. Speed is limited to usually 35 meters per second. Power transmission is limited to 370 kilowatts.
- Operating temperatures are usually restricted to –35 to 85°C.

CHAINS AND SPROCKET

A sprocket is a toothed wheel that fits onto a shaft and prevented from rotating on the shaft by a key that fits into keyways in the sprocket and shaft.

A chain is used to connect sprockets. One sprocket is the driver sprocket. The other is the driven sprocket. Motion and force can be transmitted via the chain from one sprocket to another, and so from one shaft to another. (***Figure 9-18***)

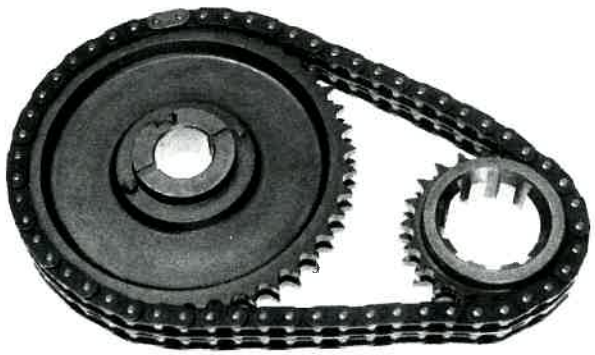


Figure 9-18. Chain and sprocket.

There are 5 types of power transmission chains:

- General purpose roller chains, widely used in industry.
- High performance roller chains which are stronger than general purpose chains.
- Lube-free chains, no lubrication required.
- Environmentally resistant chains with special corrosion resistance.
- Specialty chains used on bicycle and motorcycles.

Most chains are the roller type which are composed of link plates and pins that join the link plates, rollers, and bushes together. (***Figure 9-19***)

Advantages and Disadvantages of Chain Drives

An advantage of chain drives over most belt drives is that the chain cannot slip on the sprocket so the chain and sprocket provides a positive, non-slip drive. Chain drives can also be used over fairly long distances between the power source and required application. Some belt and pulley drives also have teeth. These toothed belt and pulley drives are used when it is important that the belt does not slide on the pulley, such as timing belts.

The disadvantage of the chain and sprocket is that it can be noisy and more expensive than a belt and pulley drive system.

Chain and sprocket drives are not used on applications such as drilling machines and lathes precisely because it does not allow slip. For safety reasons, in the event of something jamming, under great pressure, the belt can slip on the pulley rather than damaging the machine as would happen with a no-slip chain drive.

Leaf Chains

Leaf chains are a simple form of steel chain, consisting only of link plates and pins. Leaf chains are very strong, having greater tensile strength than roller chains. However because there are no rollers or bushes in leaf chains, they must be lubricated regularly and used for low speed applications such as raising and lowering the forks of a fork lift truck.

Leaf chains run over sheaves, (which are similar to pulleys) rather than over sprockets. The chain is fitted with a clevis at each end which is held in place by a clevis pin. (***Figure 9-20***)

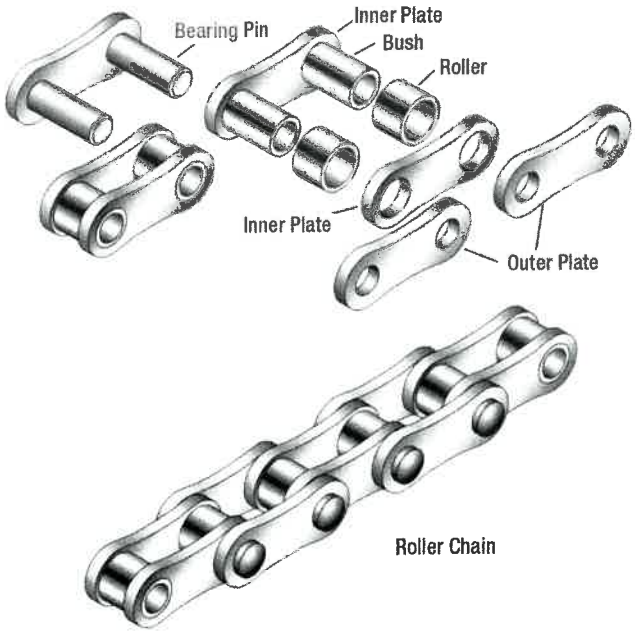


Figure 9-19. Roller chains.

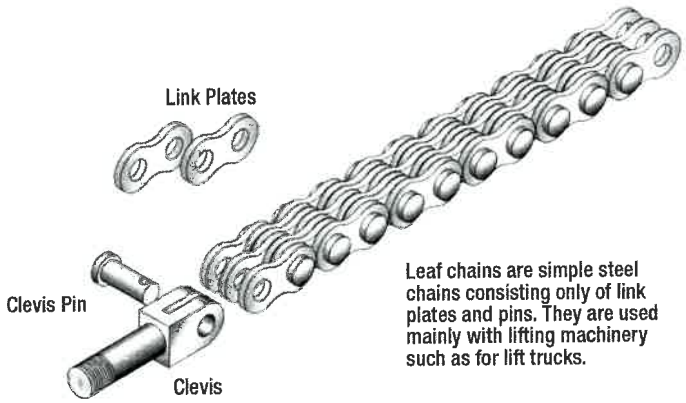


Figure 9-20. Leaf chain.

Question: 9-1
A driven gear and a drive gear are examples of a first class _____.

Question: 9-2
_____ gears are used to change the plane of rotation.

Question: 9-3
The _____ of two gears is the inverse (or opposite) of the gear ratio.

Question: 9-4
A high horsepower engine uses _____ gears to deliver power to the propeller so that it turns at an efficient speed.

Question: 9-5
Gear _____ is determined by using die on one set of gears and observing the location of the die transferred to the mated gear teeth.

Question: 9-6
Flat belts and pulleys use only _____ to transmit power.

ANSWERS

Answer: 9-1
lever.

Answer: 9-2
Bevel.

Answer: 9-3
speed.

Answer: 9-5
reduction.

Answer: 9-6
pattern.

Answer: 9-7
friction.



PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY → B1

Sub-Module 10

CONTROL CABLES

Knowledge Requirements

6.10 - Control Cables

Types of cables;	2
End fittings, turnbuckles and compensation devices;	
Pulleys and cable system components;	
Bowden cables;	
Aircraft flexible control systems.	

6.10 - CONTROL CABLES

CABLE CONTROL SYSTEMS

Flexible control cables are common, especially in smaller aircraft to actuate flight and engine controls from the cockpit. Very little play is present in controls and no lost motion exists between the actuator and unit. Consequently, cable controlled units respond quickly and accurately to cockpit control movement.

Cable linkage has several advantages. It's strong and light weight, and its flexibility makes it easy to route through the aircraft. An aircraft cable has high mechanical efficiency and can be set up without backlash, which is very important for precise control. A disadvantage is that cable tension must be adjusted frequently due to stretching and temperature changes. Turnbuckles and other means are often used to adjust tension.

Cables will not sever easily under sudden strains. They are stronger than rods or tubing of the same size. They flex without becoming deformed and can be led easily around obstacles by using pulleys. Cables can be installed over long distances without a great degree of sagging. Vibration will not cause them to harden, crystallize, or break, as may be the case with push-pull control rods. Because of the number of wires used in cables, cable failure is never abrupt, but is progressive during extended use.

CABLE CONSTRUCTION

The basic element of a cable is strand. Numbers of strands are braided together to form a wire and number of wires are braided together to form a cable.

The direction of the twist of strands and wires could be either in the same or opposite directions. When opposite, the tendency of the wires and strands to uncoil is reduced when the cable is twisted in one direction. This opposite direction twist which is most common is called a regular lay. Cables may have a right or left regular lay. If the strands are twisted around the center core, the lay is called a Lang lay. Most aircraft cables have a right regular lay.

When aircraft cables are manufactured, each strand is first formed to the spiral shape to fit the position it is to occupy in the finished cable. This process is called preforming. It is adopted to ensure flexibility in the finished cable and to relieve twisting stresses as strands they are woven into the cable. It also keeps the strands from spreading when the cable is cut. All aircraft cables are internally lubricated during construction.

TYPES OF CONTROL CABLES

Cables are designated according to the number of wires in a cable and the number of strands in each wire. The most common aircraft cables are 7 × 7 and 7 × 19. A 7 × 7 cable consists of 7 strands of 7 wires each. Six of these strands lay around a center strand. This is a medium flexibility cable and is used for trim tab, indicator, and engine controls. The 7 × 19 cable is made of 7 strands of 19 cables. This cable is extra flexible and is used in places where operation over pulleys is frequent. Aircraft control cables vary in diameter from 1/16 to 3/8 inch. (***Figure 10-1***)

Non-flexible Cables

In areas which do not pass over pulleys, non-flexible cable can be used. Non-flexible cable is available in both galvanized carbon steel and stainless steel.

Flexible Cables

Flexible steel cable made up of seven strands of seven wires each is called 7 × 7 or flexible cable and is available in 1/16 and 3/32 inch sizes in both galvanized carbon steel and stainless steel. Both types are preformed.

Extra Flexible Cable

The most widely used cable is 7 × 19 available in sizes from 1/8 inch and up. These cables are available in galvanized or stainless steel. Galvanized cable is more resistant to fatigue than stainless steel, but in applications where corrosion is a factor, stainless steel is used.

CABLE FITTINGS AND COMPENSATION DEVICES

Cables may be equipped with several types of fittings, such as terminals, thimbles, bushings, and shackles.

CABLE END FITTINGS

Cable ends may be equipped with several different types of fittings such as terminals, thimbles, bushings, and shackles. Terminal fittings are generally of the swaged type. Terminal fittings are available with threaded ends, fork ends, eye ends, and single-shank and double-shank ball ends. Threaded-end, fork-end, and eye-end terminals

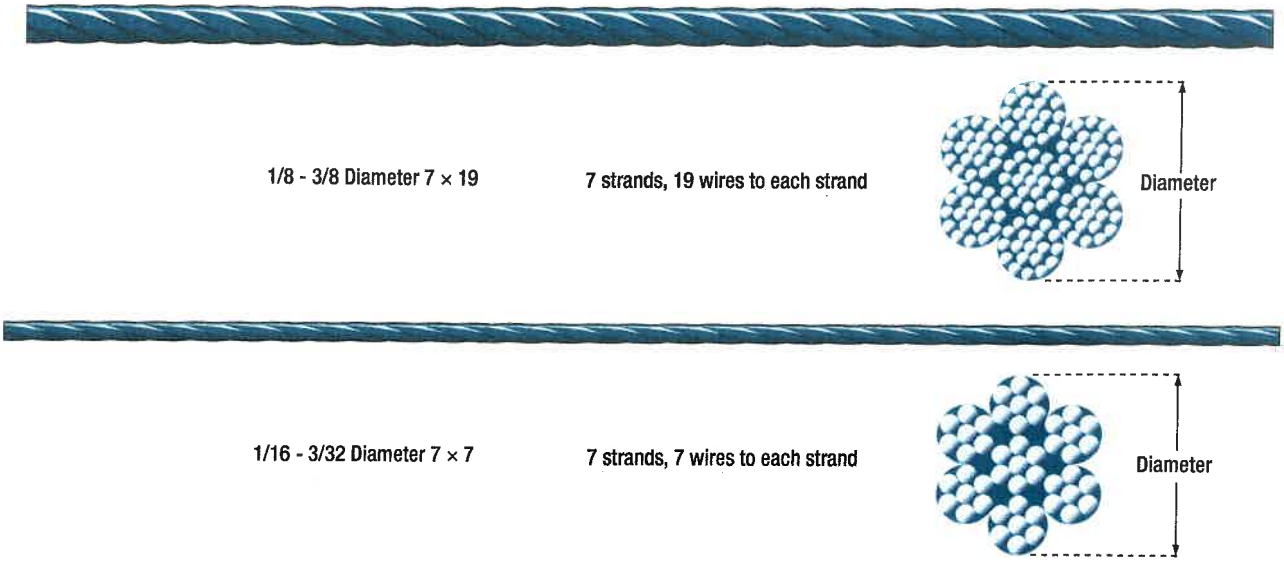


Figure 10-1. Cable cross sections.

are used to connect the cable to turnbuckles, bell cranks, and other linkage in the system. The ball terminals are used for attaching cable to quadrants and special connections where space is limited. The single-shank ball end is usually used on the ends of cables, and the double-shank ball end may be used at either the ends or in the center of a cable run. Thimble and bushing fittings may be used in place of some types of terminal fittings when facilities and supplies are limited and immediate replacement of the cable is necessary. (***Figure 10-2***).

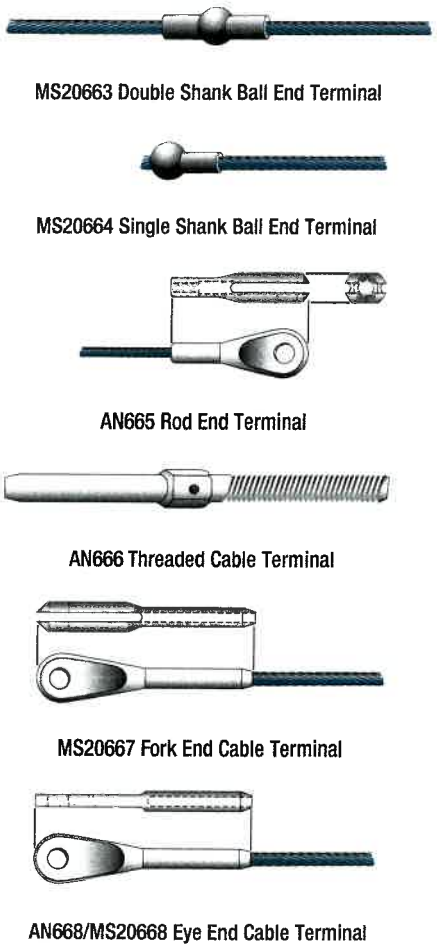


Figure 10-2. Types of terminal fittings.

CABLE TENSION

One disadvantage of a cable system relates to thermal contraction. As an aircraft climbs to a high altitude; its temperature drops and its structure contracts at a higher rate than the cable itself. As a result, cables lose their tension. Large aircraft have a rather complex automatic tensioning system to keep tension relatively constant as this occurs. Small aircraft without these automated systems must have their cable tension adjusted on the ground so they are not too tight when the airplane heats or cools.

Tensiometers

For the aircraft to operate as it was designed, the cable tension for the flight controls must be correct. To determine the amount of tension on a cable, a tensiometer is used. When properly maintained, a tensiometer is 98 percent accurate. Cable tension is determined by measuring the amount of force needed to make an offset in the cable between two hardened steel blocks called anvils. A riser or plunger is pressed against the cable to form the offset. Several manufacturers make a variety of tensiometers, each type designed for different kinds of cable, cable sizes, and cable tensions. (***Figure 10-3***)

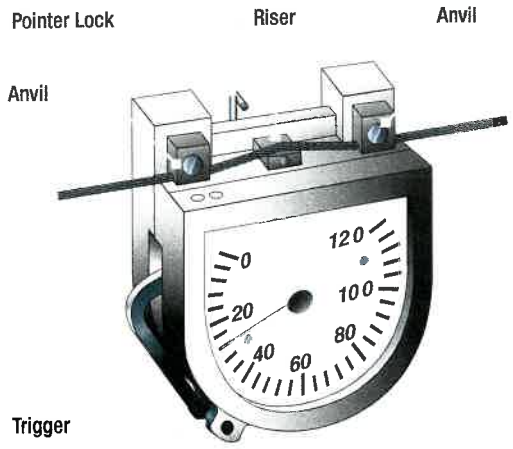


Figure 10-3. Tensiometer.

Tension Regulators

Cable tension regulators are used in some flight control systems because there is considerable difference temperature expansion of the aluminum aircraft structure and the steel control cables. Some large aircraft incorporate tension regulators in the control cable systems to maintain a given cable tension automatically. The unit consists of a compression spring and a locking mechanism that allows the spring to make correction in the system only when the cable system is in neutral.

TURNBUCKLES

A turnbuckle assembly is a mechanical screw device consisting of two threaded terminals and a threaded barrel. Turnbuckles are fitted in cable assembly for the purpose of making minor adjustments in cable length and for adjusting cable tension. One of the terminals has right-hand threads, and the other has left-hand threads. The barrel has matching right- and left-hand internal threads. The end of the barrel with the left-hand threads can usually be identified by a groove or knurl around that end of the barrel. (***Figure 10-4***)

When installing a turnbuckle in a control system, it is necessary to screw both of the terminals an equal number of turns into the barrel. It is also essential that all turnbuckle terminals be screwed into the barrel until not more than three threads are exposed on either side of the turnbuckle barrel. After a turnbuckle is properly adjusted, it must be safetied. There are a number of methods to safety a turnbuckle and/or other types of swaged cable ends that are satisfactory. A double-wrap safety wire method is preferred. Some turnbuckles are manufactured and designed to accommodate special locking devices. A typical unit is shown in ***Figure 10-5.***

Turnbuckles are commonly used in applications which require a great deal of tension; This can range in mass from about ten grams for thin cable used in light commercial applications to thousands of kilograms for structural elements in buildings and suspension bridges.

PULLEYS AND CABLE SYSTEM COMPONENTS

PULLEYS

Pulleys are used to guide cables and also to change the direction of cable movement. Pulley bearings are sealed and need no lubrication other than the lubrication done at the factory. Brackets fastened to the structure of the aircraft support the pulleys. Cables passing over pulleys are kept in place by guards. The guards are close fitting to prevent jamming or to prevent the cables from slipping off when they slacken due to temperature variations. Pulleys should be examined to ensure proper lubrication; smooth rotation and freedom from abnormal cable wear patterns which can provide an indication of other problems in the cable system. (***Figure 10-6***)

FAIRLEADS

Fairleads may be made from a nonmetallic material, such as phenolic, or a metallic material, such as soft aluminum. The fairlead completely encircles the cable where it passes through holes in bulkheads or other metal parts. Fairleads are used to guide cables in a straight line through or between structural members of the aircraft. Fairleads should never deflect the alignment of a cable more than 3° from a straight line.

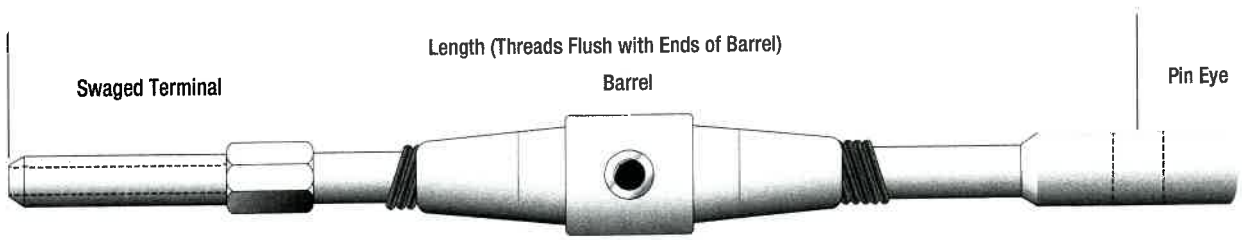


Figure 10-4. Typical turnbuckle assembly.

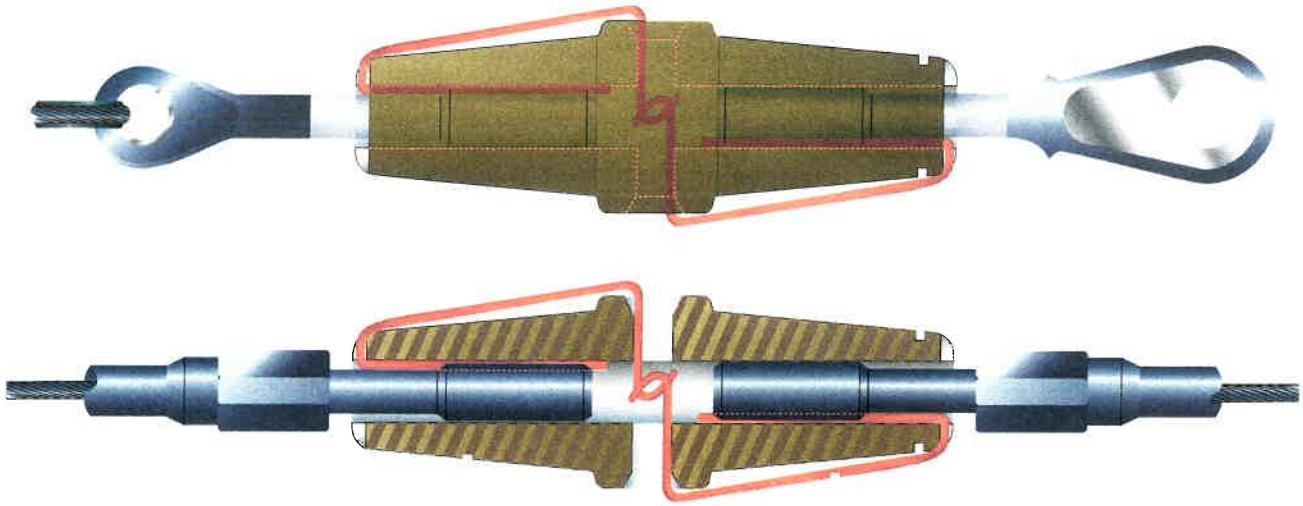


Figure 10-5. Typical turnbuckle assembly with locking mechanism.

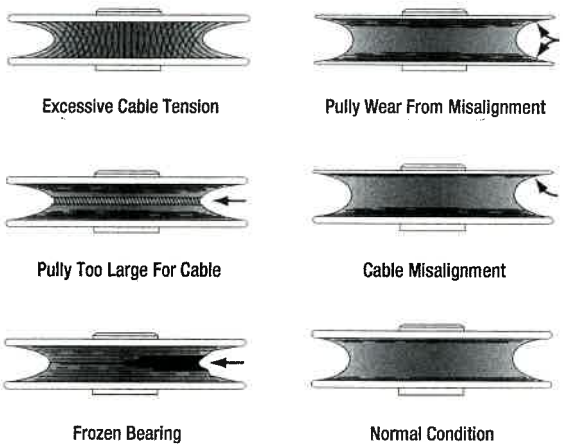


Figure 10-6. Pulley wear patterns.

PRESSURE SEALS

Pressure seals are installed where cables (or rods) move through pressure bulkheads. The seal grips tightly enough to prevent excess air pressure loss but not enough to hinder movement of the cable. Pressure seals should be inspected at regular intervals to determine that the retaining rings are in place. If a retaining ring comes off, it may slide along the cable and cause jamming of a pulley. (***Figure 10-7***)

CABLE CONNECTORS

In addition to turnbuckles, cable connectors are used in some systems. These connectors enable a cable length to be quickly connected or disconnected from a system. ***Figure 10-8*** illustrates one type of cable connector.

CONTROL SURFACE TRAVEL ADJUSTMENT

Control surfaces should move a certain distance in either direction from the neutral position. These movements must be synchronized with the movement of the flight deck controls. The flight control system must be adjusted (rigged) to obtain these requirements. The tools for measuring surface travel primarily include protractors, rigging fixtures, contour templates, and rulers. These tools are used when rigging flight control systems to assure that the desired travel has been obtained.

Generally speaking, the rigging consists of the following:

1. Positioning the flight control system in neutral and temporarily locking it there with rig pins or blocks.
2. Adjusting system cable tension and maintaining rudder, elevator, and ailerons in the neutral position.
3. Adjusting the control stops to the aircraft manufacturer's specifications.

RIGGING FIXTURES

Rigging fixtures and templates are special tools (gauges) designed by the manufacturer to measure control surface travel. Markings on the fixture or template indicate desired control surface travel.

SPRING-BACK

With a control cable properly rigged, the flight control should hit its stops at both extremes prior to the flight deck control. The spring-back is the small extra push that is needed for the flight deck control to hit its mechanical stop.

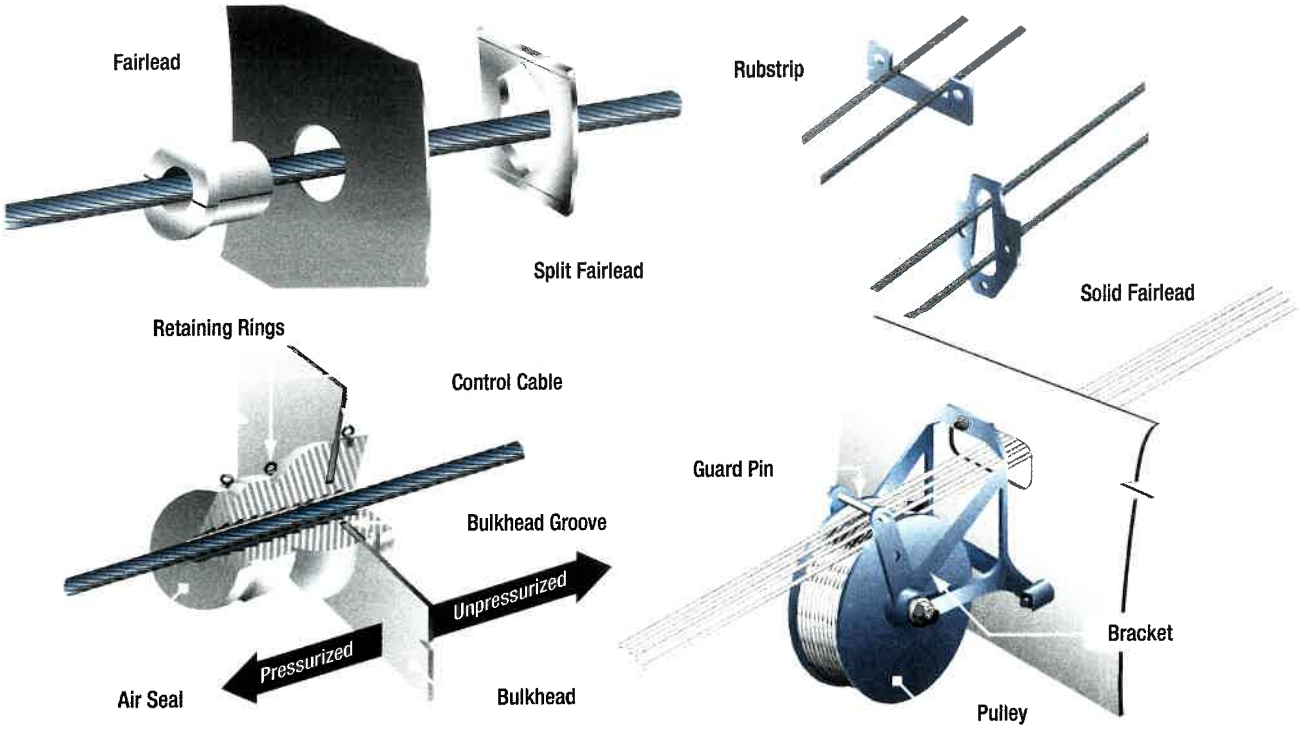


Figure 10-7. Cable guides.



Figure 10-8. Spring type connector.

CABLE DRUMS

Cable drums are used primarily in trim tab systems. As the trim tab control wheel is moved clockwise or counterclockwise, the cable drum winds or unwinds to actuate the trim tab cables. (***Figure 10-9***)

FLEXIBLE CONTROL SYSTEMS

Normal aircraft cables are only capable of performing a pulling action, due to their lack of rigidity, so, where a two-directional movement (push/pull) is required it would be necessary either to employ the use of rods, with the attendant weight penalty, or to use flexible control systems. The two most common are:

1. Bowden cables.
2. Teleflex® control systems.

BOWDEN CABLES

A Bowden cable is an inner steel cable which is free to slide inside an outer housing. The housing is typically heavy gauge helical steel wire. The inner cable moves linearly to transmit force while the housing is typically fixed at both ends. (***Figure 10-10***)

Many light aircraft use a push/pull Bowden cables for the throttle or carburetor mixture control. Usually, provision is made for adjusting the cable tension using an inline hollow bolt (often called a "barrel adjuster"), which lengthens or shortens the cable housing relative to a fixed anchor point. Turning the barrel adjuster out tightens the cable; Turning it in loosens the cable.

Bowden cables can cease to function smoothly if water or contaminants get into the housing. In cold climates Bowden cables are prone to malfunction due to water freezing. Cables also wear through use over a long time, and can be damaged through kinking or raveling. Another common failure occurs at the point where the housing enters a barrel adjuster when loose housing ends tend to fray the housing, making adjustments uncertain.

There is some controversy regarding "cable stretch". Newly installed cables can seem to elongate, requiring readjustment. While inner wires stretch very little, if at all, housings and linings may compress slightly, and all

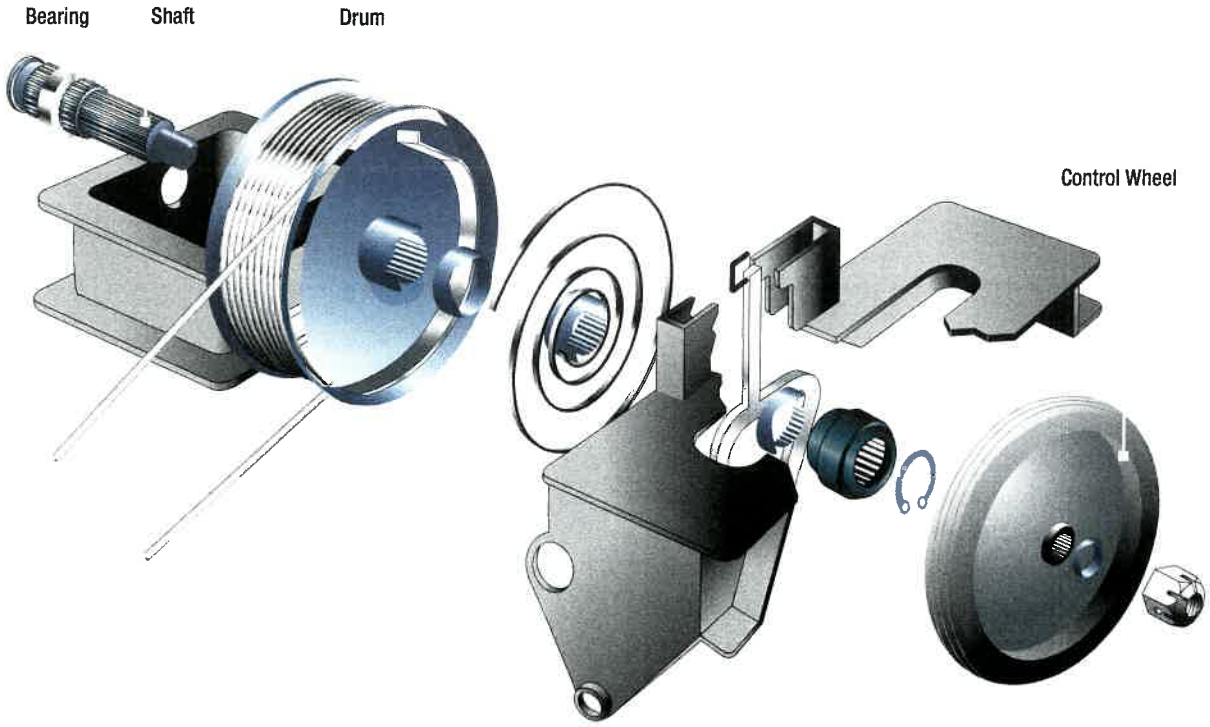


Figure 10-9. Trim tab cable drum.



Figure 10-10. Bowden cable with fittings configured for throttle control.



Figure 10-11. A Teleflex cable with a vernier fitting to allow both large and micro adjustments.

parts may generally "settle in". Lightweight assemblies such as those used on bicycles or ultralight aircraft are more susceptible to this.

TELEFLEX® CONTROLS

Teleflex® controls are installed in modern aircraft and used to operate components such as engine and propeller controls, trimming controls, and fuel valves. They are capable of transmitting both push and pull motion through a single cable. Like the Bowden system, the Teleflex® cable consists of a flexible transmitting cable operating inside a rigid or flexible metal conduit. Its advantages over the Bowden system are that it provides a more accurate and positive control throughout the range of movement and that the internal component can be temporarily locked in any desired position. (***Figure 10-11***)

The basic Teleflex® components are its cable, which is wound with a left or right hand helix wire, a protective conduit and a wide range of end fittings. Depending on the distances between the cockpit and the components to be operated, the control run may or may not be completely Teleflex®. In most instances a single length of Teleflex® cable is used, operating on the pull-push mode. In longer installations, rigid linkages may be used to bridge the gap between Teleflex® units at the extreme ends of the run.

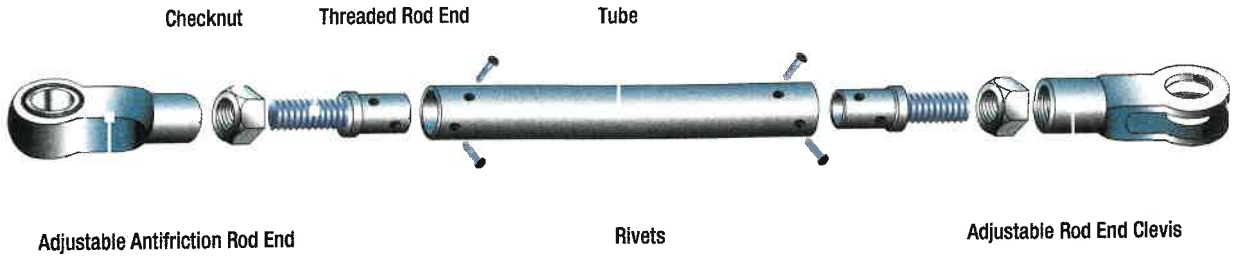


Figure 10-12. Push rod.

CONTROL RODS

PUSH-PULL RODS

Push rods are used as links in the flight control system to give push-pull motion. They may be adjusted at one or both ends. ***Figure 10-12*** shows the parts of a push rod.

Notice that it consists of a tube with threaded rod ends. An adjustable antifriction rod end, or rod end clevis, attaches at each end of the tube. The rod end, or clevis, permits attachment of the tube to flight control system parts. The checknut, when tightened, prevents the rod end or clevis from loosening. They may have adjustments at one or both ends. The rods should be perfectly straight, unless designed to be otherwise. When installed as part of a control system, the assembly should be checked for correct alignment and free movement.

It is possible for control rods fitted with bearings to become disconnected because of failure of the peening that retains the ball races in the rod end. This can be avoided by installing the control rods so that the flange of the rod end is interposed between the ball race and the anchored end of the attaching pin or bolt as shown in ***Figure 10-13***.

Another alternative is to place a washer, having a larger diameter than the hole in the flange, under the retaining nut on the end of the attaching pin or bolt. This retains the rod on the bolt in the event of a bearing failure.

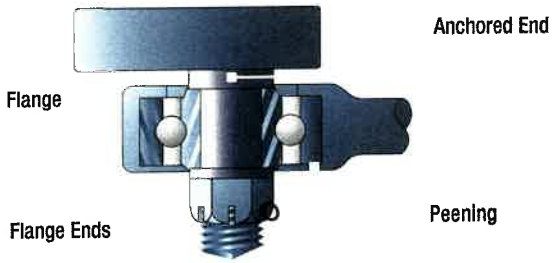


Figure 10-13. Attached rod end.

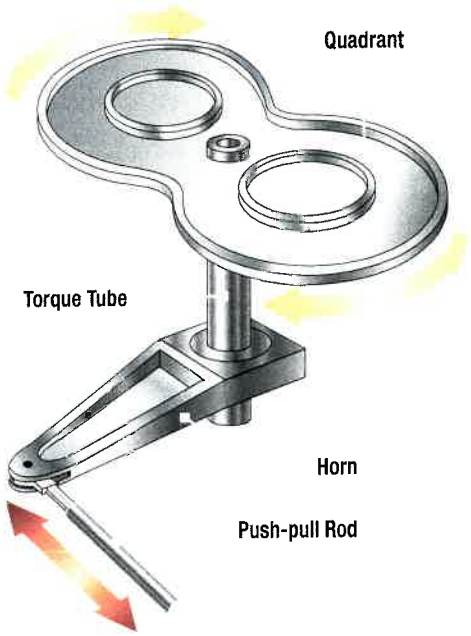


Figure 10-14. Torque tube.

TORQUE TUBES

Where an angular or twisting motion is needed in a control system, a torque tube is installed. ***Figure 10-14*** shows how a torque tube is used to transmit motion in opposite directions.

Question: 10-1
Why do larger aircraft which fly at higher altitudes require on-board tension regulators?

Question: 10-2
What determines the type of cable to use in a control system application (7 × 9 versus 7 × 19 cable)?

Question: 10-3
When inspecting a pulley you notice wear on both sides of the pulley wall. What does this mean?

Question: 10-4
Name two ways in which a turnbuckle can be prevented from loosening in flight.

Question: 10-5
Name two application in which you would use a Bowden cable in a small airplane.

Question: 10-6
What are advantages of a Teleflex control cable over a Bowden cable?

ANSWERS

Answer: 10-1
Cooling temperatures at altitude cause the metal airframe to shrink at a faster rate than the cables.

Answer: 10-2
The extent of direction changes present in the cable run.

Answer: 10-3
The pulley is misaligned.

Answer: 10-4
Safety wire locking methods or locking clips.

Answer: 10-5
Low tension non structural uses such as the throttle control, carburetor mixture control.

Answer: 10-6
Small precise adjustments are possible; it may be temporarily locked in place.



MATERIALS AND HARDWARE

ELECTRICAL CABLES AND CONNECTORS

SUB-MODULE 11

PART-66 SYLLABUS LEVELS

CERTIFICATION CATEGORY → B1

Sub-Module 11

ELECTRICAL CABLES AND CONNECTORS

Knowledge Requirements

6.11 - Electrical Cables and Connectors

Cable types, construction and characteristics; 2

High tension and co-axial cables;

Crimping;

Connector types, pins, plugs, sockets, insulators, current and voltage rating, coupling, identification codes.

6.11 - ELECTRICAL CABLES AND CONNECTORS

CABLE (WIRE) TYPES

The satisfactory performance of any modern aircraft depends to a very great degree on the continuing reliability of electrical systems and subsystems. Improperly or carelessly maintained wiring can be a source of both immediate and potential danger. The continued proper performance of electrical systems depends on the knowledge and techniques of the technician who installs, inspects, and maintains the electrical system wires and cables.

Procedures and practices outlined in this section are general recommendations and are not intended to replace the manufacturer's instructions and approved practices. A wire is described as a single, solid conductor, or as a stranded conductor covered with an insulating material. ***Figure 11-1*** illustrates these two definitions of a wire. Because of in-flight vibration and flexing, conductor round wire should be stranded to minimize fatigue breakage. The term "cable," as used in aircraft electrical installations, includes:

1. Two or more separately insulated conductors in the same jacket.
2. Two or more separately insulated conductors twisted together (twisted pair).
3. One or more insulated conductors covered with a metallic braided shield (shielded cable).
4. A single insulated center conductor with a metallic braided outer conductor (radio frequency cable).

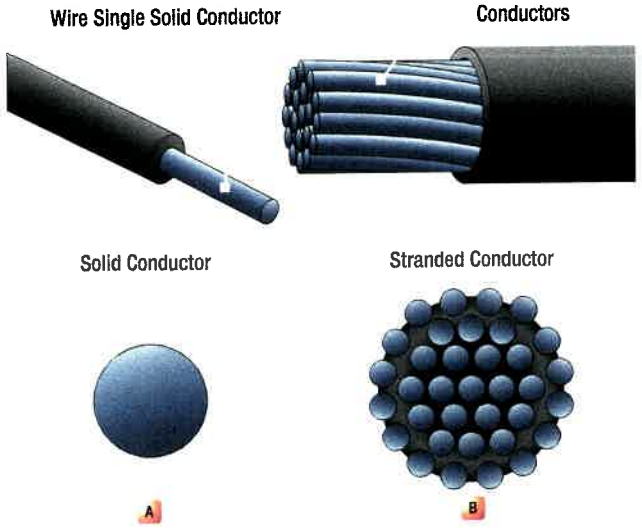


Figure 11-1. Aircraft electrical cable.

The term "wire harness" is used when an array of insulated conductors are bound together by lacing cord, metal bands, or other binding in an arrangement suitable for use only in specific equipment for which the harness was designed; it may include terminations. Wire harnesses are extensively used in aircraft to connect all the electrical components. (***Figure 11-2***)

For many years, the standard wire in light aircraft has been MIL-W-5086A, which uses a tin coated copper conductor rated at 600 volts and temperatures of 105°C. This basic wire is then coated with various insulating coatings. Commercial and military aircraft use wire that is manufactured under MIL-W-22759 specification, which complies with current military and FAA requirements.

The most important consideration in the selection of aircraft wire is properly matching the wire's construction to the application environment. Wire construction that is suitable for the most severe environmental condition to be encountered should be selected. Wires are typically categorized as being suitable for either open wiring or protected wiring application. The wire temperature rating is typically a measure of the insulation's ability to withstand the combination of ambient temperature and current related conductor temperature rise.

CONDUCTORS

The two most generally used conductors are copper and aluminum. Each has characteristics that make its use advantageous under certain circumstances. Also, each has certain disadvantages. Copper has a higher conductivity; is more ductile; has relatively high tensile strength; and can be easily soldered. Copper is more expensive and heavier than aluminum. Although



Figure 11-2. Wire harness.

aluminum has only about 60 percent of the conductivity of copper, it is used extensively. Its lightness makes possible long spans, and its relatively large diameter for a given conductivity reduces corona (the discharge of electricity from the wire when it has a high potential). The discharge is greater when small diameter wire is used than when large diameter wire is used. Some bus bars are made of aluminum instead of copper where there is a greater radiating surface for the same conductance. The characteristics of copper and aluminum are compared in ***Figure 11-3***.

Characteristic	Copper	Aluminum
Tensile strength (lb-in)	55 000	25 000
Tensile strength for same conductivity (lb)	55 000	40 000
Weight for same conductivity (lb)	100	48
Cross section for same conductivity (CM)	100	160
Specific resistance (ohm/mil ft)	10.6	17

Figure 11-3. Aircraft electrical cable.

Plating

Bare copper develops a surface oxide coating at a rate dependent on temperature. This oxide film is a poor conductor of electricity and inhibits determination of wire. Therefore, all aircraft wiring has a coating of tin, silver, or nickel that has far slower oxidation rates.

1. Tin-coated copper is a very common plating material. Its ability to be successfully soldered without highly active fluxes diminishes rapidly with time after manufacture. It can be used up to the limiting temperature of 150°C.
2. Silver-coated wire is used where temperatures do not exceed 200°C.
3. Nickel-coated wire retains its properties beyond 260°C (500°F), but most aircraft wire using such coated strands has insulation systems that cannot exceed that temperature on long-term exposure. Soldered terminations of nickel-plated conductor require the use of different solder sleeves or flux than those used with tin or silver plated conductor.

INSULATION

Two fundamental properties of insulation materials are insulation resistance and dielectric strength. These are entirely different and distinct properties. Insulation resistance is the resistance to current leakage through and over the surface of insulation materials. Insulation resistance can be measured with a megohmmeter/ insulation tester without damaging the insulation, and data so obtained serves as a useful guide in determining the general condition of the insulation. However, the data obtained in this manner may not give a true picture of the condition of the insulation. Clean, dry insulation having cracks or other faults might show a high value of insulation resistance but would not be suitable for use.

Dielectric strength is the ability of the insulator to withstand potential difference and is usually expressed in terms of the voltage at which the insulation fails because of the electrostatic stress. Maximum dielectric strength values can be measured by raising the voltage of a test sample until the insulation breaks down.

The type of conductor insulation material varies with the type of installation. Characteristics should be chosen based on environment, such as abrasion resistance, arc resistance, corrosion resistance, cut-through strength, dielectric strength, flame resistant, mechanical strength, smoke emission, fluid resistance, and heat distortion. Such types of insulation materials (e.g., PVC/nylon, Kapton®, and Teflon®) are no longer used for new aircraft designs, but might still be installed on older aircraft. Insulation materials for new aircraft designs are made of Tefzel®, Teflon®/Kapton®/ Teflon® and PTFE/ Polyimide/PTFE. The development of better and safer insulation materials is ongoing.

Since electrical wire may be installed in areas where inspection is infrequent over extended periods of time, it is necessary to give special consideration to heat-aging characteristics in the selection of wire. Resistance to heat is of primary importance in the selection of wire for aircraft use, as it is the basic factor in wire rating. Where wire may be required to operate at higher temperatures due either to high ambient temperatures, high current loading, or a combination of the two, selection should be made on the basis of satisfactory performance under the most severe operating conditions.

WIRE SHIELDING

With the increase in number of highly sensitive electronic devices found on modern aircraft, it has become very important to ensure proper shielding for many electric circuits. Shielding is the process of applying a metallic covering to wiring and equipment to eliminate Electromagnetic Interference (EMI). EMI is caused when electromagnetic fields (radio waves) induce

High Frequency (HF) voltages in a wire or component. The induced voltage can cause system inaccuracies or failure. (***Figure 11-4***)

Use of shielding with 85 percent coverage or greater is recommended. Coaxial, triaxial, twinaxial, or quadraxial cables should be used, wherever appropriate, with their shields connected to ground at a single point or multiple points, depending upon the purpose of the shielding. The airframe grounded structure may also be used as an EMI shield.

WIRE SUBSTITUTIONS

When a replacement wire is required in the repair and modification of existing aircraft, the maintenance manual for that aircraft must first be reviewed to determine if the Original Aircraft Manufacturer (OAM) has approved any substitution. If not, then the manufacturer must be contacted for an acceptable replacement.

Severe Wind And Moisture Problem (SWAMP)

SWAMP areas differ from aircraft to aircraft but are usually wheel wells, near wing flaps, wing folds, pylons, and other exterior areas that may have a harsh environment. Wires in these areas have often an exterior jacket to protect them from the environment. Wires for these applications often have design features incorporated into their construction that may make the wire unique; therefore, an acceptable substitution may be difficult, if not impossible to find. It's very important to use the wire type recommended in the aircraft manufacturer's maintenance handbook. Insulation or jacketing varies according to the environment. (***Figure 11-5***)

Wire Size Selection

Wire is manufactured in sizes according to a standard known as the American Wire Gauge (AWG). As shown in ***Figure 11-6***, the wire diameters become smaller as the gauge numbers become larger. Typical wire sizes range from a number 40 to number 0000.

Gauge numbers are useful in comparing the diameter of wires, but not all types of wire or cable can be measured accurately with a gauge. Larger wires are usually stranded to increase their flexibility. In such cases, the total area can be determined by multiplying the area of one strand (usually computed in circular mils when diameter or gauge number is known) by the number of strands in the wire or cable.



Figure 11-4. Shielded wire harness for flight control.

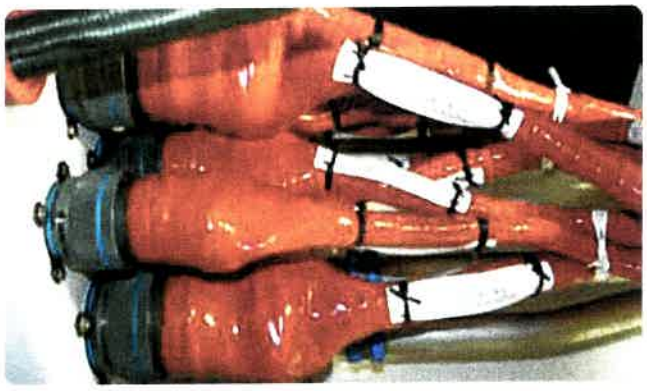


Figure 11-5. Wire harness with protective jacket.

Several factors must be considered in selecting the size of wire for transmitting and distributing electric power.

1. Wires must have sufficient mechanical strength to allow for service conditions.
2. Allowable power loss (I2 R loss) in the line represents electrical energy converted into heat. The use of large conductors reduces the resistance and therefore the I2 R loss. However, large conductors are more expensive, heavier, and need more substantial support.
3. If the source maintains a constant voltage at the input to the lines, any variation in the load on the line causes a variation in line current and a consequent variation in the IR drop in the line. A wide variation in the IR drop in the line causes poor voltage regulation at the load. The obvious remedy is to reduce either current or resistance. A reduction in load current lowers the amount of power being transmitted, whereas a reduction in line resistance increases the size and weight of conductors required. A compromise is generally reached whereby the voltage variation at the load is within tolerable limits and the weight of line conductors is not excessive.

Cross Section			Ohms per 1 000 ft		
Gauge Number	Diameter (mils)	Circular (mils)	Square Inches	25 °C (77 °F)	65 °C (149 °F)
0000	460.0	212 000.0	0.166	0.050 0	0.057 7
000	410.0	168 000.0	0.132	0.063 0	0.072 7
00	365.0	133 000.0	0.105	0.079 5	0.091 7
0	325.0	106 000.0	0.082 9	0.100	0.166
1	289.0	83 700.0	0.065 7	0.126	0.146
2	258.0	66 400.0	0.052 1	0.159	0.184
3	229.0	52 600.0	0.041 3	0.201	0.232
4	204.0	41 700.0	0.032 8	0.253	0.292
5	182.0	33 100.0	0.026 0	0.319	0.369
6	162.0	26 300.0	0.020 6	0.403	0.465
7	144.0	20 800.0	0.016 4	0.508	0.586
8	128.0	16 500.0	0.013 0	0.641	0.739
9	114.0	13 100.0	0.010 3	0.808	0.932
10	102.0	10 400.0	0.008 15	1.02	1.18
11	91.0	8 230.0	0.006 47	1.28	1.48
12	81.0	6 530.0	0.005 13	1.62	1.87
13	72.0	5 180.0	0.004 07	2.04	2.36
14	64.0	4 110.0	0.003 23	2.58	2.97
15	57.0	3 260.0	0.002 56	3.25	3.75
16	51.0	2 580.0	0.002 03	4.09	4.73
17	45.0	2 050.0	0.001 61	5.16	5.96
18	40.0	1 620.0	0.001 28	6.51	7.51
19	36.0	1 290.0	0.001 01	8.21	9.48
20	32.0	1 020.0	0.000 802	10.40	11.90
21	28.5	810.0	0.000 636	13.10	15.10
22	25.3	642.0	0.000 505	16.50	19.00
23	22.6	509.0	0.000 400	20.80	24.00
24	20.1	404.0	0.000 317	26.20	30.20
25	17.9	320.0	0.000 252	33.00	38.10
26	15.9	254.0	0.000 200	41.60	48.00
27	14.2	202.0	0.000 158	52.50	60.60
28	12.6	160.0	0.000 126	66.20	76.40
29	11.3	127.0	0.000 099 5	83.40	96.30
30	10.0	101.0	0.000 078 9	105.00	121.00
31	8.9	79.7	0.000 062 6	133.00	153.00
32	8.0	63.2	0.000 049 6	167.00	193.00
33	7.1	50.1	0.000 039 4	211.00	243.00
34	6.3	39.8	0.000 031 2	266.00	307.00
35	5.6	31.5	0.000 024 8	335.00	387.00
36	5.0	25.0	0.000 019 6	423.00	488.00
37	4.5	19.8	0.000 015 6	533.00	616.00
38	4.0	15.7	0.000 012 3	673.00	776.00
39	3.5	12.5	0.000 009 8	848.00	979.00
40	3.1	9.9	0.000 007 8	1 070.00	1 230.00

Figure 11-6. American wire gauge for standard annealed solid copper wire.

4. When current is drawn through the conductor, heat is generated. The temperature of the wire rises until the heat radiated, or otherwise dissipated, is equal to the heat generated by the passage of current through the line. If the conductor is insulated, the heat generated in the conductor is not so readily removed as it would be if the conductor were not insulated. Thus, to protect the insulation from too much heat, the current through the conductor must be maintained below a certain value. When electrical conductors are installed in locations where the ambient temperature is relatively high, the heat generated by external sources constitutes an appreciable part of the total conductor heating. Allowance must be made for the influence of external heating on the allowable conductor current, and each case has its own specific limitations. The maximum allowable operating temperature of insulated conductors varies with the type of conductor insulation being used.

If it is desirable to use wire sizes smaller than #20, particular attention should be given to the mechanical strength and installation handling of these wires (e.g., vibration, flexing, and termination). Wires containing less than 19 strands must not be used. Consideration should be given to the use of high-strength alloy conductors in small gauge wires to increase mechanical strength. As a general practice, wires smaller than size #20 should be provided with additional clamps and be grouped with at least three other wires. They should also have additional support at terminations, such as connector grommets, strain relief clamps, shrinkable sleeving, or telescoping bushings. They should not be used in applications where they are subjected to excessive vibration, repeated bending, or frequent disconnection from screw termination. (***Figure 11-7***)

Current Carrying Capacity

In some instances, the wire may be capable of carrying more current than is recommended for the contacts of the related connector. In this instance, it is the contact rating that dictates the maximum current to be carried by a wire. Wires of larger gauge may need to be used to fit within the crimp range of connector contacts that are adequately rated for the current being carried. ***Figure 11-8*** gives a family of curves whereby the bundle derating factor may be obtained.

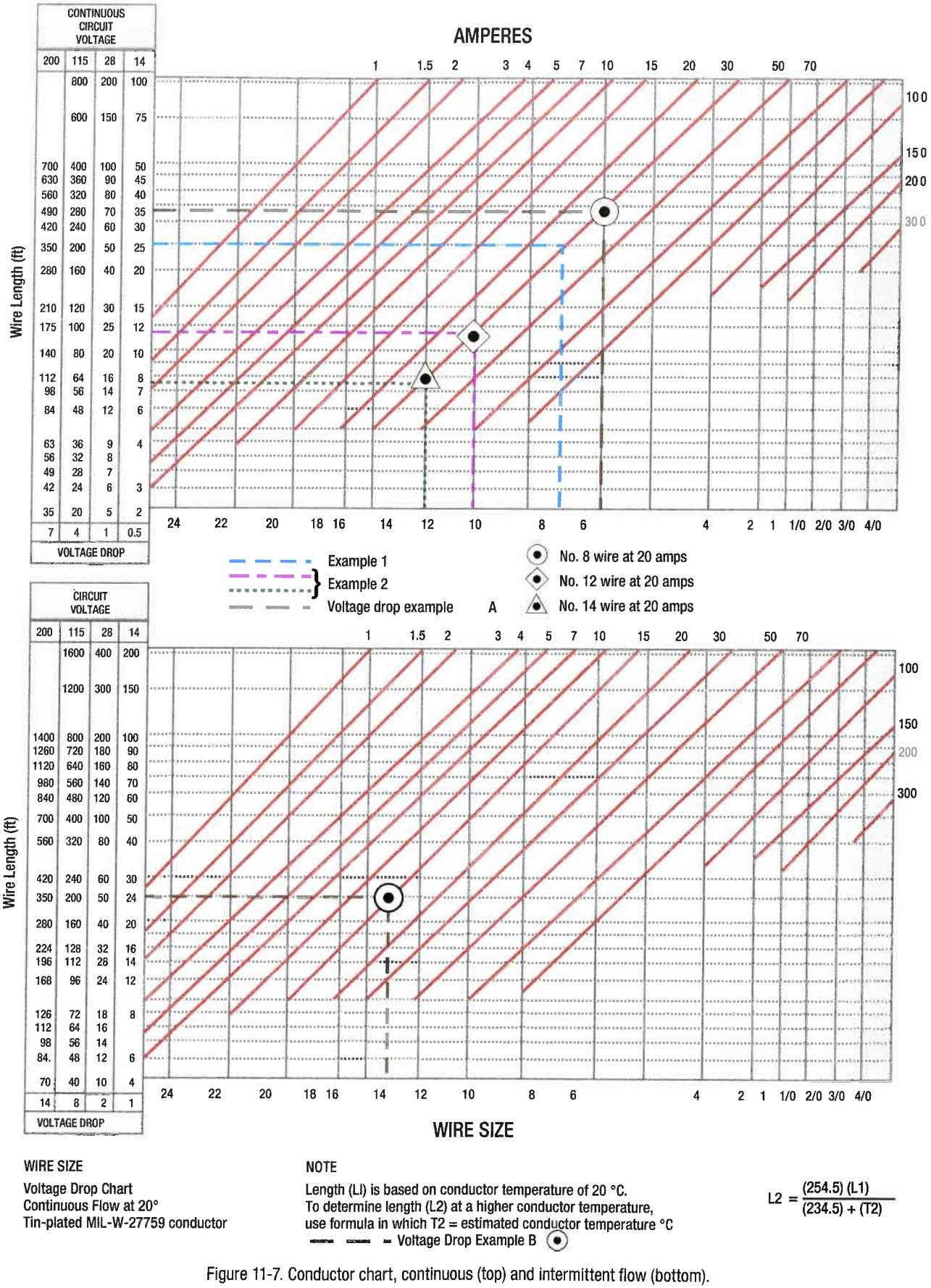
Maximum Operating Temperature

The current that causes a temperature steady state condition equal to the rated temperature of the wire should not be exceeded. Rated temperature of the wire may be based upon the ability of either the conductor or the insulation to withstand continuous operation without degradation.

1. Single Wire in Free Air—Determining a wiring system's current-carrying capacity begins with determining the maximum current that a given-sized wire can carry without exceeding the allowable temperature difference (wire rating minus ambient °C). The curves are based upon a single copper wire in free air. (***Figure 11-8***)
2. Wires in a Harness—When wires are bundled into harnesses, the current derived for a single wire must be reduced, as shown in ***Figure 11-9***. The amount of current derating is a function of the number of wires in the bundle and the percentage of the total wire bundle capacity that is being used.
3. Harness at Altitude—Since heat loss from the bundle is reduced with increased altitude, the amount of current should be derated. ***Figure 11-10*** gives a curve whereby the altitude-derating factor may be obtained.
4. Aluminum Conductor Wire—When aluminum conductor wire is used, sizes should be selected on the basis of current ratings shown in ***Figure 11-11***. The use of sizes smaller than #8 is discouraged. Aluminum wire should not be attached to engine mounted accessories or used in areas having corrosive fumes, severe vibration, mechanical stresses, or where there is a need for frequent disconnection. Use of aluminum wire is also discouraged for runs of less than 3 feet. Termination hardware should be of the type specifically designed for use with aluminum conductor wiring.

Computing Current Carrying Capacity

The following section presents some examples on how to calculate the load carrying capacity of aircraft electrical wire. The calculation is a step by step approach and several graphs are used to obtain information to compute the current carrying capacity of a particular wire.



WIRE SIZE

Voltage Drop Chart
Continuous Flow at 20°
Tin-plated MIL-W-27759 conductor

NOTE

Length (LI) is based on conductor temperature of 20 °C.
To determine length (L2) at a higher conductor temperature, use formula in which T2 = estimated conductor temperature °C

— — — Voltage Drop Example B

$$L2 = \frac{(254.5)\,(L1)}{(234.5) + (T2)}$$

Figure 11-7. Conductor chart, continuous (top) and intermittent flow (bottom).

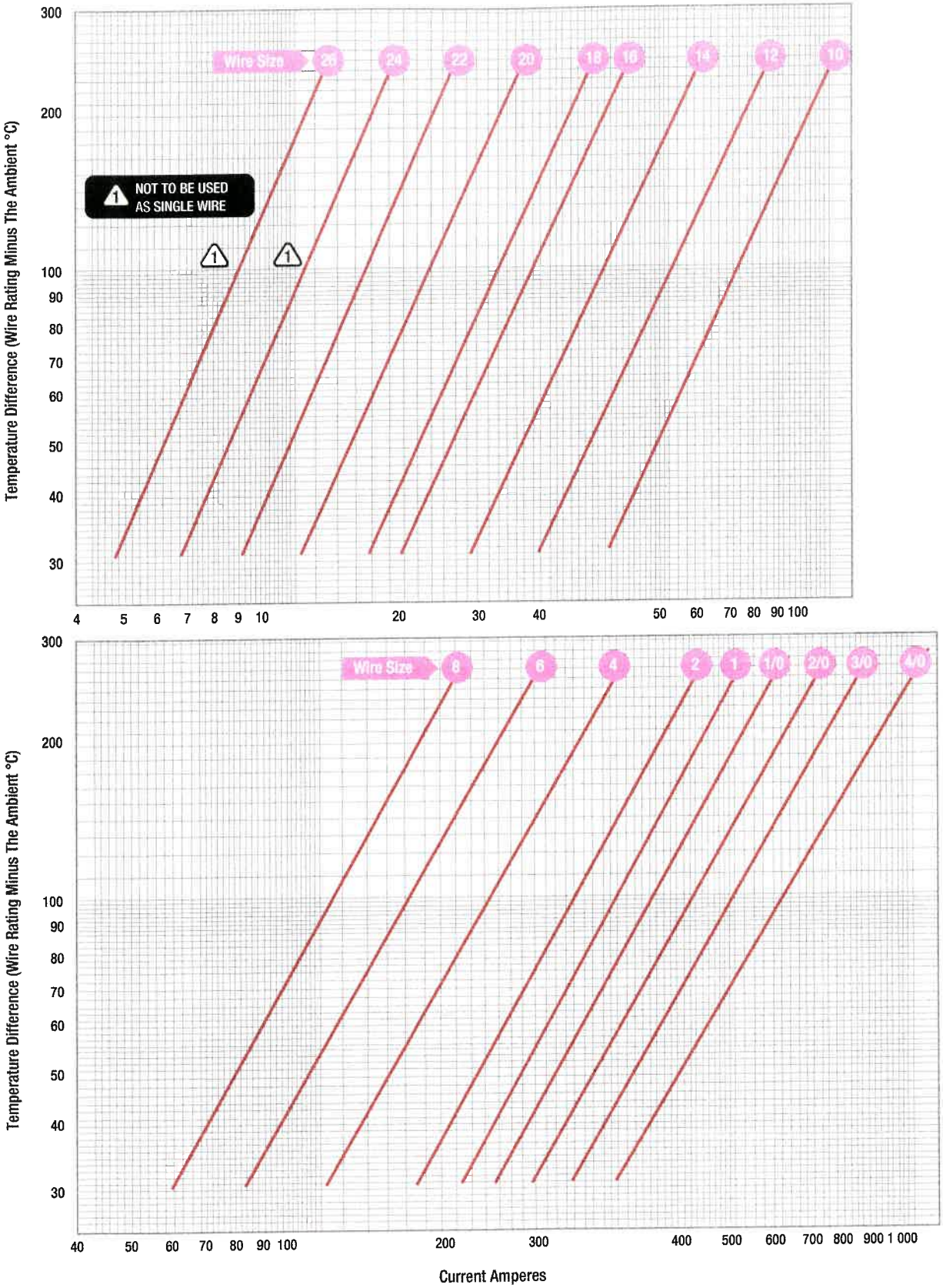


Figure 11-8. Single copper wire in free air.

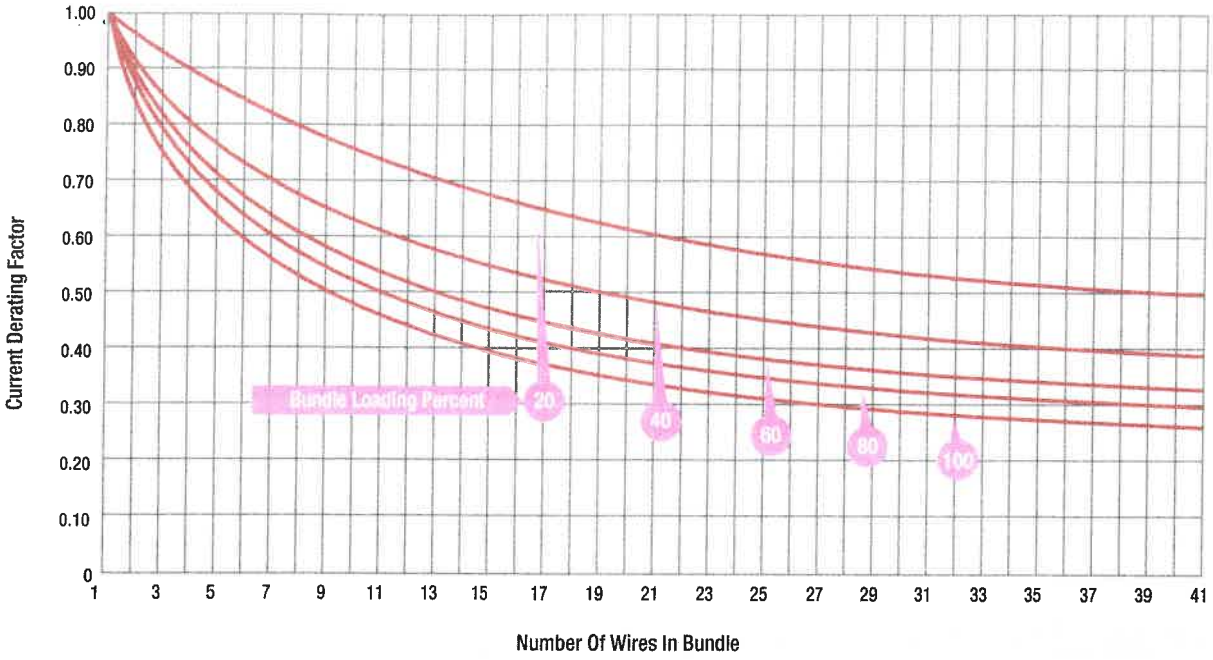


Figure 11-9. Bundle derating curve.

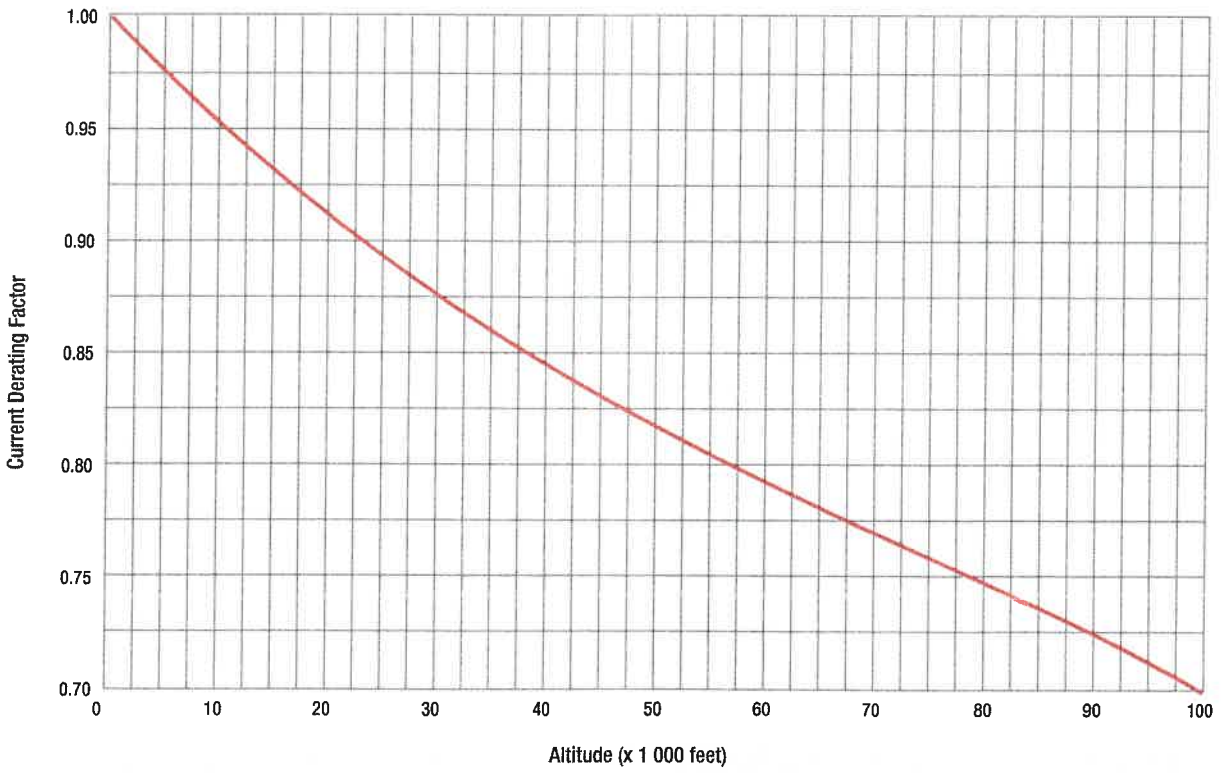


Figure 11-10. Altitude derating curve.

Wire Size	Continuous Duty Current (amp) Wires in Bundles, Groups, or Harnesses or Conduits		Max. Resistance ohms/1000feet
	Wire conductor temperature rating		
	@ 105 °C	@ 150 °C	@ 20 °C
#8	30	45	1.093
#6	40	61	0.641
#4	54	82	0.427
#2	76	113	0.268
#1	90	133	0.214
#0	102	153	0.169
#00	117	178	0.133
#000	138	209	0.109
#0000	163	248	0.085

Figure 11-11. Current-carrying capacity and resistance of aluminum wire.

Example 1:
Assume a harness (open or braided) consisting of 10 wires, size 20, 200°C rated copper, and 25 wires size 22, 200°C rated copper, is installed in an area where the ambient temperature is 60°C and the aircraft is capable of operating at a 35 000 foot altitude. Circuit analysis reveals that 7 of the 35 wires in the bundle (7/35 = 20 percent) are carrying power currents near or up to capacity.

- Step 1—Refer to single wire in free air curves in ***Figure 11-5***. Determine the change of temperature of the wire to determine free air ratings. Since the wire is in ambient temperature of 60°C and rated at 200°C, the change of the temperature is 200°C–60°C = 140°C. Follow 140°C temperature difference horizontally until it intersects with wire size line on ***Figure 11-4***. The free air rating for size 20 is 21.5 amps, and free air rating for size 22 is 16.2 amps.
- Step 2—Refer to the bundle derating curves in ***Figure 11-9***. The 20 percent curve is selected since circuit analysis indicate that 20 percent or less of the wire in the harness would be carrying power currents and less than 20 percent of the bundle capacity would be used. Find 35 (on the horizontal axis), since there are 35 wires in the bundle, and determine a derating factor of 0.52 (on the vertical axis) from the 20 percent curve.
- Step 3—Derate the size 22 free air rating by multiplying 16.2 by 0.52 to get 8.4 amps in harness rating. Derate the size 20 free air rating by multiplying 21.5 by 0.52 to get 11.2 amps in harness rating.
- Step 4—Refer to the altitude derating curve of ***Figure 11-10***. Look for 35 000 feet (on the horizontal axis) since that is the altitude at which the aircraft is operating. Note that the wire must be derated by a factor of 0.86 (found on the vertical axis). Derate the size 22 harness rating by multiplying 8.4 amps by 0.86 to get 7.2 amps. Derate the size 20 harness rating by multiplying 11.2 amps by 0.86 to get 9.6 amps.
- Step 5—To find the total harness capacity, multiply the total number of size 22 wires by the derated capacity (25 × 7.2 = 180.0 amps) and add to that the number of size 20 wires multiplied by the derated capacity (10 × 9.6 = 96.8 amps) and multiply the sum by the 20 percent harness capacity factor. Thus, the total harness capacity is (180.0 + 96.0) × 0.20 = 55.2 amps. It has been determined that the total harness current should not exceed 55.2 A, size 22 wire should not carry more than 7.2 amps and size 20 wire should not carry more than 9.6 amps.
- Step 6—Determine the actual circuit current for each wire in the bundle and for the whole bundle. If the values calculated in step 5 are exceeded, select the next larger size wire and repeat the calculations.

Example 2:
Assume a harness (open or braided), consisting of 12 size 12, 200°C rated copper wires, is operated in an ambient temperature of 25°C at sea level and 60°C at a 20 000- foot altitude. All 12 wires are operated at or near their maximum capacity.

- Step 1—Refer to the single wire in free air curve in ***Figure 11-8***, determine the temperature difference of the wire to determine free air ratings. Since the wire is in ambient temperature of 25°C and 60°C and is rated at 200°C, the temperature differences are 200°C – 25°C = 175°C and 200°C – 60°C = 140°C, respectively. Follow the 175°C and the 140°C temperature difference lines on ***Figure 11-7*** until each intersects wire size line. The free air ratings of size 12 are 68 amps and 59 amps, respectively.
- Step 2—Refer to the bundling derating curves in ***Figure 11-11***. The 100 percent curve is selected because we know all 12 wires are carrying full load. Find 12 (on the horizontal axis) since there are 12 wires in the bundle and determine a derating factor of 0.43 (on the vertical axis) from the 100 percent curve.

- Step 3—Derate the size #12 free air ratings by multiplying 68 amps and 61 amps by 0.43 to get 29.2 amps and 25.4 amps, respectively.
- Step 4—Refer to the altitude derating curve of ***Figure 11-10***, look for sea level and 20 000 feet (on the horizontal axis) since these are the conditions at which the load is carried. The wire must be derated by a factor of 1.0 and 0.91, respectively.
- Step 5—Derate the size 12 in a bundle ratings by multiplying 29.2 amps at sea level and 25.4 amps at 20 000 feet by 1.0 and 0.91, respectively to obtain 29.2 amps and 23.1 amps. The total bundle capacity at sea level and 25°C ambient temperature is 29.2 × 12 = 350.4 amps. At 20 000 feet and 60°C ambient temperature, the bundle capacity is 23.1 × 12 = 277.2 amps. Each size 12 wire can carry 29.2 amps at sea level, 25°C ambient temperature or 23.1 amps at 20 000 feet and 60°C ambient temperature.
- Step 6—Determine the actual circuit current for each wire in the bundle and for the bundle. If the values calculated in Step 5 are exceeded, select the next larger size wire and repeat the calculations.

Allowable Voltage Drop

The voltage drop in the main power wires from the generation source or the battery to the bus should not exceed 2 percent of the regulated voltage when the generator is carrying rated current or the battery is being discharged at the 5-minute rate. The tabulation shown in ***Figure 11-12*** defines the maximum acceptable voltage drop in the load circuits between the bus and the utilization equipment ground.

The resistance of the current return path through the aircraft structure is generally considered negligible. However, this is based on the assumption that adequate bonding to the structure or a special electric current return path has been provided that is capable of carrying the required electric current with a negligible voltage drop. To determine circuit resistance, check the voltage drop across the circuit. If the voltage drop does not exceed the limit established by the aircraft or product manufacturer, the resistance value for the circuit may be considered satisfactory. When checking a circuit, the input voltage should be maintained at a constant value. ***Figures 11-13 and 11-14*** show formulas that may be used to determine electrical resistance in wires and some typical examples.

The following formula can be used to check the voltage drop. The resistance/ft can be found in ***Figures 11-13 and 11-14*** for the wire size.

Calculated Voltage Drop (VD) = resistance/ft × length × current

Nominal System Voltage	Allowable Voltage Drop During Continuous Operation	Intermittent Operation
14	0.5	1
28	1	2
115	4	8
200	7	14

Figure 11-12. Tabulation chart (allowable voltage drop between bus and utilization equipment ground).

Voltage Drop	Run Lengths (feet)	Circuit Current (amps)	Wire Size from Chart	Check Calculated Voltage Drop (VD) = (resistance/feet) (length) (current)
1	107	20	No. 6	VD = (0.000 44 ohms/feet) (107 x 20) = 0.942
0.5	90	20	No. 4	VD = (0.000 28 ohms/feet) (90 x 20) = 0.504
4	88	20	No. 12	VD = (0.002 02 ohms/feet) (88 x 20) = 3.60
7	100	20	No. 14	VD = (0.003 06 ohms/feet) (100 x 20) = 6.12

Figure 11-13. Determining required tin-plated copper wire size and checking voltage drop.

Maximum Voltage Drop	Wire Size	Circuit Current (amps)	Maximum Wire Run Length (feet)	Check Calculated Voltage Drop (VD) = (resistance/feet) (length) (current)
1	No. 10	20	39	VD = (0.001 26 ohms/feet) (39 x 20) = 0.98
0.5	---		19.5	VD = (0.001 26 ohms/feet) (19.5 x 20) = 0.366
4	---		156	VD = (0.001 26 ohms/feet) (156 x 20) = 3.93
7	---		273	VD = (0.001 26 ohms/feet) (273 x 20) = 6.88

Figure 11-14. Determining required tin-plated copper wire size and checking maximum voltage drop.

Electric Wire Chart Instructions

To select the correct size of electrical wire, two major requirements must be met:

1. The wire size should be sufficient to prevent an excessive voltage drop while carrying the required current over the required distance. (***Figure 11-12***)
2. The size should be sufficient to prevent overheating of the wire carrying the required current. See Maximum Operating Temperature earlier in this chapter for computing current carrying capacity methods.

To meet the two requirements for selecting the correct wire size using ***Figure 11-7***, the following must be known before hand:

1. The wire length in feet.
2. The number of amperes of current to be carried.
3. The allowable voltage drop permitted.
4. The required continuous or intermittent current.
5. The estimated or measured conductor temperature.
6. Is the wire to be installed in conduit and/or bundle?
7. Is the wire to be installed as a single wire in free air?

Example A:
Find the wire size in ***Figure 11-7*** using the following known information:

1. Wire run is 50 feet long, including the ground wire.
2. Current load is 20 amps.
3. The voltage source is 28 volts from bus to equipment.
4. The circuit has continuous operation.
5. Estimated conductor temperature is 20°C or less. The scale on the left of the chart represents maximum wire length in feet to prevent an excessive voltage drop for a specified voltage source system (e.g., 14V, 28V, 115V, 200V). This voltage is identified at the top of scale and the corresponding voltage drop limit for continuous operation at the bottom. The scale (slant lines) on top of the chart represents amperes. The scale at the bottom of the chart represents wire gauge.

- Step 1—From the left scale, find the wire length 50 feet under the 28V source column.
- Step 2—Follow the corresponding horizontal line to the right until it intersects the slanted line for the 20-amp load.
- Step 3—At this point, drop vertically to the bottom of the chart. The value falls between No. 8 and No. 10. Select the next larger size wire to the right, in this case No. 8. This is the smallest size wire that can be used without exceeding the voltage drop limit expressed at the bottom of the left scale. This example is plotted on the wire chart in ***Figure 11-7***. Use ***Figure 11-7*** (top) for continuous flow and ***Figure 11-7*** (bottom) for intermittent flow.

Example B:
Find the wire size in ***Figure 11-7*** using the following known information:

1. Wire run is 200 feet long, including ground wire.
2. Current load is 10 amps.
3. The voltage source is 115 volts from bus to equipment.
4. The circuit has intermittent operation.

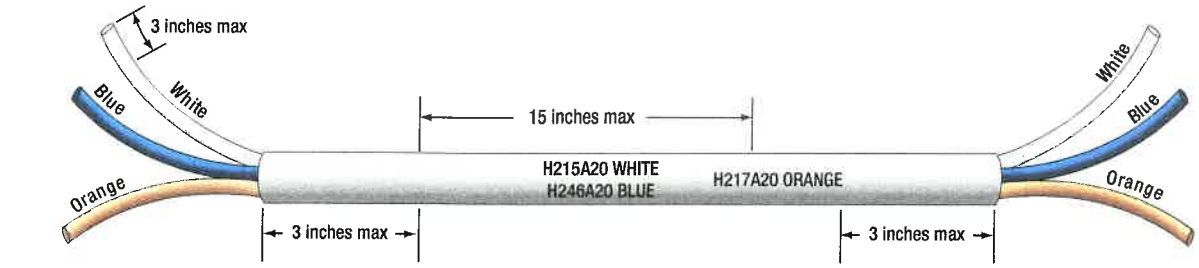
- Step 1—From the left scale, find the wire length of 200 feet under the 115V source column.
- Step 2—Follow the corresponding horizontal line to the right until it intersects the slanted line for the 10 amp load.
- Step 3—At this point, drop vertically to the bottom of the chart. The value falls between No. 16 and No. 14. Select the next larger size wire to the right—in this case, No. 14. This is the smallest size wire that can be used without exceeding the voltage drop limit expressed at the bottom of the left scale.

WIRE IDENTIFICATION

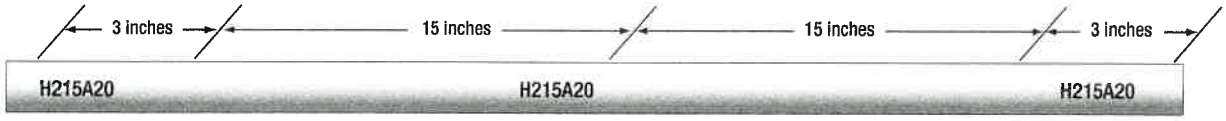
The proper identification of electrical wires and cables with their circuits and voltages is necessary to provide safety of operation, safety to maintenance personnel, and ease of maintenance. All wire used on aircraft must have its type identification imprinted along its length. It is common practice to follow this part number with the five digit/letter Commercial and Government Entity (CAGE) code identifying the wire manufacturer. You can identify the performance capabilities of existing installed wire you need to replace, and avoid the inadvertent use of a lower performance and unsuitable replacement wire.

Placement of Identification Markings

Identification markings should be placed at each end of the wire and at 15 inch maximum intervals along the length of the wire. Wires less than 3 inches (8 cm) in length need not be identified. Wires 3 to 7 inches (8-18 cm) in length should be identified approximately at the center. Added identification marker sleeves should be located so that ties, clamps, or supporting devices



A. Multiple Wires In A Sleeve



B. Single Wire Without Sleeve

Figure 11-15. Wire markings for single wire without sleeve.

need not be removed to read the identification. The wire identification code must be printed to read horizontally (from left to right) or vertically (from top to bottom).

The two methods of marking wire or cable are as follows:

1. Direct marking is accomplished by printing the cable's outer covering. (***Figure 11-15B***)
2. Indirect marking is accomplished by printing a heat shrinkable sleeve and installing the printed sleeve on the wire or cables outer covering. Indirectly marked wire or cable should be identified with printed sleeves at each end and at intervals not longer than 6 feet. (***Figure 11-16***) The individual wires inside a cable should be identified within 3 inches of their termination. (***Figure 11-15A***)

Types of Wire Markings

The preferred method is to mark directly on the wire without causing insulation degradation. Teflon-coated wires, shielded wiring, multiconductor cable, and thermocouple wires usually require special sleeves to carry identification marks. There are some special wire marking machines available that can be used to stamp directly on the type wires mentioned above. Whatever method of marking is used, the marking should be legible and the color should contrast with the wire insulation or sleeve. (***Figure 11-16***)

Several different methods can be used to mark directly on the wire: hot stamp marking, ink jet printers, and laser jet printers. (***Figure 11-17***)



Figure 11-16. Spacing of printed identification marks (indirect marking).



Figure 11-17. Laser wire printer.

The hot stamp method can damage the insulation of a newer type of wire that utilizes thin insulators. Fracture of the insulation wall and penetration to the conductor of these materials by the stamping dies have occurred. Later in service, when these openings have been wetted by various fluids or moisture, serious arcing and surface tracking have damaged wire bundles. Identification sleeves can be used if the direct marking on the wire is not possible. (***Figure 11-18***)

Flexible sleeving, either clear or opaque, is satisfactory for general use. When color-coded or striped component wire is used as part of a cable, the identification sleeve

should specify which color is associated with each wire identification code. Identification sleeves are normally used for identifying the following types of wire or cable: unjacketed shielded wire, thermocouple wire, coaxial cable, multiconductor cable, and high temperature wire. In most cases, identification tape can be used in place of sleeving. For sleeving exposed to high temperatures (over 400°F), materials, such as silicone fiberglass, should be used. Polyolefin sleeving should be used in areas where resistance to solvent and synthetic hydraulic fluids is necessary. Sleeves may be secured in place with cable ties or by heat shrinking.

The identification sleeving for various sizes of wire is shown in ***Figure 11-18***.

HIGH TENSION AND COAXIAL CABLES

HIGH TENSION CABLES

These cables are used for the transmission of high-tension voltages in both piston engine and turbine engine ignition systems, and are of the single-core stranded type suitably insulated, and screened by metal braided sheathing to prevent interference. The number of cables required for a system corresponds to that of the sparking plugs or igniter plugs as appropriate, and they are generally made up into a complete ignition cable harness.

Depending on the type of engine installation, the cables may be enclosed in a metal conduit, which also forms part of a harness, or they may be routed openly. Cables are connected to the relevant system components by special end fittings comprising either small springs or contact caps secured to the cable conductor, insulation, and a threaded coupling assembly. (***Figure 11-19***)

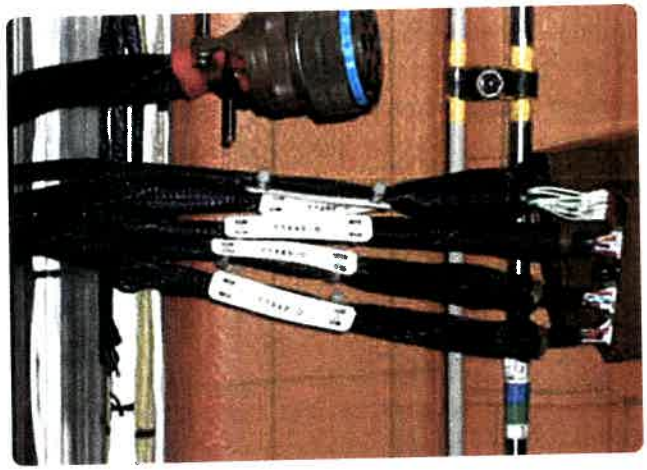


Figure 11-18. Alternate method of identifying wire bundles.

COAXIAL CABLE

Coaxial cables contain two or more separate conductors. The innermost conductor may be solid or stranded copper wire and may be plain, tinned, silver-plated or even gold plated in some applications, depending on the degree of conductivity required.

The remaining conductors are in the form of tubes, usually fine wire braid. The insulation is usually polyethylene or Teflon. Outer covering or jackets serve to weatherproof the cables and protect them from fluids or mechanical and electrical damage. The materials used for the covering are manufactured to suit operations under varying environmental conditions. (***Figure 11-20***)

Coaxial cables have two main advantages. First, they are shielded against electrostatic and magnetic fields. An electrostatic field does not extend beyond the outer conductor and the fields, due to current flow in inner and outer conductors cancel each other. Second, since Coaxial cables do not radiate energy, then likewise they will not pick up any energy, or be influenced by other strong fields.

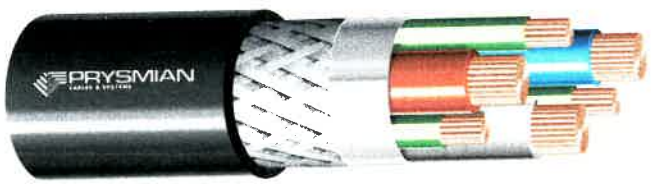


Figure 11-19. High tension cable in metal conduit.

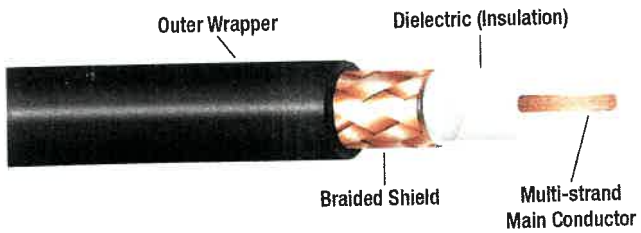


Figure 11-20. Coaxial cable.

The installations in which coaxial cables are most commonly employed are for avionics, the connection of antennae, capacitance fuel quantity indicating systems and for interconnection of tank units and amplifiers.

All wiring needs to be protected from damage. However, coaxial and triaxial cables are particularly vulnerable to certain types of damage. Personnel should exercise care while handling or working around coaxial. Coaxial damage can occur when clamped too tightly, or when they are bent sharply (normally at or near connectors). Damage can also be incurred during unrelated maintenance actions around the coaxial cable. Coaxial cable can be severely damaged on the inside without any evidence of damage on the outside. Coaxial cables with solid center conductors should not be used. Stranded center coaxial cables can be used as a direct replacement for solid center coaxial. Coaxial cable precautions include:

- Never kink coaxial cable.
- Never drop anything on coaxial cable.
- Never step on coaxial cable.
- Never bend coaxial cable sharply.
- Never loop coaxial cable tighter than the allowable bend radius.
- Never pull on coaxial cable except in a straight line.
- Never use coaxial cable for a handle, lean on it, or hang things on it (including other wires).

Coaxial Connector Connectors

Coaxial cable connectors come in several types depending on the methods of attachment with some including a flange for securing it to a bulkhead or other structure. ***Figure 11-21*** shows several types of coaxial connectors.

CRIMPING

The topic of *Crimping* has been moved to *Module 07, Submodule 07* and combined with the related *Module 07, Submodule 07* topics of wire termination, stripping, crimping tools, testing, and splicing.

CONNECTORS

Types, pins, plugs, sockets, insulators, current and voltage rating, coupling and identification codes.

AN AND MS CONNECTORS

Connectors (plugs and receptacles) facilitate maintenance when frequent disconnection is required. Electric connectors are designed in many sizes and shapes to facilitate the installation and maintenance of electric circuits and equipment in all types of flying vehicles.

For example, it may be necessary to replace a damaged section of electric harness. If the section of harness is connected to other sections by means of connectors, it is a simple matter to unplug the damaged section at both ends and remove it.

A completely new section may then be quickly installed. If the damaged section was instead connected to other sections by terminal strips, it would be necessary to disconnect each wire from the terminal studs separately; this operation would consume considerable time, especially if the harness contained many wires.

Connectors are also used to connect electric and electronic assemblies, or line replaceable units (LRUs), such as voltage regulators, flight computers, inverters, and radio equipment. When it is necessary to replace such an assembly, the connector makes it possible to disconnect the unit quickly and to reconnect the new unit with no danger of connecting any of the leads incorrectly. (***Figure 11-22***)

The proper choice and application of connectors is a significant part of the aircraft wiring system. Connectors must be kept to a minimum, selected, and installed to provide the maximum degree of safety and reliability to the aircraft. For the installation of any particular connector assembly, the specification of the manufacturer or the appropriate governing agency must be followed.

TYPES OF CONNECTORS

There is a multitude of types of connectors. The connector types that use crimped contacts are generally used on aircraft. Some common types are the round cannon type, the rectangular, and the module blocks. Environmentally resistant connectors should be used in applications subject to fluids, vibration, heat, mechanical shock, and/or corrosive elements.

When HIRF/lightning protection is required, special attention should be given to the terminations of individual or overall shields. The number and complexity of wiring systems have resulted in an increased use of electrical connectors.



Figure 11-21. Coaxial cable connectors.

Environment resistant connectors are used in applications where they are probably subjected to fluids, vibration, heat, mechanical shock, corrosive elements, etc. Firewall class connectors incorporating these same features should, in addition, be able to prevent the penetration of the fire through the aircraft firewall connector opening and continue to function without failure for a specified period of time when exposed to fire. Hermetic connectors provide a pressure seal for maintaining pressurized areas. When EMI/RFI protection is required, special attention should be given to the termination of individual and overall shields. Backshell adapters designed for shield termination, connectors with conductive finishes, and EMI grounding fingers are available for this purpose. (***Figure 11-23***)

Rectangular connectors are typically used in applications where a very large number of circuits are accommodated in a single mated pair . They are available with a great variety of contacts, which can include a mix of standard,



Figure 11-22. Electrical connectors.



Figure 11-24. Rectangular connectors.

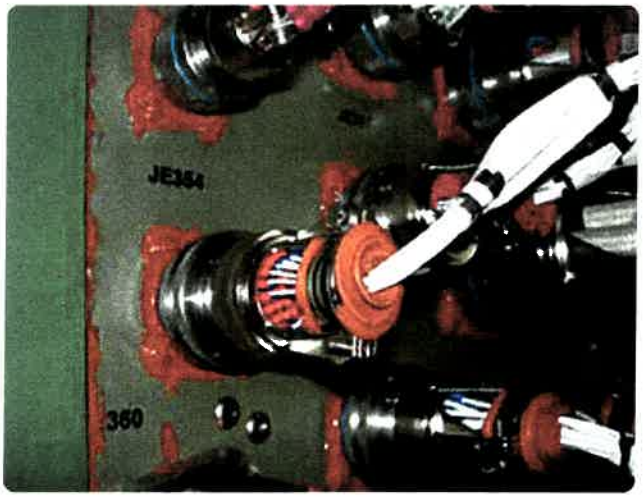


Figure 11-23. Backshells with strain relief.

coaxial, and large power types. Coupling is accomplished by various means. Smaller types are secured with screws which hold their flanges together. (***Figure 11-24***)

Larger ones have integral guide pins that ensure correct alignment, or jack screws that both align and lock the connectors. Rack and panel connectors use integral or rack-mounted pins for alignment and box mounting hardware for couplings.

Module blocks are types of junctions that accept crimped contacts similar to those on connectors. Some use internal busing to provide a variety of circuit arrangements. They are useful where a number of wires are connected for power or signal distribution. When used as grounding modules, they save and reduce hardware installation on the aircraft.

Standardized modules are available with wire end grommet seals for environmental applications and are track mounted. Function module blocks are used to provide an easily wired package for environment resistant mounting of small resistors, diodes, filters, and suppression networks. Connectors must be identified by an original identification number derived from MIL Specification (MS) or OEM specification. ***Figure 11-25*** provides information about MS style connectors.

IN-LINE JUNCTIONS

In-line terminal junctions are sometimes used in lieu of a connector when only a few wires are terminated and when the ability to disconnect the wires is desired. The inline terminal junction is environment resistant. The terminal junction splice is small and may be tied to the surface of a wire bundle when approved by the OEM.

Pins and sockets are typically identified by numbers or letters. It is usual in many types of connectors to signify this sequencing by a spiraling guideline embossed on the faces of the inserts. (***Figure 11-26***)

CONNECTOR IDENTIFICATION CODES

Figure 11-27 below explains the code used to describe an MS connector. The MS number is the basic configuration of the connector.

MS 3100 – Wall Receptacle
MS 3101 – Cable Receptacle
MS 3101 – Box Receptacle
MS 3106 - Straight Plug
MS 3107 - Quick-Disconnect Plug
MS 3108 - Angle Plug



MS27480 E 10 A 6 P B

MIL SPECIFICATION
CLASS
SHELL SIZE
POLARIZATION
CONTACT STYLE
INSERT ARRANGEMENT
FINISH

MS27472	Wall mount receptacle
MS27473	Straight plug
MS27474	Jam nut receptacle
MS27475	Hermetic wall mount receptacle
MS27476	Hermetic box mount receptacle
MS27477	Hermetic jam nut receptacle
MS27478	Hermetic solder mount receptacle
MS27479	Wall mount receptacle (note 1)
MS27480	Straight plug(note 1)
MS27481	Jam nut receptacle (note 1)
MS27482	Hermetic wall mount receptacle (note 1)
MS27483	Hermetic jam nut receptacle (note 1)
MS27484	Straight plug, EMI grounding
MS27497	Wall receptacle, back panel mounting
MS27499	Box mounting receptacle
MS27500	90° Plug (note 1)
MS27503	Hermetic solder mount receptacle (note 1)
MS27504	Box mount receptacle (note 1)
MS27508	Box mount receptacle, back panel mounting
MS27513	Box mount receptacle, long grommet
MS27664	Wall mount receptacle, back panel mounting (note 1)
MS27667	Thru-bulkhead receptacle

NOTE

1. Active	Supersedes
MS27472	MS27479
MS27473	MS27480
MS27474	MS27481
MS27475	MS27482
MS27477	MS27483
MS27473 with MS27507 elbow	MS27500
MS27478	MS27503
MS27499	MS27504
MS27497	MS27664

CLASS
- E Environment-resisting box and thru-bulkhead mounting types only (see class T)
- P Potting—includes potting form and short rear grommet
- T Environment-resisting wall and jam-nut mounting receptacle and plug types: thread and teeth for accessory attachment
- Y Hermetically sealed

FINISH
- A Silver to light iridescent yellow color cadmium plate over nickel (conductive) –65 °C to +150 °C (inactive for new design)
- B Olive drab cadmium plate over suitable underplate (conductive), –65 °C to 175 °C
- C Anodic (nonconductive), –65 °C to + 175 °C
- D Fused tin, carbon steel(conductive), –65 °C to +150 °C
- E Corrosion resistant steel (cres), passivated (conductive), –65 °C to +200 °C
- F Electroless nickel coating (conductive), –65 °C to +200 °C
- N Hermetic seal or environment resisting cres (conductive plating), –65 °C to +200 °C

CONTACT STYLE
- A Without pin contacts
- B Without socket contacts
- C Feed through
- P Pin contact—including hermetics with solder cups
- S Socket contacts—including hermetics with solder cups
- X Pin contacts with eyelet (hermetic)
- Z Socket contacts with eyelet (hermetic)

POLARIZATION
- A, B Normal—no letter required
- C, or D

Figure 11-25. MS connector information sheet.

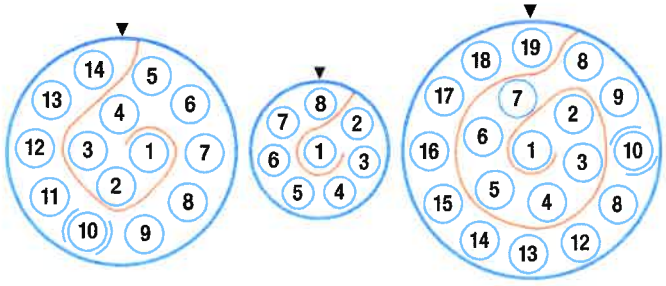


Figure 11-26. In-line terminal junction pin patterns.



Standard

Type (Straight Plug)

Class

Size (in 1/16")

Contact Arrangement Number

Contact Style (Socket)

Insert Rotation

Figure 11-27. AN connector markings.

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The letter following the configuration tells the class of connector it is.

A = General purpose, solid aluminum alloy shell.
B = General purpose, split aluminum alloy shell.
C = Pressurized, solid aluminum alloy shell.
D = Environmental resistant, solid aluminum alloy shell.
E = Fire and flame-proof, solid steel shell.

- The size of the connector is indicated next with the higher the number, the larger the connector.
- The contact arrangement identifies the number and size of the connectors and their physical arrangement. You must use a chart to identify each arrangement by the code number.
- The contact style may be either an S or P to indicate a socket or pin.
- The final letter indicates the rotation of the insert in the connector.

If a component has two or more connectors of the same type, it is possible to connect the wrong plug to a receptacle. To prevent this, the inserts may be rotated in their relationship to the index slot so that only the correct plug may be inserted into the receptacle.

CURRENT AND VOLTAGE RATING

Selected connectors must be rated for continuous operation under the maximum combination of ambient temperature and circuit current load. Hermetic connectors and connectors used in circuit applications involving high inrush currents should be derated. It is good engineering practice to conduct preliminary testing in any situation where the connector is to operate with most or all of its contacts at maximum rated current load.

When wiring is operating with a high conductor temperature near its rated temperature, connector contact sizes should be suitably rated for the circuit load. This may require an increase in wire size. Voltage derating is required when connectors are used at high altitude in non pressurized areas.

SPARE CONTACTS FOR FUTURE WIRING

To accommodate future wiring additions, spare contacts are normally provided. Locating the unwired contacts along the outer part of the connector facilitates future access. A good practice is to provide two spares on connectors with 25 or fewer contacts; 4 spares on connectors with 26 to 100 contacts; and 6 spares on connectors with more than 100 contacts.

Spare contacts are not normally provided on receptacles of components that are unlikely to have added wiring. Connectors must have all available contact cavities filled with wired or unwired contacts. Unwired contacts should be provided with a plastic grommet sealing plug.

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Question: 11-1
A _____ is described as a single, solid conductor, or as a stranded conductor covered with an insulating material.

Question: 11-2
Name 4 factors to be considered in selecting the size of the wire for transmitting and distributing electrical power.

Question: 11-3
The _____ that causes the rated maximum operating temperature of the wire to be reached should not be exceeded.

Question: 11-4
The _____ in the main power wires from the generation source or the battery to the bus should not exceed 2 percent of the regulated voltage when the generator is carrying rated current or the battery is being discharged at the 5-minute rate.

Question: 11-5
What is meant by a spiral line drawn on the inside of a connecting socket?

Question: 11-6
If a connector is designed with additional fire protection abilities, how is this identified?

Question: 11-7
What is the typical usage of a high tension cable?

Question: 11-8
What is the primary advantage of a coaxial cable?

ANSWERS

Answer: 11-1
wire.

Answer: 11-2
Sufficient mechanical strength.
Allowable power loss.
Voltage variation.
Wire temperature.

Answer: 11-3
current.

Answer: 11-4
voltage drop.

Answer: 11-5
It illustrates the sequence of pin numbers within the socket.

Answer: 11-6
By the connector's Class rating; in this case it would have an "E" rating.

Answer: 11-7
A high tension cable is used to carry extremely high voltages such as in an engine's ignition system.

Answer: 11-8
It prevents leakage of electromagnetic energy, both going into the wire, or emanating from the wire.

A&P	/	Airframe and Powerplant
AC	/	Advisory Circular
AC	/	Alternate Current
AD	/	Airworthiness Directives
AISI	/	American Iron and Steel Institute
AMC	/	Acceptable Means of Compliance
AMS	/	Aerospace Material Specification
AN	/	Air Force-Navy
AND	/	Air Force-Navy Design
ANSI	/	American National Standards Institute
APU	/	Auxiliary Power Unit
AS	/	Aeronautical Standard
AS	/	Australia Standards
ASA	/	American Standards Association
ASP	/	Adjustable Sustain Preload
ASTM	/	American Society for Testing and Materials
AWG	/	American Wire Gauge
BMI	/	Bismaleimides
BSI	/	British Standards Institute
CAGE	/	Commercial and Government Entity
CRES	/	Corrosion Resistant Steel
CFR	/	Code of Federal Regulations
DC	/	Direct Current
DVD	/	Double Vacuum Debulk Method
EASA	/	European Aviation Safety Agency
EMI	/	Electromagnetic Interference
EPA	/	Environmental Protection Agency
EPDM	/	Ethylene Propylene Diene Rubber
FAA	/	Federal Aviation Administration
FSDO	/	Flight Standards District Office
GM	/	Guidance Material
GPO	/	Government Printing Office
HIRF	/	High Intensity Radiated Fields
hPa	/	HectoPascals
HSS	/	High-speed Steel
HVLP	/	High-volume, Low-pressure
IA	/	Inspection Authorization
ID	/	Inner Diameter
ISO	/	International Organization for Standardization
kPa	/	KiloPascals
LED	/	Light Emitting Diode
LSA	/	Light Sport Aircraft
MHz	/	Megahertz
MIL-SPEC	/	Military Specification
MS	/	Military Standard
MSDS	/	Material Safety Data Sheets

NAF	/	Naval Aircraft Factory
NAS	/	National Aerospace Standards
NC	/	American National Coarse
NC	/	Numerical Controlled
NDE	/	Nondestructive Evaluation
NDI	/	Nondestructive Inspection
NF	/	American National Fine
Ni	/	Nickel
OEM	/	Original Equipment Manufacturer
Pa	/	Pascals
PBI	/	Polybenzimidazoles
PEEK	/	Polyether Ether Ketone
PMA	/	Parts Manufacturer Approval
psi	/	Pounds per Square Inch
PTFE	/	Polytetrafluoroethylene
PVC	/	Polyvinyl Chloride
RFI	/	Radio Frequency Interference
RPM	/	Revolutions per Minute
SAE	/	Society of Automotive Engineers
SRM	/	Structural Repair Manual
STC	/	Supplemental Type Certificate
SWAP	/	Severe Wind and Moisture Problem
TC	/	Thermocouple
TCDS	/	Type Certificate Data Sheet
TSO	/	Technical Standard Order
UNC	/	American Standard Unified Coarse
UNF	/	American Standard Unified Fine
UV	/	Ultraviolet
VNE	/	Never Exceed Speed
WWI	/	World War I
WWII	/	World War II



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