MALLA REDDY COLLEGE OF ENGINEERING AND TECHNOLOGY

AIRFRAME STRUCTURAL DESIGN

IV YEAR I SEMESTER

DEPARTMENT OF AERONAUTICAL ENGINEERING

MALLA REDDY COLLEGE OF ENGINEERING & TECHNOLOGY

	L	T/P/D	C
IV Year B. Tech, ANE-I Sem	**4**	**-/-/-**	**3**

(R17A2120) AIRFRAME STRUCTURAL DESIGN

Objectives:

- To know about detailed structural components present in aircraft
- To acquire the knowledge about the design parameters how why and where they will be used in manufacturing
- Students can acquire the knowledge about the loading conditions done on the structure

UNIT I
INTRODUCTION
Principal structural components of aircraft. Design requirements- structural integrity, stiffness, service life. Baseline aerodynamic configuration, external loading, weight, operating conditions, conformity to government regulations. Design procedure- structural lay out, structural modeling, design criteria, load estimation, stress analysis, choice of materials, sizing- estimation of strength, stiffness, mass .optimization, trade-off. Structural index- use in design. Idealization of structures, materials- constitutive relations- equilibrium, compatibility conditions significance. Sizing of structural elements of given geometry and loading Analysis of box beams- single cell, multi cell- in bending, shear, torsion- normal stresses, shear flow, deformation- restraint against warping, secondary stresses.

Unit II
FASTENERS AND STRUCTURAL JOINTS
Fasteners and fittings- role , significance, general design considerations, criteria for allowable strength. Margine of safety. Fastener systems, types, fastener information, dimensions, materials, allowable strength- tensile, shear, bending. Rivets, bolts and screws, nuts-detail design consideration.Fastener selection.fittings- lugs, bushings and bearings-loading design and analysis. Joints – splices, eccentric, gusset, welded, brazed, bonded- types, methods of joining, failure modes. Fatigue design considerations. Stress concentration- causes, methods of reduction. Fastener load distribution and by pass load-severity factor, structural joint life prediction. Shim control and requirement

UNIT III
DESIGN OF WINGAND TAIL STRUCTURES
The wing- role- summary of wing loads, structural components- wing box, leading and trailing edges. Wing layout- location of spars, ailerons and flaps, rib spacing and direction, root rib bulkhead, span wise stiffeners, wing covers- skin-stringer panels, integrally stiffened panels, access holes, attachment of leading edge and trailing edge panels Spars- general rules of spar design. Ribs and bulkheads- rib spacing and arrangement .Wing root joints, carry through structure. Fighter wing design- problems with swept wings Wing box- loads, stress .Wing box,root bulkhead-estimation of loads, stress analysis, design parameters, optimization, sizing, margin of safety.. Leading and trailing edge assembly- control surfaces, flaps- structure. Tail unit- horizontal, vertical tail, elevator, rudder- configuration, structural layout, design considerations.

UNIT IV
DESIGN OF FUSELAGE AND LANDING GEAR
Function of fuselage- loading, general requirements. Ultimate strength of stiffened cylindrical structure. Principal structural components –skin and stringers, frame and floor beam, pressure bulkheads, wing & fuselage intersection- layout, stress analysis, sizing. Forward fuselage, aft, fuselage structures, fuselage openings- windows, doors- design considerations. Landing gear- purpose, types, general arrangement, loads-

design considerations- ground handling, take-off, landing, braking, pavement loading, support structure. stowage and retraction, gear lock- kinematic design Shock absorbers- function, types, components, operation, loads, materials, design. Wheels and brakes, tire selection .

UNIT V

FATIGUE LIFE, DAMAGE TOLERANCE, FAIL SAFE- SAFE DESIGN-WEIGHT CONTROL AND BALANCE

Catastrophic effects of fatigue failure- examples- modes of failure- design criteria- fatigue stress, fatigue performance, fatigue life. Fatigue design philosophy- fail-safe, safe life. Service behaviour of aircraft structures- effect of physical and load environment design and of detail of fabrication Structural life- methods of estimation- the scatter factor- significanceFail-safe design- the concept, requirements, damage tolerance-estimation of fatigue strength

Text Books:

1. NIU.M.C. Airframe Structural Design, second edition, HongkongConmlit Press, 1988, ISBN: 962-7128-09-0
2. NIU.M.C. Airframe Stress Analysis And Sizing, second edition, HongkongConmlit Press, 1987, ISBN: 962-7128-08-2

Out comes:

- Students will be acquainted with design criteria of aircraft component
- Students will be acquainted with manufacturing procedure from the design criteria
- Students will easily design their own components based on the design criteria they have learned

UNIT-I

INTRODUCTION

INTRODUCTION

Aircraft structural design, analysis, manufacturing and validation testing tasks have become more complex, regardless of the materials used, as knowledge is gained in the flight sciences, the variety of material forms and manufacturing processes is expanded, and aircraft performance requirements are increased.

A greatly expanded design data base of applied loads is now available for more complete and thorough definition of critical design conditions, thanks to the expanding use of computational fluid dynamics (CFD), advanced wind tunnel testing techniques, and increasingly comprehensive aeroelastic and structural dynamic analysis computer codes. Similarly, computer-aided design tools make it easier and quicker to consider a much greater variety of alternative structural designs. The use of high-speed, large-memory computers permits, in turn, more detailed internal structural loads analysis for each of the many loading conditions and design alternatives, with fine grid analysis determining more precise load paths, stress distributions, and load deflection characteristics for subsequent aeroelastic analysis.

Expansion of structural synthesis, analysis, and testing capabilities and the widening options available are making the choice of materials for both the airframe and the engine one that is intrinsically woven into the structural concept, detailed part design, and manufacturing process selection. A fundamental aspect, of course, is knowledge of the physical properties of these materials. Characteristics such as static tensile strength, compression and shear strength, stiffness, fatigue resistance, fracture toughness, and resistance to corrosion or other environmental conditions, can all be important in the design.

Each of these aspects must be considered and dealt with concurrently if modern structural designs for aircraft are to approach optimum configurations and, thereby, success in international

Introduction:

- The airframe is the basic structure of an aircraft and is designed to withstand all aerodynamic forces, as well as the stresses imposed by the weight of the fuel, crew, and payload
- Although similar in concept, aircraft can be broken down into fixed and rotary wing structures
- The airplane is controllable around its lateral, longitudinal, and vertical axes by deflection of flight control surfaces
- These control devices are hinged or movable surfaces with which the pilot adjusts the airplane's attitude during takeoff, flight maneuvering, and landing
- They are operated by the pilot through connecting linkage by means of rudder pedals and a control stick or wheel

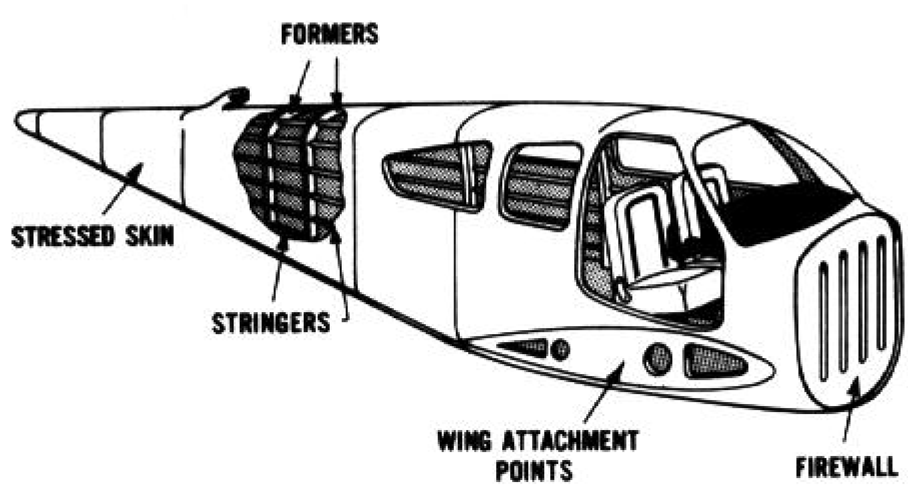
I. Principle Structure:

- **Fuselage:**
  Main structural unit

    - The fuselage is the principal structural unit of an aircraft
    - The fuselage is designed to accommodate the crew, passengers, cargo, instruments, and other essential equipment

    - **Types of Fuselage Construction:**
        - The construction of aircraft fuselages evolved from the early wood truss structural arrangements to monocoque shell structures to the current semimonocoque shell structures.

            - **Truss Structure:**
                - In this construction method, strength and rigidity are obtained by joining tubing (steel or aluminum) to produce a series of triangular shapes, called trusses
                    - Lengths of tubing, called longerons, are welded in place to form a wellbraced framework
                    - Vertical and horizontal struts are welded to the longerons and give the structure a square or rectangular shape when viewed from the end
                    - Additional struts are needed to resist stress that can come from any direction
                    - Stringers and bulkheads, or formers, are added to shape the fuselage and support the covering
                - As designs progressed these structures were enclosed, first with cloth and eventually with metals
                - These upgrades streamlined shape and increased performance
                - In some cases, the outside skin can support all or a major portion of the flight loads



Aircraft Fuselage

- Most modern aircraft use a form of this stressed skin structure known as monocoque or semimonocoque construction
  - **Monocoque:**
    - Monocoque (French for "single shell") construction uses stressed skin to support almost all loads much like an aluminum beverage can
    - In monocoque construction, rigs, formers, and bulkheads of varying sizes give shape and strength to the stressed skin fuselage [***Figure 1***]
    - Although very strong, monocoque construction is not highly tolerant to deformation of the surface
    - For example, an aluminum beverage can supports considerable forces at the ends of the can, but if the side of the can is deformed slightly while supporting a load, it collapses easily
    - Because most twisting and bending stresses are carried by the external skin rather than by an open framework, the need for internal bracing was eliminated or reduced, saving weight and maximizing space
    - One of the notable and innovative methods for using monocoque construction was employed by Jack Northrop
    - In 1918, he devised a new way to construct a monocoque fuselage used for the Lockheed S-1 Racer
    - The technique utilized two molded plywood half-shells that were glued together around wooden hoops or stringers
    - To construct the half shells, rather than gluing many strips of plywood over a form, three large sets of spruce strips were soaked with glue and laid in a semi-circular concrete mold that looked like a bathtub
    - Then, under a tightly clamped lid, a rubber balloon was inflated in the cavity to press the plywood against the mold
    - Twenty-four hours later, the smooth half-shell was ready to be joined to another to create the fuselage
    - The two halves were each less than a quarter inch thick

- Although employed in the early aviation period, monocoque construction would not reemerge for several decades due to the complexities involved
- Every day examples of monocoque construction can be found in automobile manufacturing where the unibody is considered standard in manufacturing

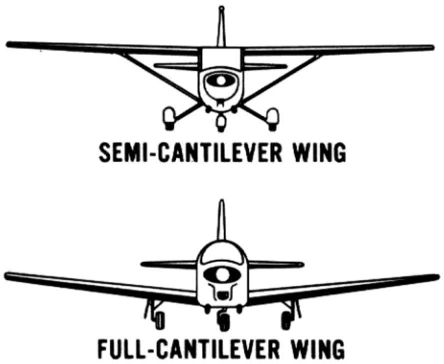
Semimonocoque:

- Semimonocoque construction, partial or one-half, uses a substructure to which the airplane's skin is attached. The substructure, which consists of bulkheads and/or formers of various sizes and stringers, reinforces the stressed skin by taking some of the bending stress from the fuselage. The main section of the fuselage also includes wing attachment points and a firewall. On single-engine airplanes, the engine is usually attached to the front of the fuselage. There is a fireproof partition between the rear of the engine and the flight deck or cabin to protect the pilot and passengers from accidental engine fires. This partition is called a firewall and is usually made of heat-resistant material such as stainless steel. However, a new emerging process of construction is the integration of composites or aircraft made entirely of composites [***Figure 2***]

Pilot Handbook of Aeronautical Knowledge, Monoplane (left) and Biplane (right)

Wings: Airfoils to produce lift

- Wings are airfoils attached to each side of the fuselage and are the main lifting surfaces that support the airplane in flight
- Wings may be attached at the top ("high-wing"), middle ("mid-wing"), or lower ("low-wing") portion of the fuselage
- The number of wings can also vary
  - Airplanes with a single set of wings are referred to as monoplanes, while those with two sets are called biplanes [**Figure 4**]



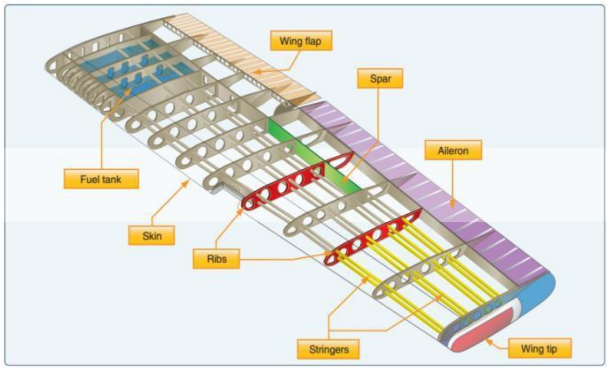


Figure 4: Wing Bracing

Figure 5:

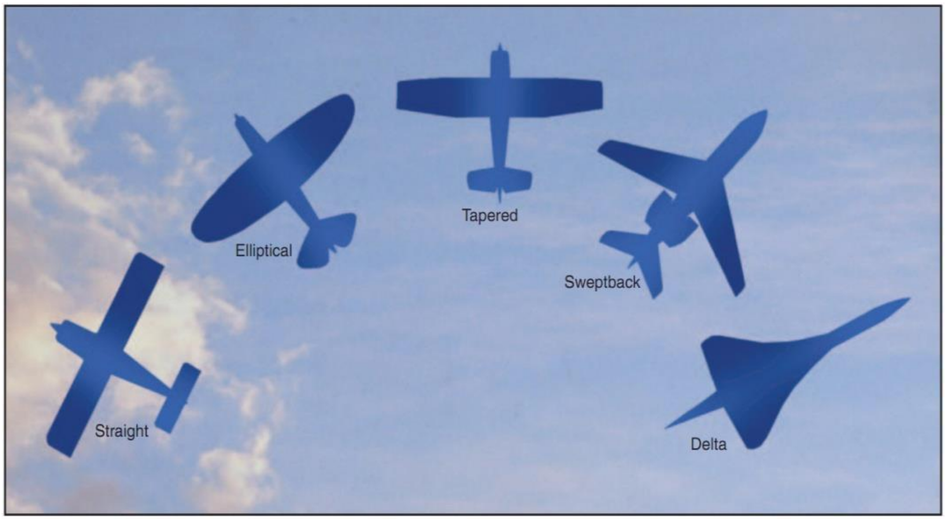
Figure:Wing Construction

- Many high-wing airplanes have external braces, or wing struts that transmit the flight and landing loads through the struts to the main fuselage structure [**Figure 5**]
- Since the wing struts are usually attached approximately halfway out on the wing, this type of wing structure is called semi-cantilever
- A few high-wing and most low-wing airplanes have a full cantilever wing designed to carry the loads without external struts
- The principal structural parts of the wing are spars, ribs, and stringers [**Figure 6**]
- These are reinforced by trusses, I-beams, tubing, or other devices, including the skin
- The wing ribs determine the shape and thickness of the wing (airfoil)
- In most modern airplanes, the fuel tanks are either an integral part of the wing's structure or consist of flexible containers mounted inside of the wing
- Attached to the rear, or trailing edges, of the wings are two types of control surfaces referred to as ailerons and flaps
- **Alternate Types of Wings:**
    - Design variations provide information on the effect controls have on lifting surfaces from traditional wings to wings that use both flexing (due to billowing) and shifting (through the change of the aircraft's CG). For example, the wing of the weight-shift control aircraft is highly swept in an effort to reduce drag and allow for the shifting of weight to provide controlled flight. [Figure 3-9] Handbooks specific to most categories of aircraft are available for the interested pilot and can be found on the Federal Aviation Administration (FAA) website at www.faa.gov
- **Ailerons:**
    - Ailerons (French for "little wing") are control surfaces on each wing which control the aircraft about its longitudinal axis allowing the aircraft to "roll" or "bank"
        - This action results in the airplane turning in the direction of the roll/bank
        - With aileron deflection, there is asymmetrical lift (rolling moment) about the longitudinal axis and drag (adverse yaw)
    - They are located on the trailing (rear) edge of each wing near the outer tips

        - They extend from about the midpoint of each wing outward toward the tip, and move in opposite directions to create aerodynamic forces that cause the airplane to roll
    - The yoke manipulates the airfoil through a system of cables and pulleys and act in an opposing manor
        - Yoke "turns" left: left aileron rises, decreasing camber and angle of attack on the right wing which creates downward lift
            - At the same time, the right aileron lowers, increasing camber and angle of attack which increases upward lift and causes the aircraft to turn left
        - Yoke "turns" right: right aileron rises decreasing camber and angle of attack on the right wing which creates downward lift
            - At the same time, the left aileron lowers, increasing camber and angle of attack on the left wing which creates upward lift and causes the aircraft to turn right
    - Although uncommon, some ailerons are configured with trim tabs which relieve pressure on the yoke on the aileron for rolling

- **Wing Planform:**
    - The shape and design of a wing is dependent upon the type of operation for which an aircraft is intended and is tailored to specific types of flying: [**Figure 7**]
        - **Rectangular:**
            - Rectangular wings are best for training aircraft, as well as low speed aircraft
            - Designed with twist to stall at the wing root first, to provide aileron control in stalls
        - **Elliptical:**
            - Elliptical wings are most efficient, but difficult to produce (spitfire)
        - **Tapered:**
            - More efficient than a rectangle wing but easier to produce than an elliptical design
        - **Swept:**
            - Usually associated with swept-back, but can also be swept-foreword
            - Sweptback wings are best for high speed aircraft for delaying Mach tendencies
            - Stall at the tips first, providing poor stall characteristics
        - **Delta:**
            - Advantages of a swept wing, with good structural efficiency and low frontal area
            - Disadvantages are the low wing loading and high wetted area needed to obtain aerodynamic stability

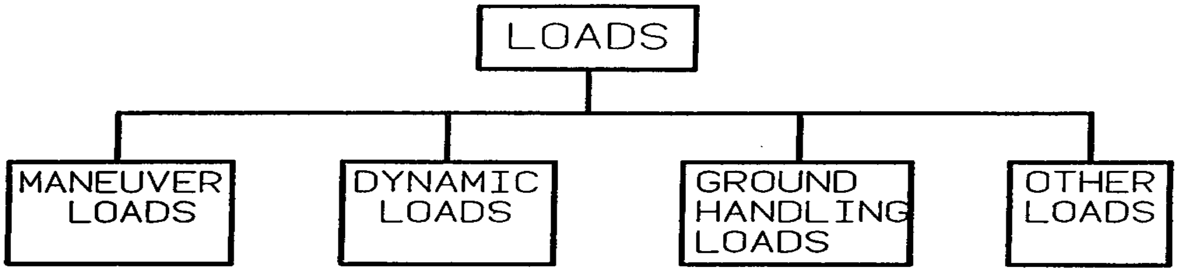
    - which provides information on the effect controls have on lifting surfaces from traditional wings to wings that use both flexing (due to billowing) and shifting (through the change of the aircraft's CG). For example, the wing of the weight-shift control aircraft is highly swept in an effort to reduce drag and allow for the shifting of weight to provide controlled flight. [Figure 3-9] Handbooks specific to most categories of aircraft are available for the interested pilot and can be found on the Federal Aviation Administration (FAA) website at www.faa.gov



Airplane Flying Handbook, Airfoil types

- **Empennage (Vertical Tail and Horizontal Tail)**

Loads can be divided into 4 groups as shown below.



Structural Design Criteria

Structural design criteria embodied within FAR 25 require the consideration of several key characteristics of the structure:

- **Gross weight requirements - loadability**
- **Performance capabilities**
- **Stiffness**
- **Aerodynamics characteristics**
- **Landing gear features and characteristics**
- **Operational altitudes**
- **Loads analysis requirements**

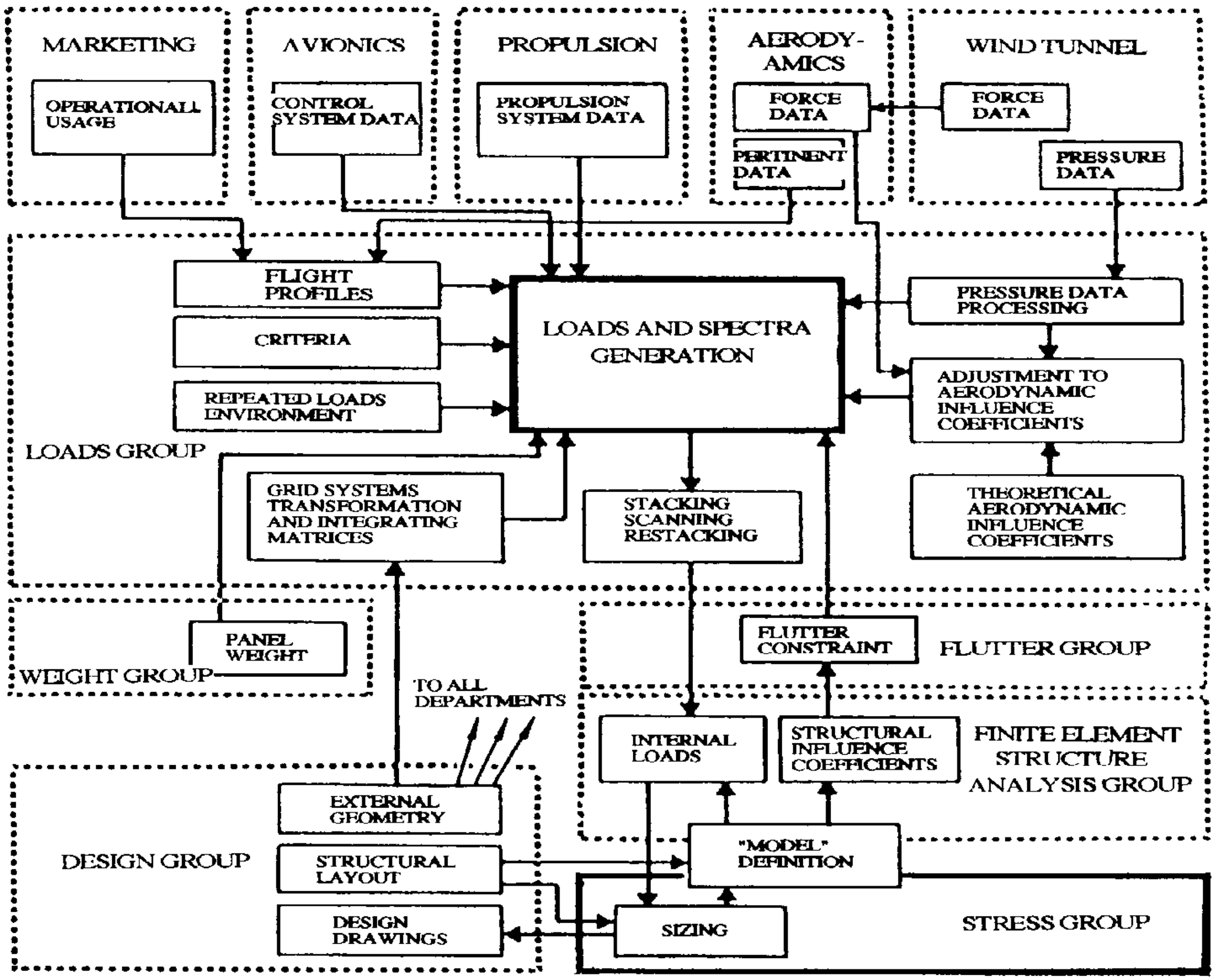
The final basic loads must be an acceptable compromise of many considerations summarized in the load organization interface chart shown below. The overall design is a continual tug-of-war and compromise between many design groups within a typical aerospace (all trying to meet different requirements).

Structural Design Criteria embody the following the following elements:

- **Structural Stiffness Requirements**

  Provide adequate strength to prevent/control

  - *Flutter*
  - *Control surface reversal* - whereby the control surface has lost its effectiveness due to weak torsional stiffness of the wing box
  - *Static divergence* - whereby the wing structure becomes torsionally unstable as the angle of attack increases due to the applied loads



- **Aerodynamic Characteristics**

  Information to be determined by analysis and/or wind tunnel tests

  - Basic aircraft stability characteristics
  - Pressure distributions over the wing, empennage and fuselage
  - Control surface hinge moments
  - Pressure distributions of high lift devices, such as flaps and slats

- **Load Analysis**

  Design loads determined for the various structural components using computer analysis. Required analyses include

  - Maneuver loads analysis
  - Gust loads analysis (static and dynamic approaches)
  - Landing loads analysis (including both rigid airframe and dynamic analyses)
  - Ground handling loads (including both rigid and elastic aircraft characteristics)
  - Control surface reversal characteristics and load distributions

- Dynamic analyses for control surface oscillatory conditions due to autopilot or yaw damper failure
- Flutter analysis and tests including both wind tunnel and flight testing to verify adequacy of damping characteristics of the aircraft throughout the speed range

This long list of load analyses requirements brings up one extremely important observation. There are many, many ***load cases*** that the structural designer must consider and design for. A typical aircraft structure will be analyzed for tens, perhaps hundreds, of loading situations!

Weight and Balance

Structural design loads affect the weight of the structure, and the weight of the aircraft influences the magnitude of design loads. This interdependence suggests that a judicious selection of the preliminary design weight is mandatory to the economical design of an aircraft (or rocket) structure.

- **Weight Requirements**
  - Center of gravity (c.g.) limits
  - Weight distribution of fixed items such as engines, fuel tanks, etc. to allow maximum use of aircraft cargo and passenger compartments
  - Fuel requirements - affects c.g. requirements and gross weight capabilities
- **Major Aircraft Weight**

  Aircraft gross weight and detailed distribution of weight both have a large influence on structural design loads.

  - <u>Take-off gross weight</u>. Considered for taxing and flight conditions.
  - <u>Design landing weight</u>. Critical for both wing and fuselage down-bending during landing.
  - <u>Zero fuel weight</u>. This condition is usually critical for wing up-bending.

- **Center of Gravity Envelope**

  A center of gravity envelope must be defined early in the aircraft design to mediate significant changes in stability and control characteristics of the aircraft.

- **Weight Distribution**

  Careful distribution of dead weight (fuselage, wings, cargo, etc.) is essential. Whether these large masses are placed about the centerbody or placed at extreme forward or aft locations will influence greatly the magnitude of down-bending experienced by the fuselage forebody or aftbody during a hard landing.

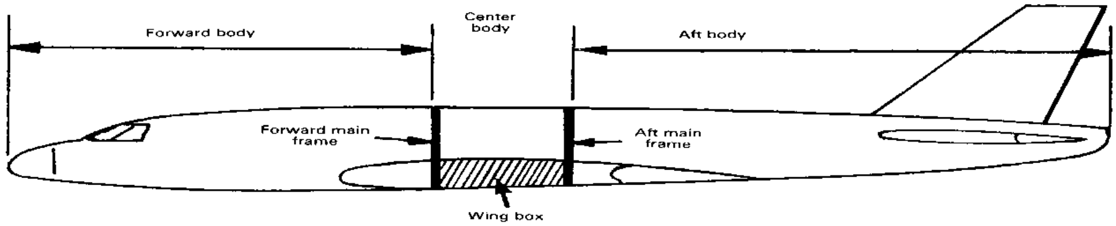
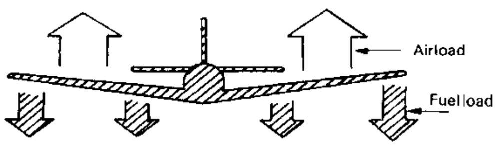
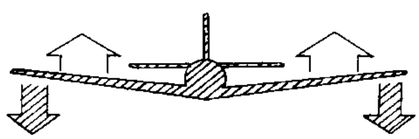
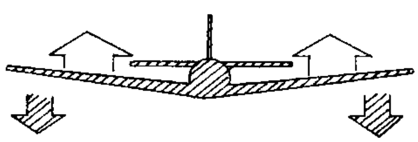


Fig. 3.7.1 Divided fuselage into three sections.

Down-bending means that the fuselage forebody and aftbody will bend downwards relative to the center body.In a similar way, the placement of fuel will greatly affect wing up-bending (or down-bending). Careful placement of fuel can provide *bending relief* during flight. Fuel usage during flight:

 ***(a) Fuel weight provides relief to wing bending.***	Fuel weight provides relief to wing loading.
 ***(b) Inboard fuel expended. Airload reduced because of reduced gross weight. Outboard fuel providing relief.***	Inboard fuel expended. Airload reduced because of reduce gross weight. Outboard fuel providing wing bending relief.
 ***(c) Outboard fuel nearly expended. Bending relief decaying faster than airload bending; therefore, net bending increasing slightly.*** *Fig. 3.4.6 Illustration of the effect of fuel weight in wing.*	Outboard fuel nearly expended. Bending relief decaying faster tanairload bending; therefore, net bending increasing slightly.

Flight Loads

Many different types of flight loads have to be considered.

- Maneuver Loads (symmetrical and unsymmetrical)
- Gust Loads (symmetrical and unsymmetrical)

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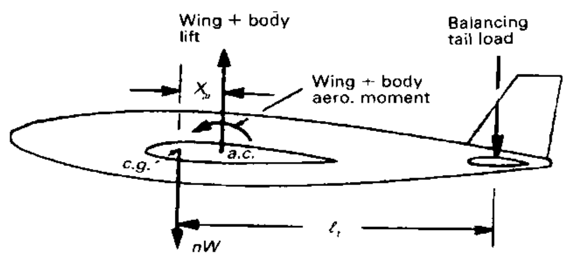

(tail load) ℓ_t = (wing + body lift) X_a + (wing + body) moment

Fig. 3.3.2 Airplane in balanced flight.
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<td>Aircraft in balanced flight</td>
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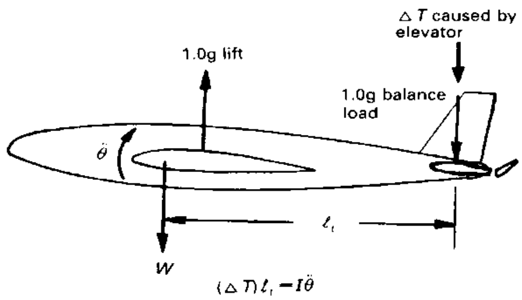

Fig. 3.3.3 Force diagram due to deflected elevator.
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<td>With deflected elevator, additional tail force causes down-bending of fuselage to increase.</td>
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Pitching maneuvers create dramatically different loading on lifting surfaces. These, in turn change the loading condition on the structure.

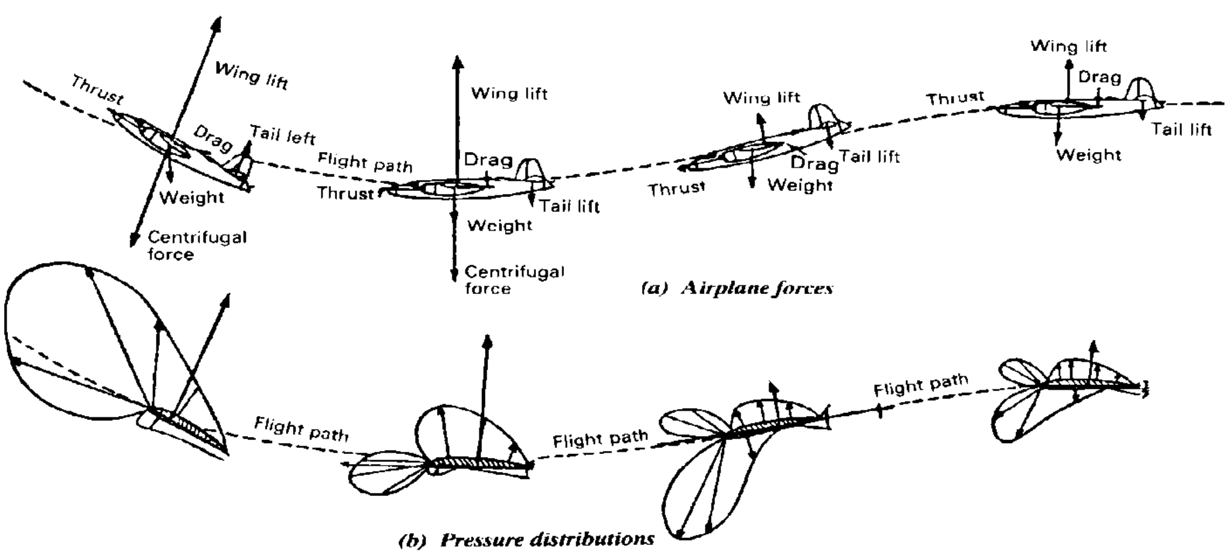
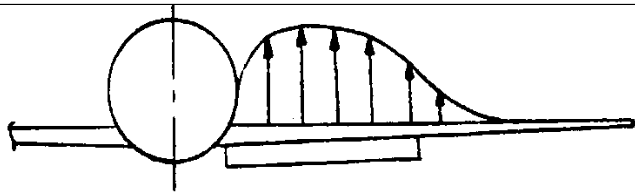
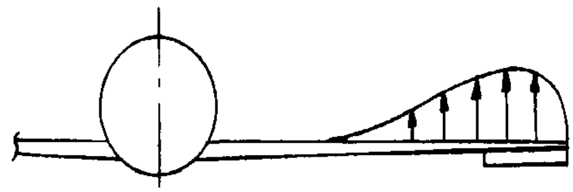


Fig. 3.3.1 Typical airplane pitching maneuver.

Roll Maneuvers due to flap and aileron deflections significantly change load distribution on the wing (two upper figures below).



(a) Load distribution from flap



Load distribution from aileron

Fig. 3.5.10 Wing load distribution due to deflected ailerons.

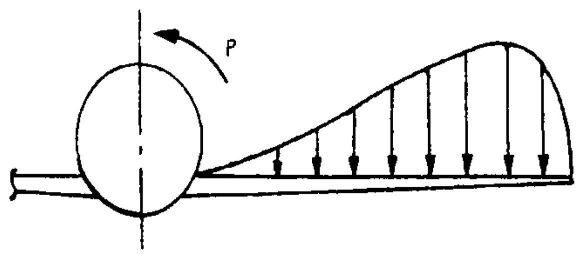


Fig. 3.5.11 Damping load distribution on wing span due to rolling.

A roll maneuver (lower right figure above) will produce a damping load distribution on the wing, i.e., because the roll maneuver (CCW above) wants to push air upwards, a downward force is produced which tends to dampen the roll. Note that this wing loading is significantly different than a typical elliptical-shaped loading due to aerodynamic lift.

A typical pressure distribution (vertical component) on the fuselage of an aircraft in level flight is shown to the right. What happens in a pitch up or down maneuver?

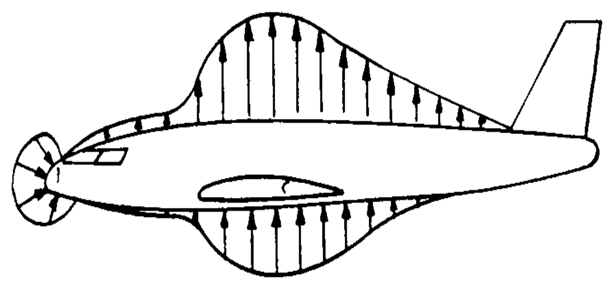


Fig. 3.7.2 Pressure distribution on fuselage.

How do changes in angle of attack change the loading that a typical wing box must carry, and where maximum tensile and compressive stresses occur?

High positive angle of attack (+HAA) will place the upper, forward part of the wing box in maximum compression and the lower, aft portion in maximum tension.

At low positive angle of attach (+LAA), the upper, aft part of the wing box will be in maximum compression, and the lower, forward portion in maximum tension.

What does all this mean? *Lots and lots of load cases for the structural designer to investigate and design for…*

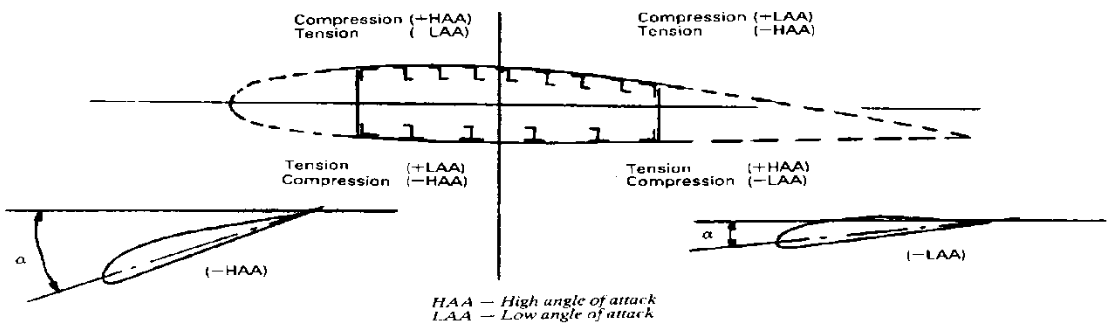


Fig. 3.5.6 Critical conditions for wing box structure.

Ground Loads

- Landing Conditions
    - Level Landing (LL)
    - Tail-Down Landing
    - One-Wheel Landing
    - Lateral-Drift Landing (LD)
    - Rebound Landing

Ground Handling

Typically we think of aerodynamic loads as being of primary importance under normal situations.

- **Landing Conditions:**

    A. **Level Landing:** Wheels are supporting the weight of the aircraft during landing impact.

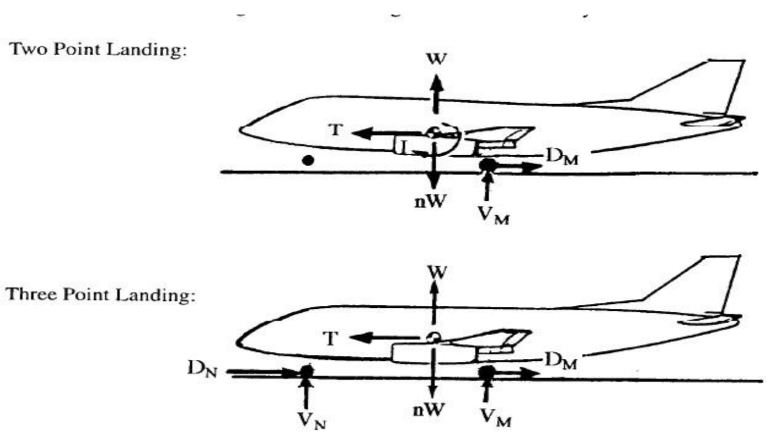
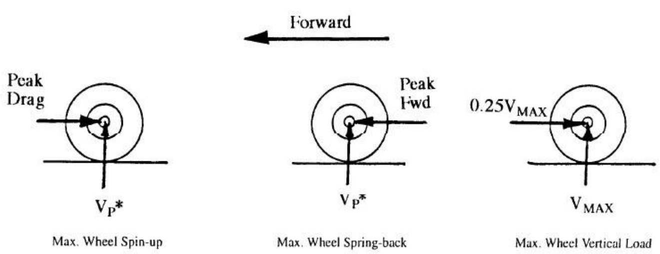
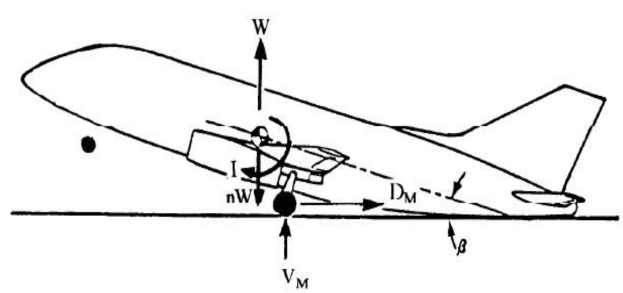


Fig. 3.5.1 Level Landing



(* Ground load at instant of peak horizontal load)

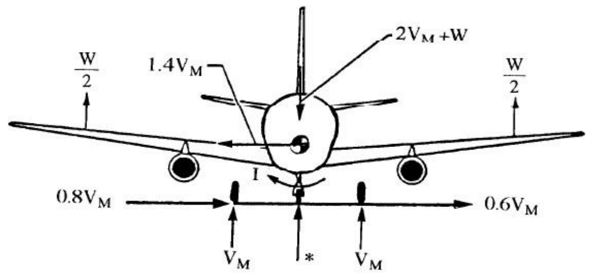
B. Tail down landing: Is done at extreme angle of attack.



(β – Angle for main gear and tail structure contacting ground unless exceeds stall angle)

C. Lateral Drift Landing:

The aircraft with level landing but only main wheel on the ground.



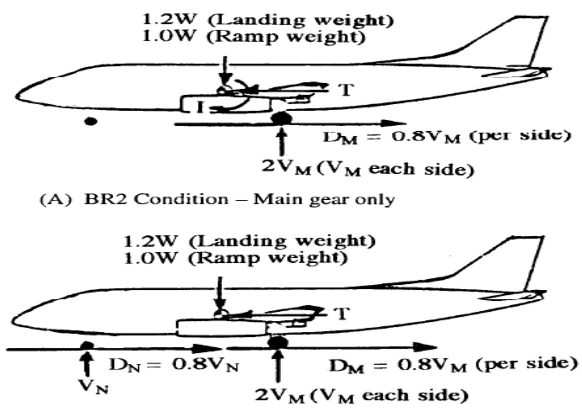
D. Rebound Landing: The landing gear and its supporting structure must be investigated for the loads occurring during rebound of the airplane from the landing surface. With the landing gear fully extended and not in contact with the ground, a load factor of 20.0 must act on the unsprung weights of the landing gear. This load factor must act in the direction of motion of the unsprung weights as they reach their limiting positions in extending with relation to the sprung parts of the landing gear.

- **Ground Handling:**

A. Taxing

B. Takeoff run

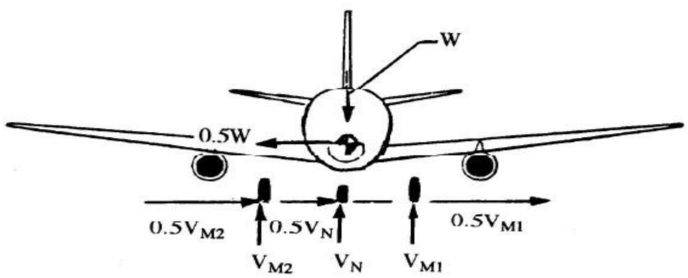
C. Brake & Roll



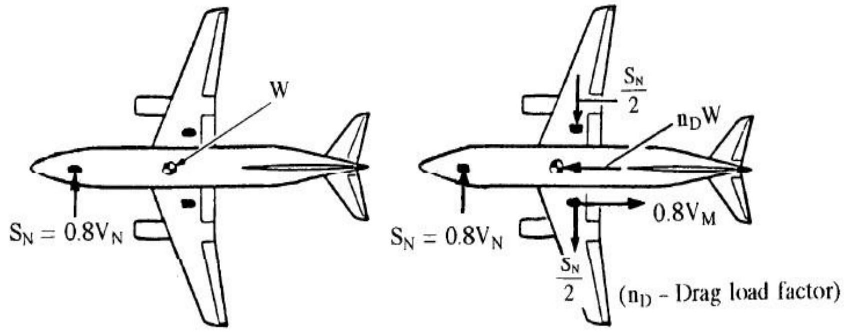
(A) BR2 Condition – Main gear only

(B) BR3 Condition – Main and nose gear

D. Ground Turning

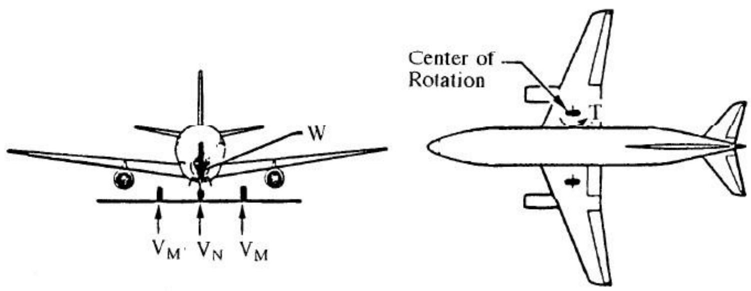


E. Nose Gear Yawing



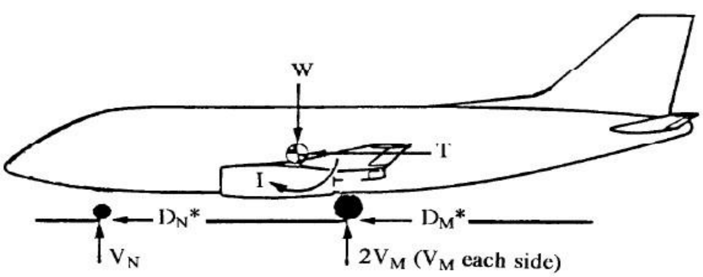
(A) Nose Gear Side Load (B) Unsymmetrical Braking

F. Pivoting



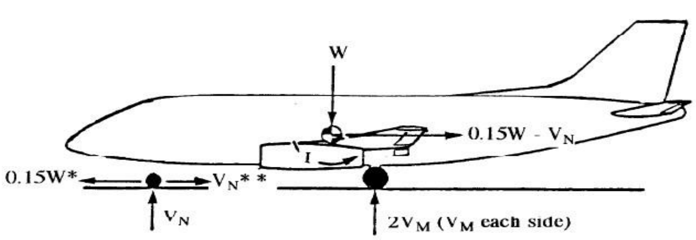
V_N (nose gear) and V_M (main gear) are static ground reactions.

G. Reverse Braking



* 1.2 load for nominal max. static brake torque (limited to 0.55V)

H. Towing



* Applied at tow fitting
** Applied at axle

- **Unsymmentrical loads on multiple wheels**

UNIT-II

FASTENERS AND STRUCTURAL JOINTS

AIRCRAFT HARDWARE

Because of the small size of most hardware items, their importance is often overlooked. The safe and efficient operation of any aircraft is greatly dependent upon correct selection and use of aircraft structural hardware and seals. Aircraft hardware is discussed in detail in the *Structural Hardware Manual*, NAVAIR 01-1A-8.

Aircraft hardware is usually identified by its specification number or trade name. Threaded fasteners and rivets are usually identified by AN (Air Force-Navy), NAS (National Aircraft Standard), and MS (Military Standard) numbers. Quick-release fasteners are usually identified by factory trade names and size designations.

AIRCRAFT STRUCTURAL HARDWARE

The term *aircraft structural hardware* refers to many items used in aircraft construction. These items include such hardware as rivets, fasteners, bolts, nuts, screws, washers, cables, guides, and common electrical system hardware.

RIVETS

The fact that there are thousands of rivets in an airframe is an indication of how important riveting is. A glance at any aircraft will show the thousands of rivets in the outer skin alone. Besides the riveted skin, 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 bracing members and other parts together.

Rivets that are satisfactory for one part of the aircraft are often unsatisfactory for another part. Therefore, it is important that you know the strength and driving properties of the various types of rivets and how to identify, drive, or install them.

Solid Rivets

Solid rivets are classified by their head shape, by the material from which they are manufactured, and by their size. Rivet head shapes and their identifying code numbers are shown in *Figure 6-1*. The prefix MS identifies hardware that conforms to written military standards. The prefix AN identifies specifications that are developed and issued under the joint authority of the Air Force and the Navy.

Rivet Identification Code

The rivet codes shown in *Figure 6-1* are sufficient to identify rivets only by head shape. To be meaningful and precisely identify a rivet, certain other information is encoded and added to the basiccode. A letter, or letters, following the head-shaped code identify the material or alloy from which the rivet was made. *Table 6-1* includes a listing of the most common of these codes. The alloy code is followed by two numbers separated by a dash. The first number is the numerator of a fraction, which specifies the shank diameter in thirty-seconds of an inch. The second number is the numerator of a fraction in sixteenths of an inch, and identifies the length of the rivet. The rivet code is shown in *Figure 6-2*.

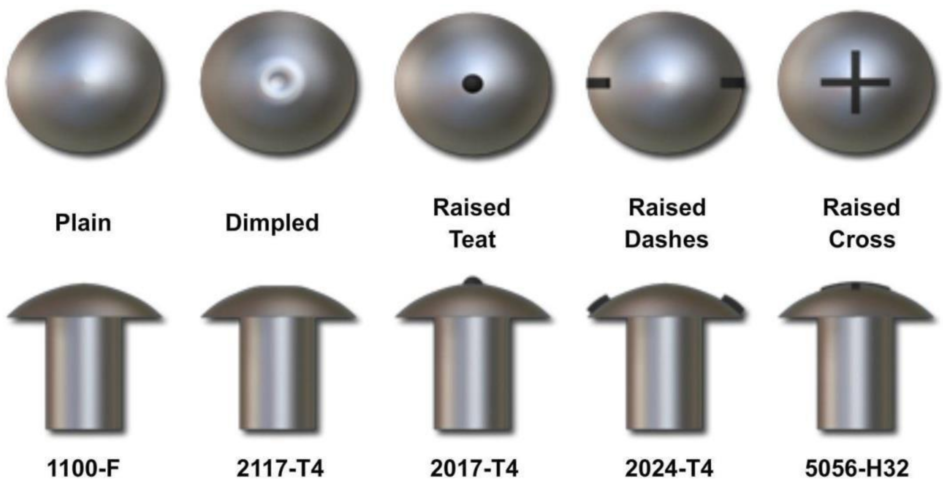
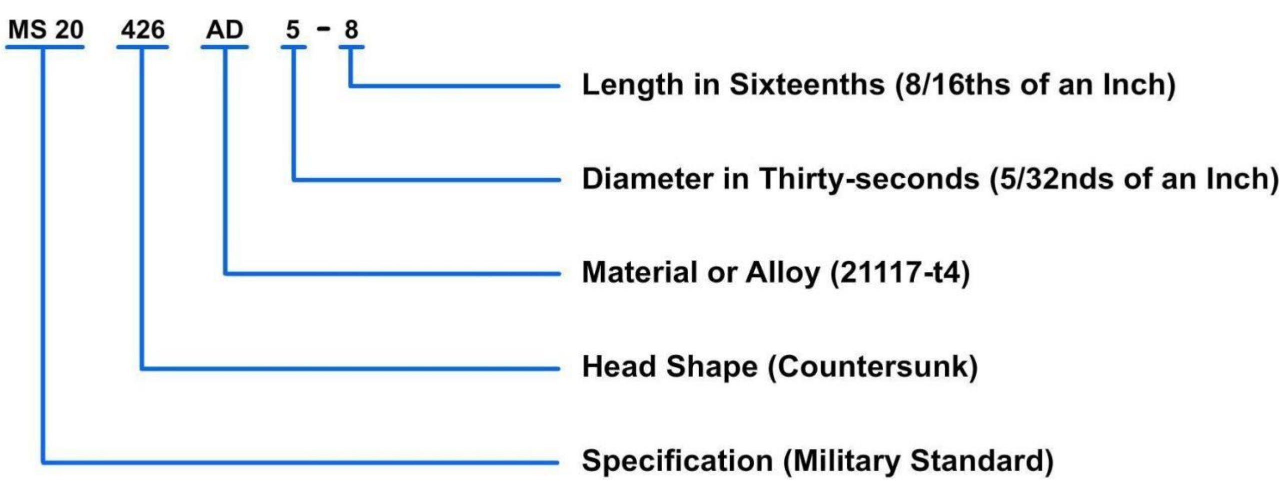


Figure 6-1 — Rivet head shapes and code numbers.



Rivet Composition

Most of the rivets used in aircraft construction are made of aluminum alloy. A few special-purpose rivets are made of mild steel, Monel, titanium, and copper. Those aluminum alloy rivets made of 1100, 2117, 2017, 2024, and 5056 are considered standard.

ALLOY 1100 RIVETS — Alloy 1100 rivets are supplied as fabricated (F) temper and are driven in this condition. No further treatment of the rivet is required before use, and the rivet's properties do not change with prolonged periods of storage. They are relatively soft and easy to drive. The cold work resulting from driving increases their strength slightly. The 1100-F rivets are used only for riveting nonstructural parts. These rivets are identified by their plain head, as shown in *Table 6-1*.

Table 6-1 — Rivet Material Identification

MATERIAL OR ALLOY	CODE LETTERS	HEAD MARKING ON RIVET
1100-F	A	Plain
2117-T4	AD	Indented Dimple
2017-T4	D	Raised Teat
2024-T4	DD	Raised Double Dash
5056-H32	B	Raised Cross

ALLOY 2117 RIVETS — Like the 1100-F rivets, these rivets need no further treatment before use and can be stored indefinitely. They are furnished in the solution-heat-treated (T4) temper, but change to the solution-heat-treated and cold-worked (T3) temper after driving. The 2117-T4 rivet is in general use throughout aircraft structures and is by far the most widely used rivet, especially in repair work. In most cases the 2117-T4 rivet may be substituted for 2017-T4 and 2024-T4 rivets for repair work by using a rivet with the next larger diameter. This is desirable since both the 2017-T4 and 2024-T4 rivets must be heat-treated before they are used or kept in cold storage. The 2117-T4 rivets are identified by a dimple in the head.

ALLOY 2017 AND 2024 RIVETS — Both these rivets are supplied in the T4 temper and must be heat-treated. These rivets must be driven within 20 minutes after quenching or refrigerated at or below 32 °F to delay the aging time 24 hours. If either time is exceeded, reheat treatment is required. These rivets may be reheated as many times as desired, provided the proper solution heat-treatment temperature is not exceeded. The 2024-T4 rivets are stronger than the 2017-T4 and, therefore, are harder to drive. The 2017-T4 rivet is identified by the raised teat on the head, while the 2024-T4 has two raised dashes on the head.

ALLOY 5056 RIVETS — These rivets are used primarily for joining magnesium alloy structures because of their corrosion-resistant qualities. They are supplied in the H32 temper (strain-hardened and then stabilized). These rivets are identified by a raised cross on the head. The 5056-H32 rivet may be stored indefinitely with no change in its driving characteristics.

BlindRivets

In places accessible from only one side or where space on one side is too restricted to properly usea bucking bar, blind rivets are usually used. Blind rivets may also be used to secure nonstructural parts to theairframe.

Self-Plugging Mechanical Lock

Figure 6-3 shows a blind rivet that uses a mechanical lock between the head of the rivet and the pull stem. This lock holds the shank firmly in place from the head side.

The self-plugging rivet is made of 5056-H14 aluminum alloy and includes the conical recess and locking collar in the rivet head. The stem is made of 2024- T36 aluminum alloy. Pull grooves that fit

into the jaws of the rivet gun are provided on the stem end that protrudes above the rivet head. The blind end portion of the stem incorporates a head and a land (the raised portion of the grooved surface)with an extruding angle that expands the rivet shank.

Applied loads for self-plugging rivets are comparable to those for solid shank rivets of the same shear strength, regardless of sheet thickness. The composite shear strength of the 5056-H14 shank and the

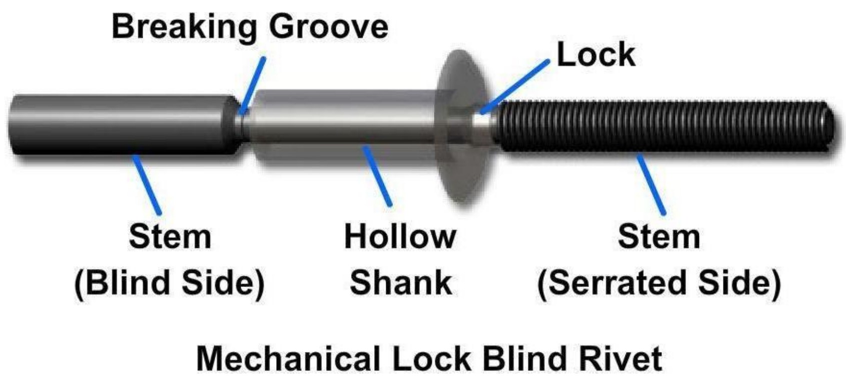


Figure 6-3 — Self-plugging rivet (mechanical lock).

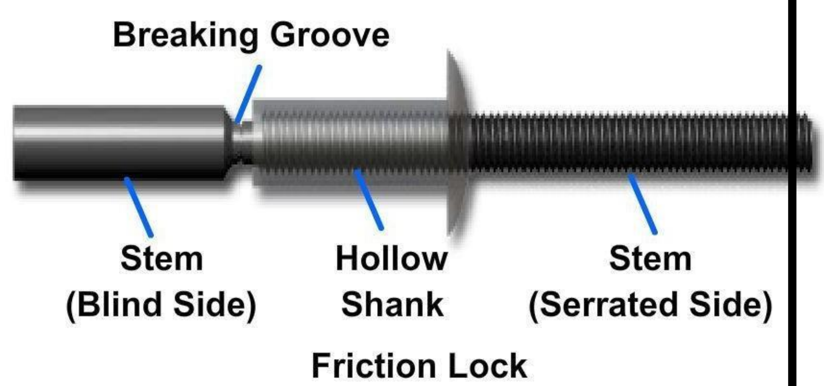
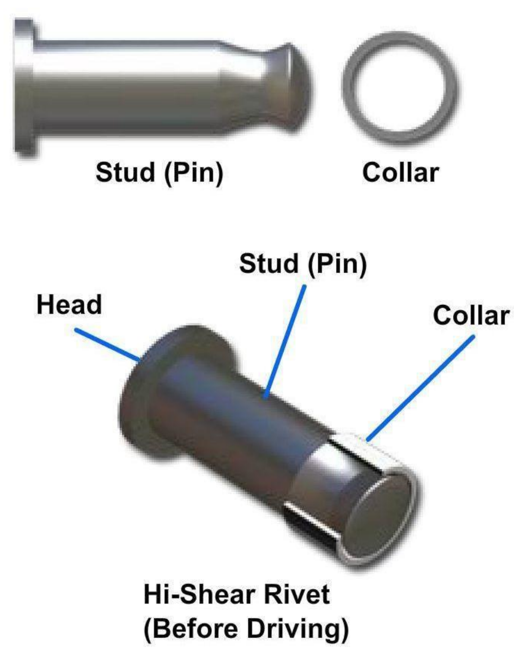


Figure 6-4 — Self-plugging rivet (friction lock).

Self-Plugging Friction Lock

Self-plugging friction lock rivets are available in universal and flush head styles and are manufactured from 2117 and 5056 aluminum alloy and Monel. Self- plugging friction lock rivets cannot be substituted for solid rivets, nor can they be used in critical applications, such as control surface hinge brackets, wing attachment fittings, landing gear fittings, and fluid-tight joints. *Figure 6-4* shows a self-plugging friction lockrivet.



Hi-Shear Rivets®

Hi-shear® (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. They are available in two headstyles—Figure 6-5 — Hi-Shear® rivet.

rivet thick-gauge sheets together. They are never used with a grip length that is less than the shank diameter. Hi-Shear® rivets

Hi-Shear® rivets are identified by code numbers similar to the solid rivets. The size of the rivet is measured in increments of thirty-seconds of an inch for the diameter and sixteenths of an inch for the grip length. For example, an NAS1055-5-7 rivet would be a Hi-Shear® rivet with a countersunk head. Its diameter would be 5/32 of an inch and its maximum grip length would be 7/16 of an inch.

The collars are identified by a basic code number and a dash number that correspond to the diameter of the rivet. An *A* before the dash number indicates an aluminum alloy collar. The NAS528-A5 collar would be used on a 5/32-inch-diameter rivet pin. Repair procedures involving the installation or replacement of Hi-Shear® rivets generally specify the collar to be used.

Rivnuts

The rivnut is a hollow rivet made of 6063 aluminum alloy, counterbored and threaded on the inside. It is manufactured in two head styles—flat and countersunk—and in two shank designs—open and closed ends. See *Figure 6-6*. Each of these rivets is available in three sizes: 6-32, 8-32, and 10-32. These numbers indicate the nominal diameter and the actual number of threads per inch of the machine screw that fits into therivnut.

Open-end rivnuts are more widely used and are generally the recommended and preferred type. However, in sealed flotation or pressurized compartments, the closed-end rivnut must be used.

FASTENERS (SPECIAL)

Fasteners on aircraft are designed for many different functions. Some are made for high-strength requirements, while others are designed for easy installation and removal.

Lock-Bolt Fasteners

Lock-bolt fasteners are designed to meet high-strength requirements. Used in many structural applications, their shear and tensile strengths equal or exceed the requirements of AN and NAS bolts.

The lock-bolt pin, shown in View *A*of *Figure 6-7*, consists of a pin and collar. It is available in two head styles: protruding and countersunk. Pin retention is accomplished by swaging the collar into the locking grooves on the pin.

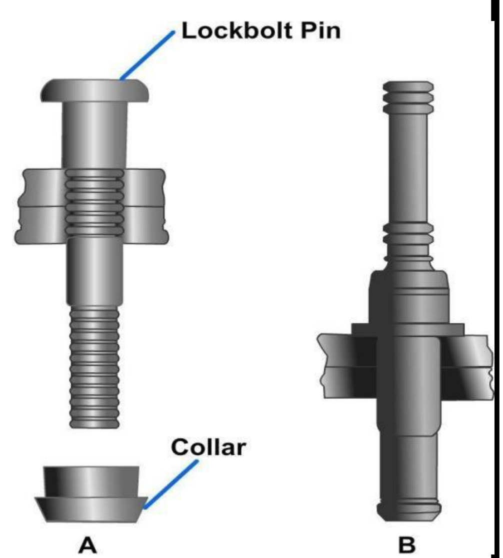


Figure 6-7 — Lock bolts.

The blind lock bolt, shown in *View B* of *Figure 6-7*, is similar to the self-plugging rivet shown in *Figure 6-3*. It features a positive mechanical lock for pin retention.

Hi-Lok® Fasteners

The Hi-Lok® fastener, shown in *Figure 6-8*, combines the features of a rivet and a bolt and is used for high-strength, interference-free fit of primary structures. The Hi-Lok® fastener consists of a threaded pin and steel. The threaded end of the pin is recessed with a hexagon socket to allow installation from one side. The major diameter of the threaded part of the pin has been truncated (cut undersize) to accommodate a 0.004-inch maximum interference-free fit. One end of the collar is internally recessed with a 1/16-inch, built-in variation that automatically provides for variable material thickness without the use of washers and without fastener preload changes. The other end of the collar has a torque-off wrenching device that controls a predetermined residual tension of preload (10%) in the fastener.

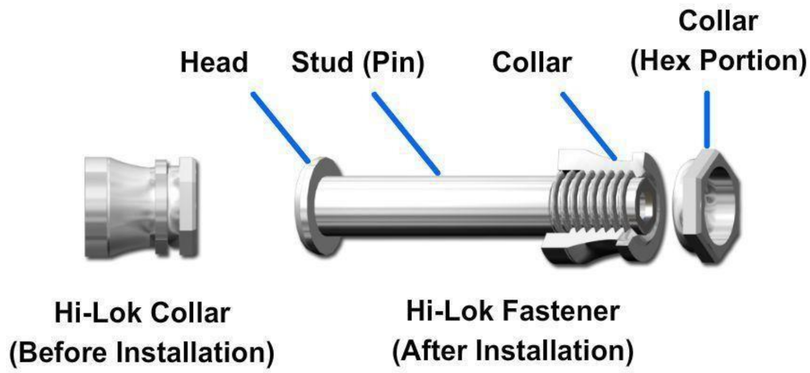


Figure 6-8 — Hi-Lok®fastener.

Jo-Bolt® Fasteners

The Jo-Bolt®, shown in *Figure 6-9*, is a high- strength, blind structural fastener that is used on difficult riveting jobs when access to one side of the work is impossible. The Jo-Bolt® consists of three factory-assembled parts: an aluminum alloy or alloy steel nut, a threaded alloy steel bolt, and a corrosion-resistant steel sleeve. The head styles available for Jo-bolts are the 100-degree flush head, the hexagon protruding head, and the 100-degree flush millable head.

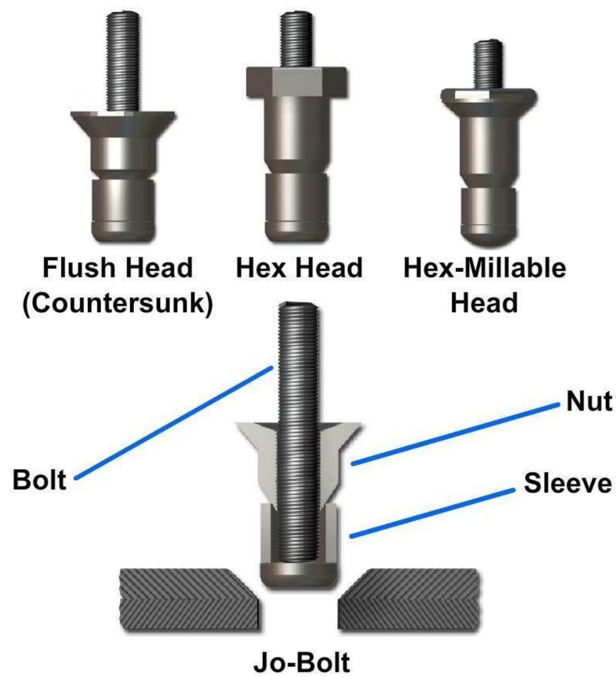


Figure 6-9 —Jo-Bolt®.

FASTENERS (THREADED)

Although thousands of rivets are used in aircraft construction, many parts require frequent dismantling or replacement. For these parts, use some form of threaded fastener. Furthermore, some joints require greater strength and rigidity than can be provided by riveting. Manufacturers solve this problem by using various types of screws, bolts, nuts, washers, and fasteners.

Bolts and screws are similar in that both have a head at one end and a screwthread at the other, but there are several differences between them. The threaded end of a bolt is always relatively blunt, while that of a screw may be either blunt or pointed. The threaded end of a bolt must be screwed into a nut, but the threaded end of the screw may fit into a nut or other female arrangement, or directly into the material being secured. A bolt has a fairly short threaded section and a comparatively long grip length (the unthreaded part); a screw may have a longer threaded section and no clearly defined grip length. A bolt assembly is generally tightened by turning its nuts. Its head may or may not be designed to be turned. A screw is always designed to be turned by its head. Another minor but frequent difference between a screw and a bolt is that a screw is usually made of lower strength materials.

Threads on aircraft bolts and screws are of the American National Standard type. This standard contains two series of threads: national coarse (NC) and national fine (NF). Most aircraft threads are of the NF series.

Threads are also produced in right-hand and left-hand types. A right-hand thread advances into engagement when turned clockwise. A left-hand thread advances into engagement when turned counterclockwise.

Threads are sized by both the diameter and the number of threads per inch. The diameter is designated by screw gauge number for sizes up to 1/4 inch, and by nominal size for those 1/4 inch and larger. Screw gauge numbers range from 0 to 12, except that numbers 7, 9, and 11 are omitted. Threads are designated by the diameter, number of threads per inch, thread series, and class in parts catalogs, on blueprints, and on repair diagrams.

For example, No. 8-32NF-3 indicates a No. 8 size thread, 32 threads per inch, national fine series, and a class 3 thread. Also, 1/4-20NC-3 indicates a 1/4-inch thread, 20 threads per inch, national coarse series, and a class 3 thread. A left-hand thread is indicated by the letters *LH* following the class of thread.

Bolts

Many types of bolts are used on aircraft. Before discussion of some of these types, it might be helpful to view a list containing information about commonly used bolt terms. Important information about the names of bolt parts and bolt dimensions that must be considered in selecting a bolt is shown in *Figure 6- 10*.

The three principal parts of a bolt are the head, thread, and grip. The head is the larger diameter of the bolt and may be one of many shapes or designs. The head keeps the bolt in place in one direction, and the nut used on the threads keeps it in placein the otherdirection.

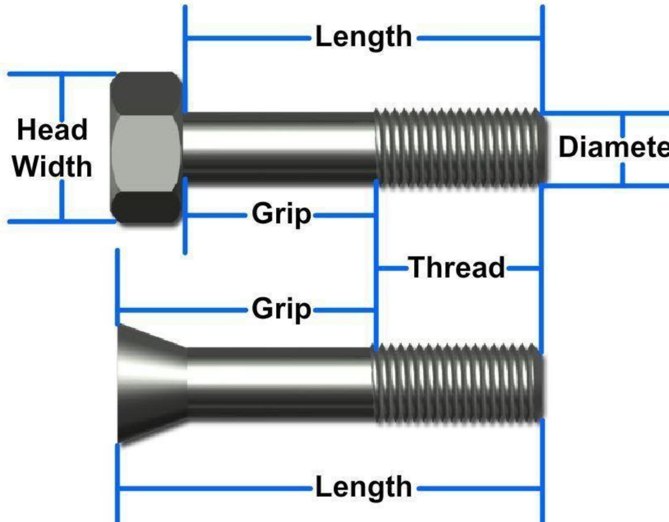


Figure 6-10 — Bolt terms and dimensions.

To choose the correct replacement, several bolt dimensions must be considered. One is the length of the bolt.. Correct length selection is indicated when the chosen bolt extends through the nut at least two full threads. In the case of flat-end bolts or chamfered (rounded) end bolts, at least the full chamfer plus one full thread should extend through the nut. See *Figure 6-10*. If the bolt is too short, it may not extend out of the bolt hole far enough for the nut to be securely fastened. If it is too long, it may extend so far that it interferes with the movement of nearby parts. Unnecessarily long bolts can affect weight and balance and reduce the aircraft payload capacity.

In addition, if a bolt is too long or too short, its grip is usually the wrong length. As shown in *Figure 6- 11*, grip length should be approximately the same as the thickness of the material to be fastened. If the grip is too short, the threads of the bolt will extend into the bolt hole and may act like a reamer when the material is vibrating. To prevent reaming, no more than two threads should extend into the bolt hole. Also, users should be certain that any threads that enter the bolt hole extend only into the thicker member that is being fastened. If the grip is too long, the nut will run out of threads before it can be tightened. In this event, a bolt with a shorter grip should be used, or if the bolt grip extends only a short distance through the hole, a washer may be used.

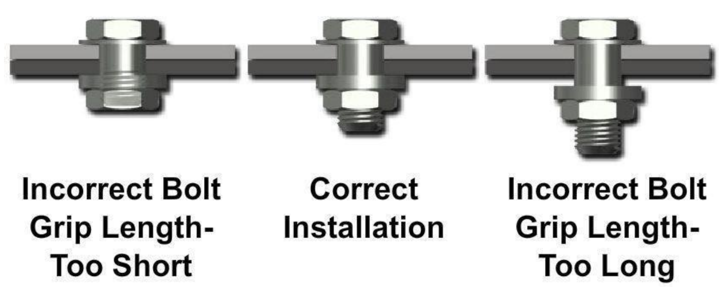


Figure 6-11 — Correct and incorrect grip lengths.

A second bolt dimension that must be considered is diameter. *Figure 6-11* shows that the diameter of the bolt is the thickness of its shaft. If this thickness is 1/4 of an inch or more, the bolt diameter is usually given in fractions of an inch (for example, 1/4, 5/16, 7/16, and 1/2). However, if the bolt is less than 1/4 of an inch thick, the diameter is usually expressed as a whole number. For instance, a bolt that is 0.190 inch in diameter is called a No. 10 bolt, while a bolt that is 0.164 inch in diameter is called a No. 8.

The results of using a bolt of the wrong diameter should be obvious. If the bolt is too big, it cannot enter the bolt hole. If the diameter is too small, the bolt has too much play in the bolt hole, and it is likely not as strong as the correct bolt.

The third and fourth bolt dimensions to consider when choosing a bolt replacement are head thickness and width. If the head is too thin or too narrow, it may not be strong enough to bear the load imposed on it. If the head is too thick or too wide, it may extend so far that it interferes with the movement of adjacentparts.

BOLT HEADS — The most common type of head is the hex head. See *Figure 6-12*. This type of head may be thick for greater strength or relatively thin in order to fit in places having limited clearances. In addition, the head may be common or drilled to lockwire the bolt. A hex-head bolt may have a single hole drilled through it between two of the sides of the hexagon and still be classed as common. The drilled head-hex bolt has three holes drilled in the head, connecting opposite sides of the hex. Seven additional types of bolt heads are shown in *Figure 6-12*.

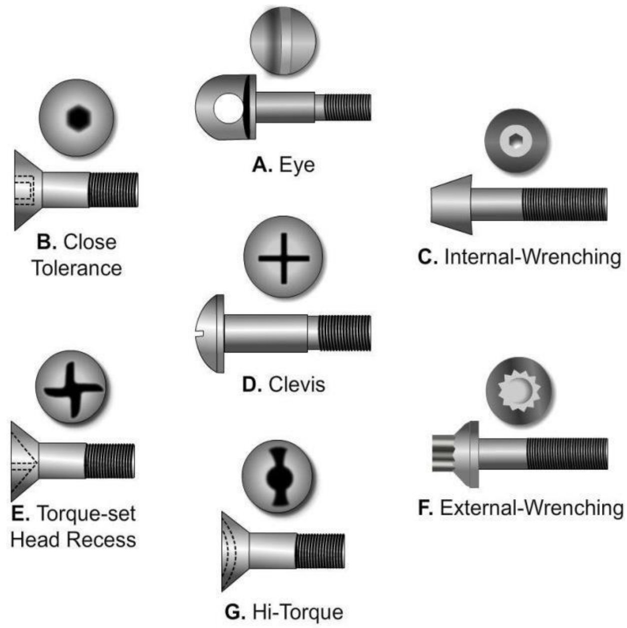


Figure 6-12 — Different types of bolts.

*View A*shows an eyebolt, often used in flight control systems.

View B shows a countersunk-head, close-tolerance bolt.

View C shows an internal-wrenching bolt. Both the countersunk-head bolt and the internal-wrenching bolt have hexagonal recesses (six-sided holes) in their heads. They are tightened and loosened by use of appropriately sized Allen wrenches.

View D shows a clevis bolt with its characteristic round head. This head may be slotted, as shown, to receive a common screwdriver or recessed to receive a Reed-and-Prince or a Phillips screwdriver.

View E shows a torque-set wrenching recess that has four driving wings, each one offset from the one opposite it. There is no taper in the walls of the recess. This permits higher torque to be applied with less of a tendency for the driver to slip or cam out of the slots.

View F shows an external-wrenching head that has a washer face under the head to provide an increased bearing surface. The 12-point head gives a greater wrench-gripping surface.

View G shows a hi-torque style driving slot. This single slot is narrower at the center than at the outer portions. This design, and the center dimple, provides the slot with a bow tie appearance. The recess is also undercut in a taper from the center to the outer ends, producing an inverted keystone shape. These bolts must be installed with a special hi-torque driver adapter. They must also be driven with some type of torque-limiting or torque-measuring device. Each diameter of bolt requires the proper size of driver for that particular bolt. The bolts are available in standard and reduced 100-degree flush heads. The reduced head requires a driver one size smaller than the standard head.

BOLT THREADS — Another structural feature in which bolts may differ is threads. These usually come in one of two types: coarse and fine. The two are not interchangeable. For any given size of bolt there are a different number of coarse and fine threads per inch. For instance, consider the 1/4-inch bolts. Some are called 1/4-28 bolts because they have 28 fine threads per inch. Others have only 20 coarse threads per inch and are called 1/4-20 bolts. To force one size of threads into another size, even though both are 1/4 of an inch, can strip the finer threads of softer metal. The same result is true concerning the other sizes of bolts; therefore, it is important to be certain that selected bolts have the correct type of threads.

BOLT MATERIALS — The type of metal used in an aircraft bolt helps to determine its strength and its resistance to corrosion. Therefore, it is important that material is considered in the selection of replacement bolts. Like solid shank rivets,bolts
have distinctive head markings that help to identify the material from which they aremanufactured.the tops of several hex-head bolts—each marked to indicate the type of bolt material.

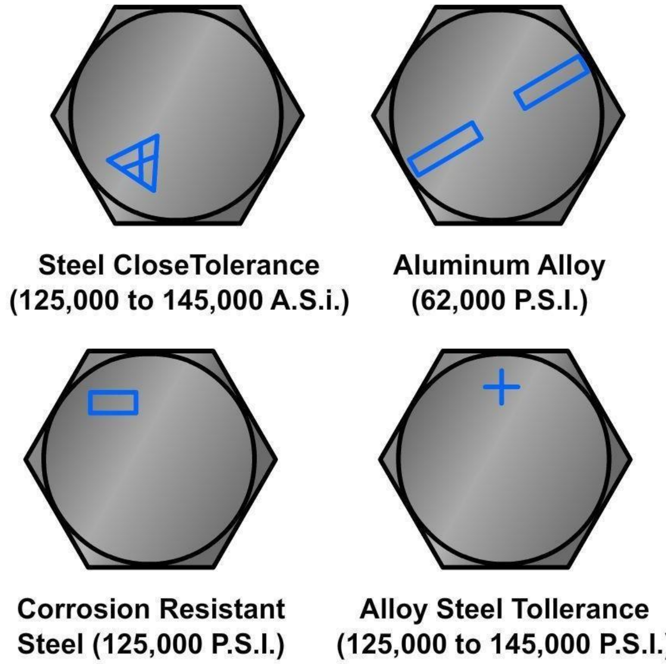


Figure 6-13 — Bolt head markings.

BOLT IDENTIFICATION — Unless current directives specify otherwise, every unserviceable bolt should be replaced with a bolt of the same type. Of course, substitute and interchangeable items are sometimes available, but the ideal fix is a bolt-for-bolt replacement. The part number of a needed bolt may be obtained by referring to the illustrated parts breakdown (IPB) for the aircraft concerned. Exactly what this part number means depends upon whether the bolt is AN, NAS, or MS.

AN Part Number — There are several classes of AN bolts, and in some instances their part numbers reveal slightly different types of information.However, most AN numbers contain the same type of information.

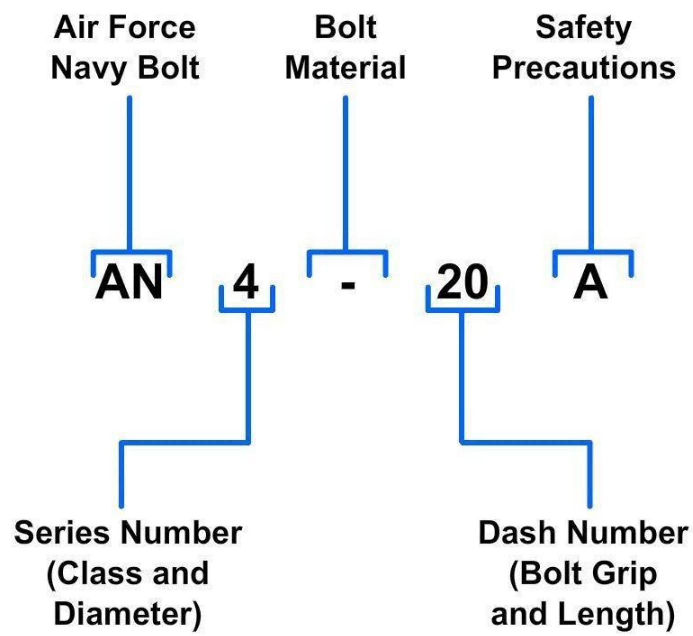


Figure 6-14 — AN bolt part number breakdown.

A breakdown of a typical AN bolt part number. Like the AN rivets discussed earlier, it starts with the letters *AN*. Next, a number follows the letters. This number usually consists of two digits. The first digit (or absence of it) shows the class of the bolt. For instance the series number has only one digit, and the absence of one digit shows that this part number represents a general-purpose hex-head bolt. However, the part numbers for some bolts of this class have two digits. In fact, general-purpose hex-head bolts include all part numbers from AN3 to AN20.Other series numbers and the classes of bolts they represent are asfollows:

- AN21 through AN36—clevis boltsAN42 through AN49—eyebolts

The series number shows another type of information other than bolt class. With a few exceptions, it indicates bolt diameter in sixteenths of an inch. For instance, in *Figure 6-14*, the last digit of the series number is 4; therefore, this bolt is 4/16 of an inch (1/4 of an inch) in diameter. In the case of a series number ending in 0 — for instance, AN30—the 0 stands for 10, and the bolt has a diameter of 10/16 of an inch (5/8 of an inch).

Refer to *Figure 6-14* again and observe that a dash follows the series number. When used in the part numbers for general-purpose AN bolts, clevis bolts, and eyebolts, this dash indicates that the bolt is made of carbon steel. With these types of bolts, the letter *C*, used in place of the dash, means corrosion-resistant steel. The letter *D* means 2017 aluminum alloy. The letters *DD* stand for 2024 aluminum alloy. For some bolts of this type, a letter *H* is used with these letters or with the dash. If it is used, the letter *H* shows that the bolt has been drilled for safetying.

Next, observe the number 20 that follows the dash. This is called the dash number. It represents the bolt's grip (as taken from special tables). In this instance the number 20 stands for a bolt that is 2 1/32 inches long.

The last character in the AN number shown in *Figure 6-14* is the letter *A*. This signifies that the bolt is not drilled for cotter pin safetying. If no letter were used after the dash number, the bolt shank would be drilled for safetying.

MS Part Number — MS is another series of bolts used in aircraft construction. In the part number shown in *Figure 6-15*, the MS indicates that the bolt is a Military Standard bolt. The series number (20004) indicates the bolt class and diameter in sixteenths of an inch (internal-wrenching, 1/4-inch diameter). The letter *H* before the dash number indicates that the bolt has a drilled head for safetying.The
dash number (9) indicates the bolt grip in sixteenths of an inch.

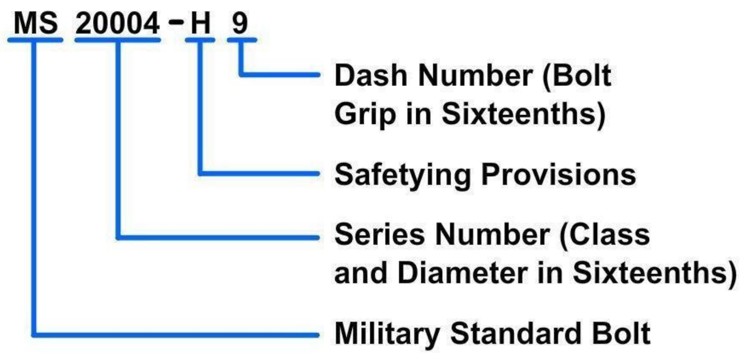


Figure 6-15 — MS bolt part number breakdown.

NAS Part Number — Another series of bolts used in aircraft construction is the NAS. See *Figure 6-16*. In considering the NAS144-25 bolt (special internal-wrenching type), the bolt identification code starts with the letters *NAS*. Next, the series has a three-digit number, 144. The first two digits (14) show the class of the bolt. The next number (4) indicates the bolt diameter in sixteenths of an inch. The dash number (25) indicates bolt grip in sixteenths of an inch.

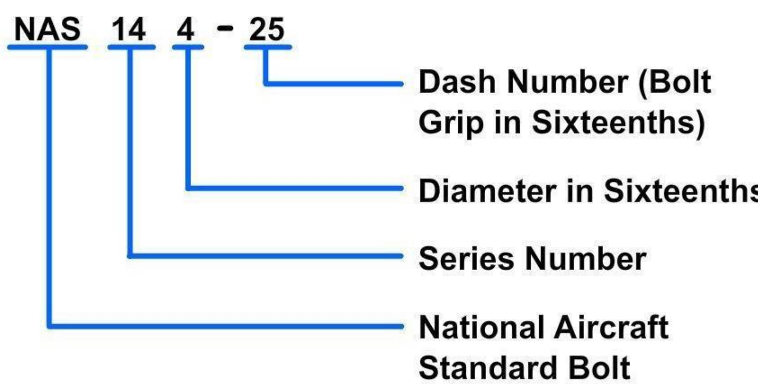


Figure 6-16 — NAS bolt part number breakdown.

Nuts

Aircraft nuts differ in design and material, just as bolts do, because they are designed to do a specific job with the bolt. For instance, some of the nuts are made of cadmium-plated carbon steel, stainless steel, brass, or aluminum alloy. The type of metal used is not identified by markings on the nuts themselves. Instead, the material must be recognized from the luster of the metal.

Nuts also differ greatly in size and shape. In spite of these many and varied differences, they all fall under one of two general groups: self-locking and nonself-locking. Nuts are further divided into types such as plain nuts, castle nuts, check nuts, plate nuts, channel nuts, barrel nuts, internal-wrenching nuts, external-wrenching nuts, shear nuts, sheet spring nuts, wing nuts, and Klincher locknuts.

NONSELF-LOCKING NUTS — Nonself-locking nuts require the use of a separate locking device for security of installation. There are several types of these locking devices mentioned in the following paragraphs in connection with the nuts on which they are used. Since no single locking device can be used with all types of nonself-locking nuts, one must be selected that is suitable for the type of nut being used.

SELF-LOCKING NUTS — Self-locking nuts provide tight connections that will not loosen under vibrations. Self-locking nuts approved for use on aircraft meet critical strength, corrosion-resistance, and temperature specifications. The two major types of self-locking nuts are prevailing torque and free spinning. The two general types of prevailing torque nuts are the all-metal nuts and the nonmetallic insert nuts. New self-locking nuts must be used each time components are installed in critical areas throughout the entire aircraft, including all flight, engine, and fuel control linkage and attachments. The flexloc nut is an example of the all- metal type. The elastic stop nut is an example of the nonmetallic insert type. All-metal self-locking nuts are constructed with the threads in the load-carrying portion of the nut out of phase with the threads in thelocking
portion, or with a saw cut top portion with a pinched-in thread. The locking action of these types depends upon the resiliency of the metal when the locking section and load-carrying section are forced into alignment when engaged by the bolt or screw threads.

Figure 6-17—Nuts.

PLAIN HEX NUTS — These nuts are available in self-locking or nonself-locking styles. When the nonself-locking nuts are used, they should be locked with an auxiliary locking device such as a check nut or lock washer. See *Figure 6-17*.

CASTLE NUTS — These nuts are used with drilled shank bolts, hex-head bolts, clevis bolts, eyebolts, and drilled-head studs. These nuts are designed to be secured with cotter pins or safety wire.

CASTELLATED NUTS — Like the castle nuts, these nuts are castellated for safetying. They are not as strong or cut as deep as the castlenuts.

CHECK NUTS — These nuts are used in locking devices for nonself-locking plain hex nuts, setscrews, and threaded rod ends.

Figure 6-18—Self-locking plate nuts.

PLATE NUTS — These nuts are used for blind mounting in inaccessible locations and for easier maintenance. They are available in a wide range of sizes and shapes. One-lug, two-lug, and right- angle shapes are available to accommodate the specific physical requirements of nut locations.
Floating nuts provide a controlled amount of nut movement to compensate for subassembly misalignment. They can be either self-locking or nonself-locking. See *Figure 6-18*.

CHANNEL NUTS — These nuts are used in applications requiring anchored nuts equally spaced around openings such as access and inspection doors and removable leading edges. Straight or curved channel nut strips offer a wide range of nut spacing and provide a multinut unit that has all the advantages of floating nuts. They are usually self-locking.

BARREL NUTS — These nuts are installed in drilled holes. The round portion of the nut fits in the drilled hole and provides a self-wrenching effect. They are usually self-locking.

INTERNAL-WRENCHING NUTS — These nuts are generally used where a nut with a high tensile strength is required or where space is limited and the use of external-wrenching nuts would not permit the use of conventional wrenches for installation and removal. This is usually where the bearing surface is counterbored. These nuts have a nonmetallic insert that provides the locking action.

POINT WRENCHING NUTS — These nuts are generally used where a nut with a high tensile strength is required. These nuts are installed with a small socket wrench. They are usually self-locking.

SHEAR NUTS — These nuts are designed for use with devices such as drilled clevis bolts and threaded taper pins that are normally subjected to shearing stress only. They are usually self-locking.

SHEET SPRING NUTS — These nuts areused with standard and sheet metal self-tapping screws to support line clamps, conduit clamps, electrical equipment, and access doors. The most common types are the float, the two-lug anchor, and the one-lug anchor. The nuts have an arched spring lock that prevents the screw from working loose.They should be used only where originally used in the fabrication of the aircraft.

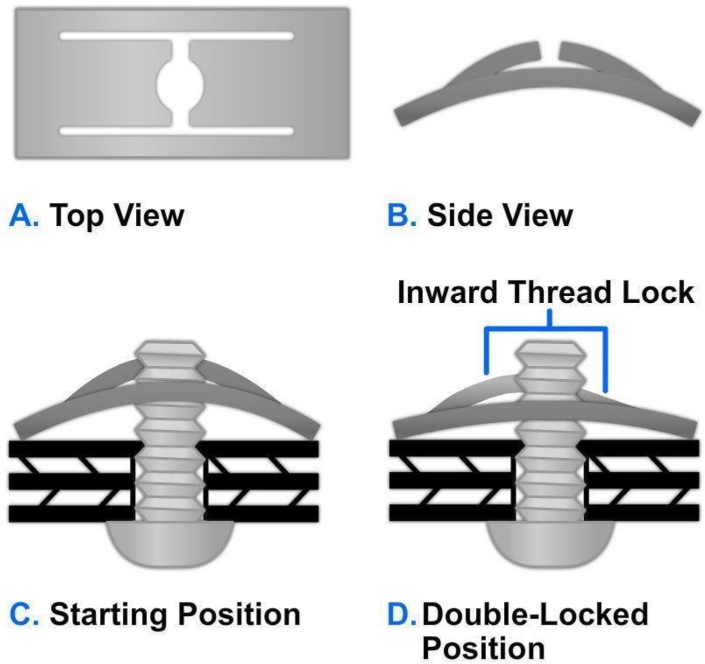
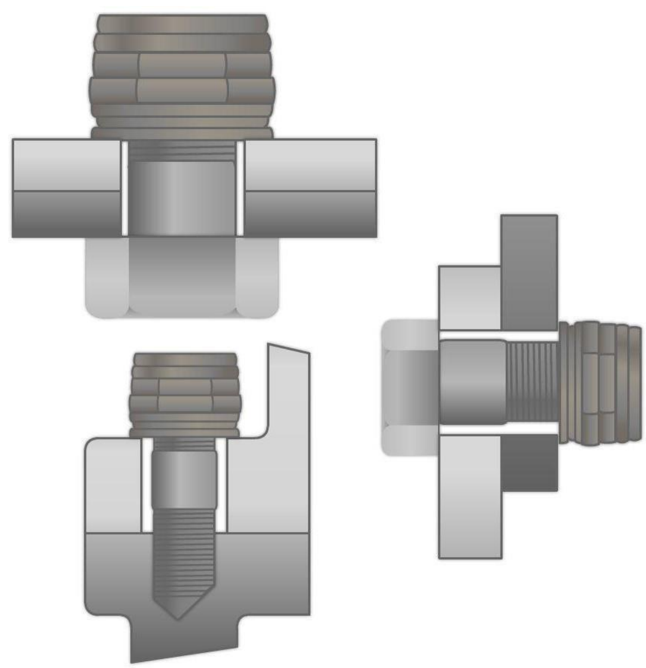


Figure 6-19 — Sheet spring nut.

WING NUTS — These nuts are used where the desired tightness is obtained by the use of your fingers and where the assembly is frequently removed.

KLINCHER LOCKNUTS — Klincher locknuts are used to ensure a permanent and vibration-proof, bolted connection that holds solidly and resists thread wear. It will withstand extremely high or low temperatures and exposure to lubricants, weather, and compounds without impairing the effectiveness of the locking element.



Screws

The most common threaded fastener used in aircraft construction is the screw. The three most used types are the structural screw, machine screw, and the self- tapping screw.

STRUCTURAL SCREWS — Structural screws are used for assembling structural parts. They are made of alloy steel and are heat-treated. Structural screws have a definite grip length and the same shear and tensile strengths as the equivalent size bolt. They differ from structural bolts only in the type of head. These screws are available in round-head, countersunk-head, and brazier-head types, either slotted or recessed for the various types of screwdrivers. See *Figure 6-21*.

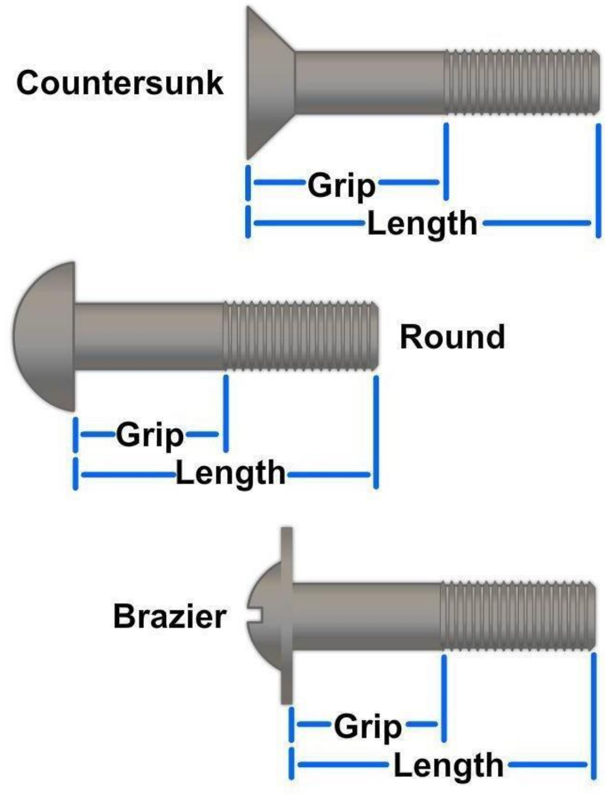


Figure 6-21 — Structural screws.

MACHINE SCREWS — The commonly used machine screws are the flush-head, round-head, fillister-head, socket-head, pan-head, and truss-head types.

Flush-Head — Flush-head machine screws are used in countersunk holes where a flush finish is desired. These screws are available in 82 and 100 degrees of head angle and have various types of recesses and slots for driving.

Round-Head — Round-head machine screws are frequently used to assemble highly stressed aircraft components.

Fillister-Head — Fillister-head machine screws are used as general-purpose screws. They may also be used as cap screws in light applications, such as the attachment of cast aluminum gearbox cover plates.

Socket-Head — Socket-head machine screws are designed to be screwed into tapped holes by internal wrenching. They are used in applications that require high-strength precision products, compactness of the assembled parts, or sinking of the head into holes.

Pan- and Truss-Head — Pan-head and truss-head screws are general-purpose screws used where head height is unimportant. These screws are available with cross-recessed heads only.

SELF-TAPPING SCREWS — A self-tapping screw is one that cuts its own internal threads as it is turned into the hole. Self-tapping screws can be used only in comparatively soft metals and materials. Self-tapping screws may be further divided into two classes or groups: machine self-tapping screws and sheet metal self-tapping screws.

Machine self-tapping screws are usually used for attaching removable parts, such as nameplates, to castings. The threads of the screw cut mating threads in the casting after the hole has been predrilled. Sheet metal self-tapping screws

are used for such purposes as temporarily attaching sheet metal in place for riveting. They may also be used for permanent assembly of nonstructural parts, where it is necessary to insert screws in blind applications.

⚠

CAUTION

Self-tapping screws should never be used to replace standard screws, nuts, or rivets in the original structure. Over a time, vibration and stress will loosen this type of

Washers

Washers such as ball socket and seat washers, taper pin washers, and washers for internal-wrenching nuts and bolts have been designed for special applications. See *Figure 6-22*.

Ball socket and seat washers are used where a bolt is installed at an angle to the surface, or where perfect alignment with the surface is required at all times. These washers are used together.

Taper pin washers are used in conjunction with threaded taper pins. They are installed under the nut to effect adjustment where a plain washer would distort.

Washers for internal-wrenching nuts and bolts are used in conjunction with NAS internal-wrenching bolts. The washer used under the head is countersunk to seat the bolt head or shank radius. A plain washer is used under the nut.

Figure 6-22 — Various types of special washers.

Turnlock Fasteners

Turnlock fasteners are used to secure panels that require frequent removal. These fasteners are available in several different styles and are usually referred to by the manufacturer's trade name.

CAMLOC FASTENERS — The 4002 series Camloc fastener consists of four principal parts: the receptacle, the grommet, the retaining ring, and the stud assembly. See *Figure 6-23*. The receptacle is an aluminum alloy forging mounted in a stamped sheet metal base. The receptacle assembly is riveted to the access door frame, which is attached to the structure of the aircraft. The grommet is a sheet metal ring held in the access panel with the retaining ring. Grommets are furnished in two types: the flush type and the protruding type. Besides serving as a grommet for the hole in the access panel, it also holds the stud assembly. The stud assembly consists of a stud, a cross pin, a spring, and a spring cup. The assembly is designed so it can be quickly inserted into the grommet by compressing the spring. Once installed in the grommet, the stud assembly cannot be removed unless the spring is again compressed.

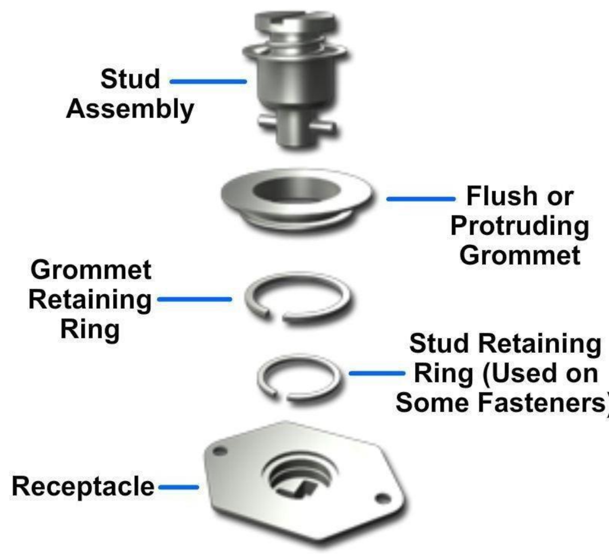


Figure 6-23 — Camloc 4002 series fastener.

The Camloc high-stress panel fastener, shown in *Figure 6-24*, is a high-strength, quick-release rotary fastener and may be used on flat or curved inside or outside panels. The fastener may have either a flush or a protruding stud. The studs are held in the panel with flat or cone-shaped washers—the latter being used with flush fasteners in dimpled holes. This fastener may be distinguished from screws by the deep No. 2 Phillips recess in the stud head and by the bushing in which the stud is installed.

A threaded insert in the receptacle provides an adjustable locking device. As the stud is inserted and turned counterclockwise one-half turn or more, it screws out the insert to permit the stud key to engage the insert cam when turned clockwise. Rotating the studclockwise one-fourth turn engages the insert. Continued rotation screws the insert in and tightens the fastener. Turning the stud one-fourth turn counterclockwise will release the stud, but will not screw the insert out far enough to permit re-engagement. The stud should be turned at least one-half turn counterclockwise to reset theinsert.

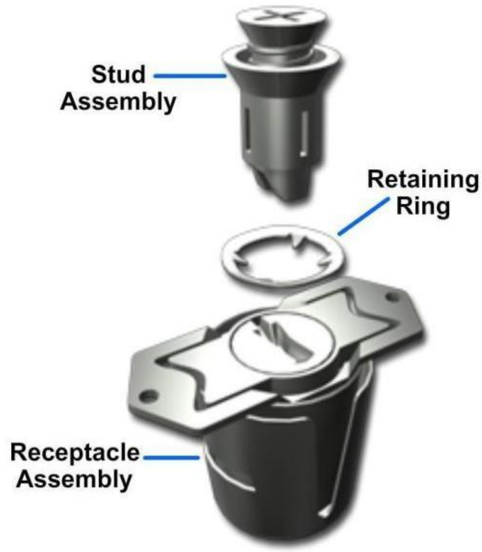


Figure 6-24 — Camloc high-stress panel fastener.

DZUS FASTENERS — Dzus fasteners are available in two types. A light-duty type is used on box covers, access hole covers, and lightweight fairings. The heavy- duty type is used on cowling and heavy fairings. The main difference between the two Dzus fasteners is a grommet, which is used only on the heavy-duty fasteners. Otherwise, their construction features are about the same.

Figure 6-25 shows the parts of a light-duty Dzus fastener. Notice that they include a spring and a stud. The spring is made of cadmium-plated steel music wire and is usually riveted to an aircraft structural member. The stud comes in a number of designs (as shown in *Views A*, *B*, and *C*) and mounts in a dimpled hole in the cover assembly.

When the panel is being positioned on an aircraft, the spring riveted to the structural member enters the hollow center of the stud. Then, when the stud is turned about one-fourth turn, the curved jaws of the stud slip over the spring and compress it. The resulting tension locks the stud in place and secures the panel.

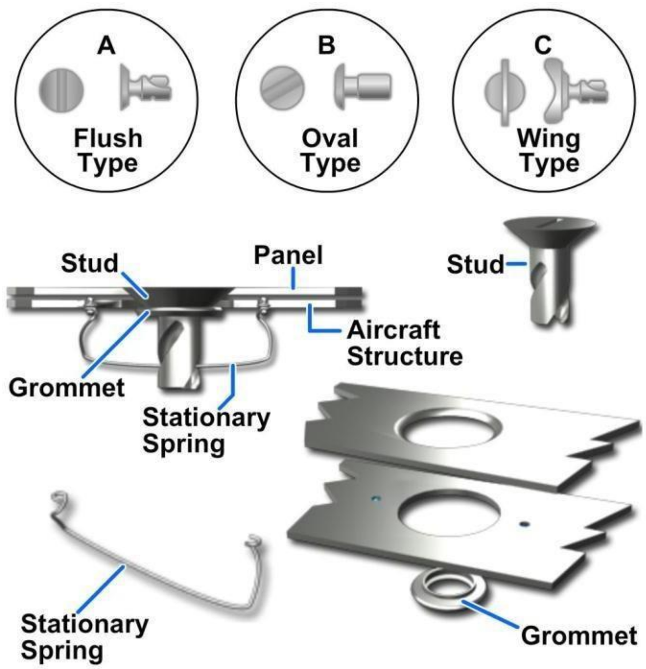


Figure 6-25 — Dzus fastener.

Miscellaneous Fasteners

Some fasteners cannot be classified as rivets, turnlocks, or threaded fasteners. Included in this category are connectors, couplings, clamps, taper and flat-head pins, snap rings, studs, and heli-coil inserts.

FLEXIBLE CONNECTORS AND COUPLINGS — A variety of clamping devices are used to connect ducting sections to each other or to various components. Whenever lines, components, or ducting are disconnected or removed for any reason, suitable plugs, caps, or coverings should be installed on the openings to prevent the entry of foreign materials. Various parts should also be tagged to ensure correct reinstallation. Care should be exercised during handling and installation to ensure that flanges are not scratched, distorted, or deformed. Flange surfaces should be free of dirt, grease, and corrosion. The protective flange caps should be left on the ends of the ducting until the installation progresses to the point that removal is necessary.

In most cases it is mandatory to discard and replace seals and gaskets. It is important to ensure that seals and gaskets are properly seated and that mating and alignment of flanges are fitted. This will prevent the excessive torque required to close the joint, which imposes structural loads on the clamping devices.
Adjacent support clamps and brackets should remain loose until installation of the coupling has been completed.

Some of the most commonly used plain-band couplings are shown in *Figure 6-26*. When a hose is installed between two duct sections, the gap between the duct ends should be a minimum of 1/8 of an inch and a maximum of 3/4 of an inch.
When the clamps are installed on the connection, the clamps should be 1/4 of an inch from the end of the connector. Misalignment between the ducting ends should not exceed 1/8 of an inch.

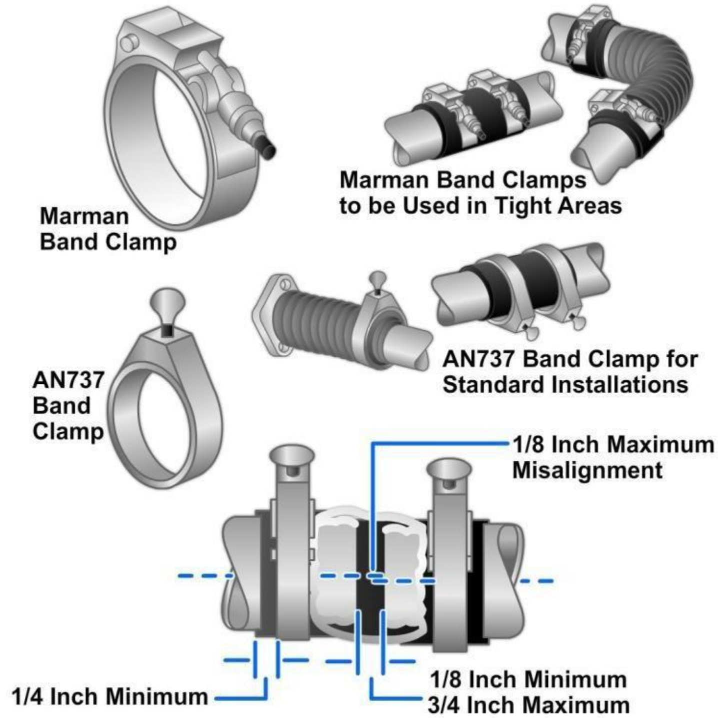


Figure 6-26 — Flexible line connectors.

Marman clamps are commonly used in ducting systems and should be tightened to the torque value indicated on the coupling. Tighten all couplings in the manner and to the torque value specified on the clamp or in the applicable maintenance instruction manual (MIM).

When flexible couplings are installed—such as the one shown in *Figure 6-27*—the following steps are recommended to assure proper security:

1. Fold back half of the sleeve seal and slipit onto thesleeve.
2. Slide the sleeve (with the sleeve seal partially installed) onto theline.
3. Position the split sleeves over theline beads.
4. Slide the sleeve over the split sleevesand fold over the sleeve seal so it covers the entire sleeve.
5. Install the coupling over the sleeveseal and torque to correctvalue.

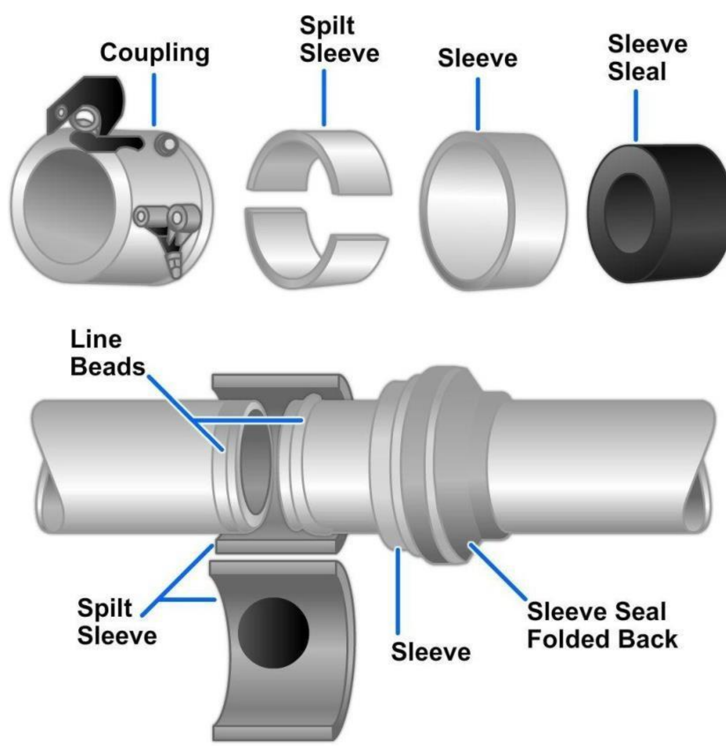


Figure 6-27 — Flexible line coupling.

RIGID COUPLINGS — The rigid line coupling shown in *Figure 6-28* is referred to as a V-band coupling. When installed in restricted areas, some of the stiffness of the coupling can be overcome by tightening the coupling over a spare set of flanges and a gasket to the recommended torque value of the joint. Before the coupling is removed, it should be tapped a few times with a plastic mallet.

When rigid couplings are installed, the steps below should be followed:

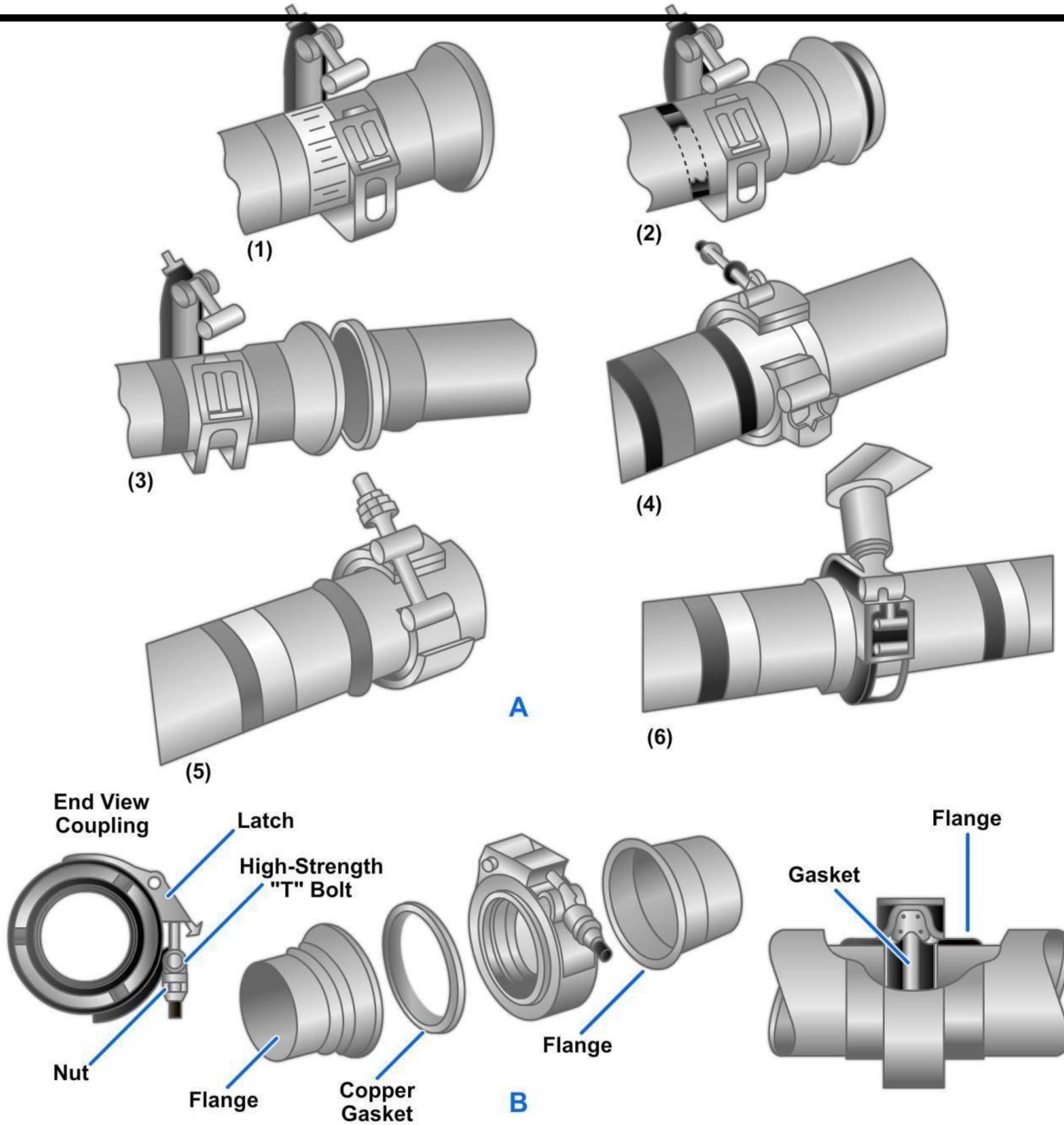


Figure 6-28 — Installation of rigid line couplings.

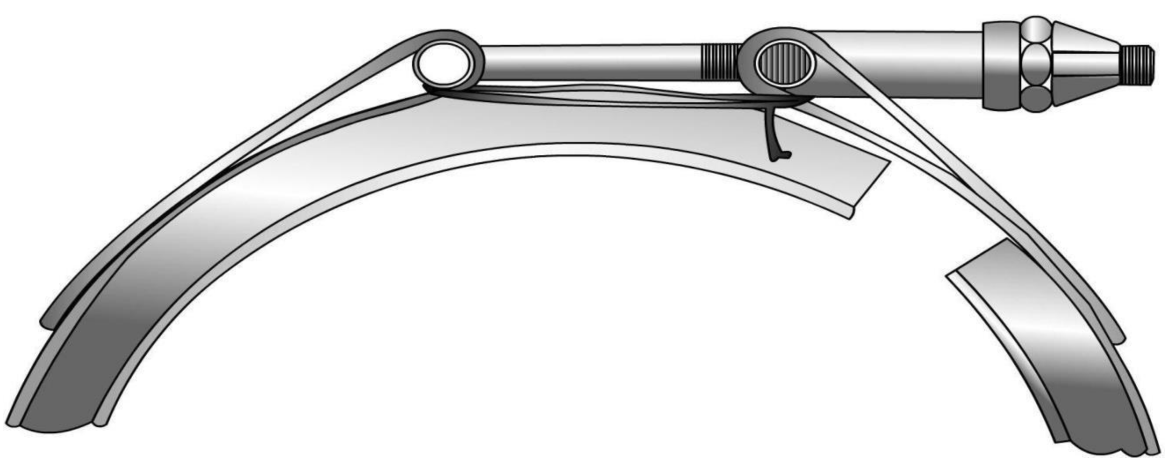


Figure 6-29 — Safetying a V-band coupling.

TAPER PINS — Taper pins are used in joints that carry shear loads and where the absenceof clearance is essential. See *Figure 6-30*. The threaded taper pin is used with a taper pin washer and a shear nut if the taper pin is drilled, or with a self-locking nut if undrilled. When a shear nut is used with the threaded taper pin and washer, the nut is secured with a cotterpin.

FLAT-HEAD PINS — The flat-head pin is used with tie rod terminals or secondary controls that do not operate continuously. The flat-head pin should be secured with a cotter pin. The pin is normally installed with the head up. See *Figure 6-30*. This precaution is taken to maintain the flat-head pin in the installed position in case of cotter pin failure.

SNAP RINGS — A snap ring is a ring of metal, either round or flat in cross section, that 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. The internal types fit in a groove inside a cylinder. Special pliers are designed to install each type of snap ring.

Snap rings can be reused as long as they retain their shape and springlike action. External snap rings may be safety wired, but internal types are never safetied.

STUDS — There are four types of studs used in aircraft structural applications. They are the coarse thread, fine thread, stepped, and lockring studs. Studs may be drilled or undrilled on the nut end. Coarse (NAS183) and fine (NAS184) thread studs are manufactured from alloy steel and are heat-treated. They have identical threads on both ends. The stepped stud has a different thread on each end of the stud. The lockring stud may be substituted for undersize or oversize studs. The lockring on this stud prevents it from backing out due to vibration, stress, or temperaturevariations.

Refer to the *Structural Hardware Manual*, NAVAIR 01-1A-8, for more detailed information on studs.

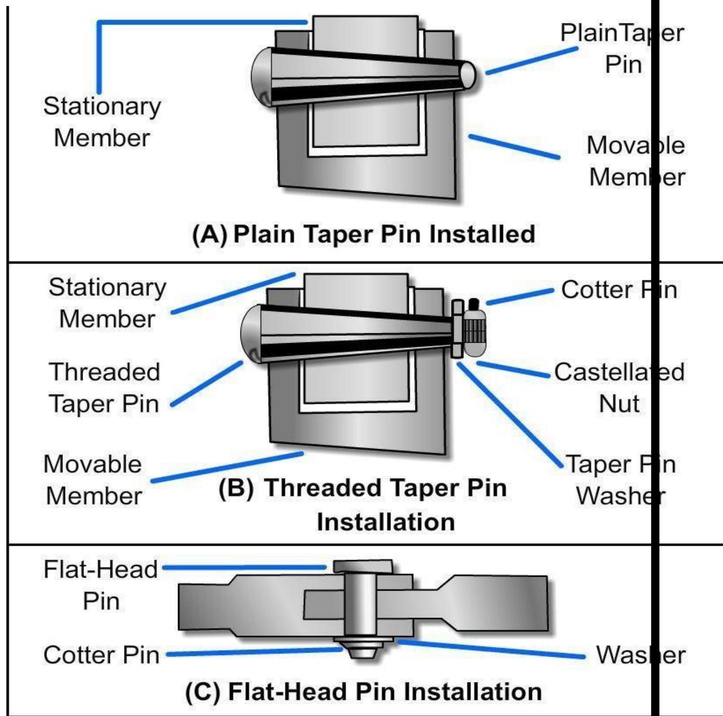


Figure 6-30 — Types of aircraft pins.

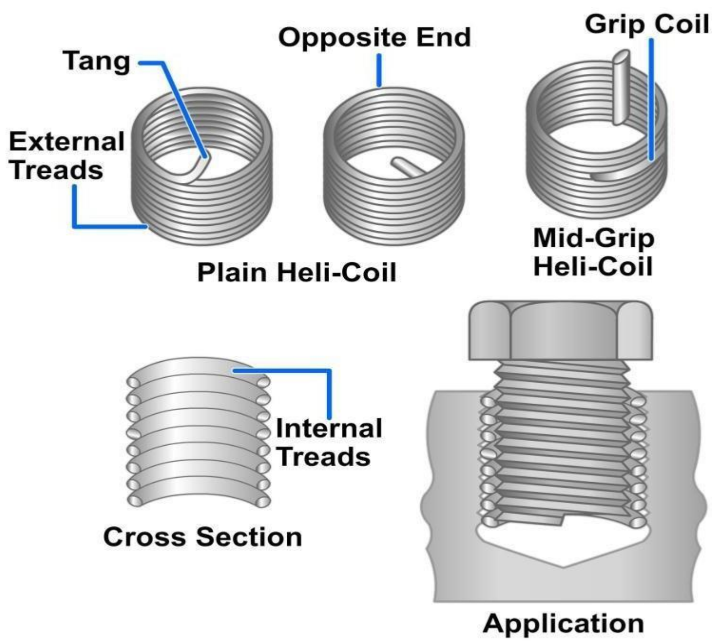


Figure 6-31 — Heli-coil insert.

HELI-COIL INSERTS — Heli-coil thread inserts are primarily designed to be used in materials that are not suitable for threading because of their softness. The inserts are made of a diamond cross- sectioned stainless steel wire made with a tang that forms a portion of the bottom coil offset and is used to drive the insert. This tang is left on the insert after installation, except when its removal is necessary to provide clearance for the end of the bolt. The tang is notched to break off from the body of the insert, thereby providing full penetration for the fastener.

The second type of insert used is the self-locking, mid-grip insert, which has a specially formed grip coil midway on the insert. This produces a gripping effect on the engaging screw. For quick identification, the self-locking, mid-grip inserts are dyed red.

CABLES

A cable is a group of wires or a group of strands of wires twisted together into a strong wire rope. The wires or strands may be twisted in various ways. The relationship of the direction of twist of each strand to each other and to the cable as a whole is called the *lay*. The lay of the cable is an important factor in its strength. If the strands are twisted in a direction opposite to the twist of the strands around the center strand or core, the cable will not stretch (or set) as much as one in which they are all twisted in the same direction. This direction of twist (in opposite direction) is most commonly adopted, and it is called a *regular* or an *ordinary lay*. Cables may have a right regular lay or a left regular lay. If the strands are twisted in the direction of twist around the center strand or core, the lay is called a *lang lay*. There is a right and left lang lay. The only other twist arrangement—twisting the strands alternately right and left, and then twisting them all either to the right or to the left about the core—is called a *reverse lay*. Most aircraft cables have a right regularlay.

When aircraft cables are manufactured, each strand is first formed to the spiral or helical shape to fit the position it is to occupy in the finished cable. The process of such forming is called preforming, and cables made by such a process are said to be *preformed*. The process of preforming is adopted to ensure flexibility in the finished cable and to relieve bending and twisting stresses in the strands as 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.

Aircraft control cables are fabricated either from flexible, preformed carbon steel wire or from flexible, preformed, corrosion-resistant steel wire. The small corrosion-resistant steel cables are made of steel containing not less than 17 percent chromium and 8 percent nickel, while the larger ones (those of the 5/16-, 3/8-, and 7/16-inch diameters) are made of steel that, in addition to the amounts of chromium and nickel just mentioned, also contains not less than 1.75 percentmolybdenum.

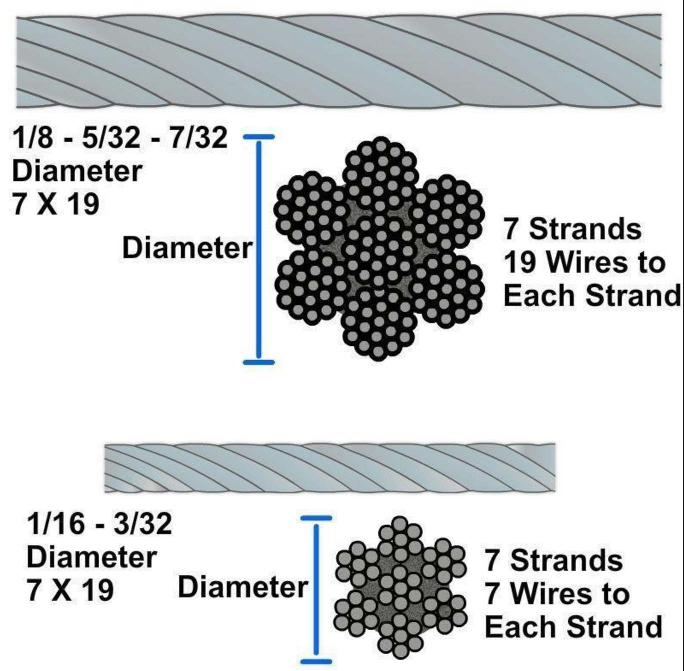


Figure 6-32 — Cable cross section.

Cables may be designated 7 × 7, 7 × 19, or 6 × 19 according to their construction. A 7 × 7 cable consists of six strands of seven wires each, laid around a center strand of seven wires. A 7 × 19 cable consists of six strands of 19 wires, laid around a 19-wire central strand. A 6 × 19 IWRC cable consists of six strands of 19 wires each, laid around an independent wire rope center.

The size of cable is given in terms of diameter measurement. A 1/8-inch cable or a 5/16-inch cable means that the cable measures 1/8 inch or 5/16 inch in diameter, as shown in *Figure 6-32*. Note that the cable diameter is that of the smallest circle that would enclose the entire cross section of the cable. Aircraft control cables vary in diameters, ranging from 1/16 of an inch to 3/8 of an inch.

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 ballends.

Threaded-end, fork-end, and eye-end terminals 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 either at the ends or in the center of a cable run. *Figure 6-33* shows the various types of terminal fittings.

Thimble, bushing, and shackle 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 6-34* shows these fittings.

Turnbuckles

A turnbuckle is a mechanical screw device that consists of two threaded terminals and a threaded barrel. *Figure 6-35* shows a typical turnbuckle assembly. Turnbuckles are fitted in the cable assembly to make minor adjustments in cable length and to adjust 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 threads internally. The end of the barrel, with left-hand threads inside, can usually be identified by either a groove or knurl around the end of thebarrel.

Barrels and terminals are available in both long and short lengths.

When you install a turnbuckle in a control system, it is necessary to screw both of the terminals an equal number of turns into the turnbuckle barrel. It is also essential that all turnbuckle terminals be screwed into the barrel, at least, until not more than three threads are

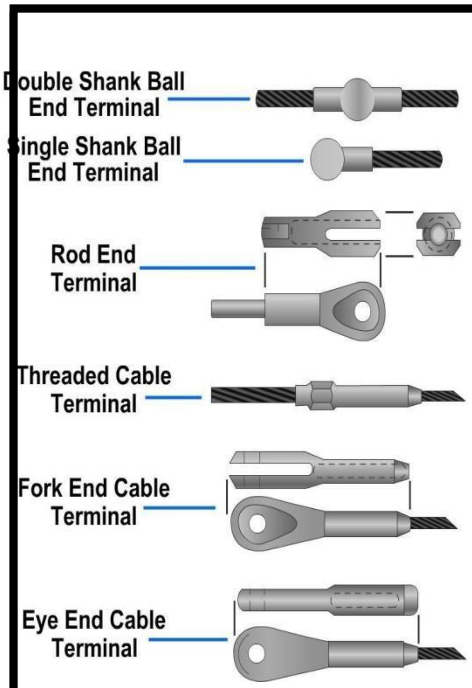


Figure 6-33 — Types of cable terminal fittings.

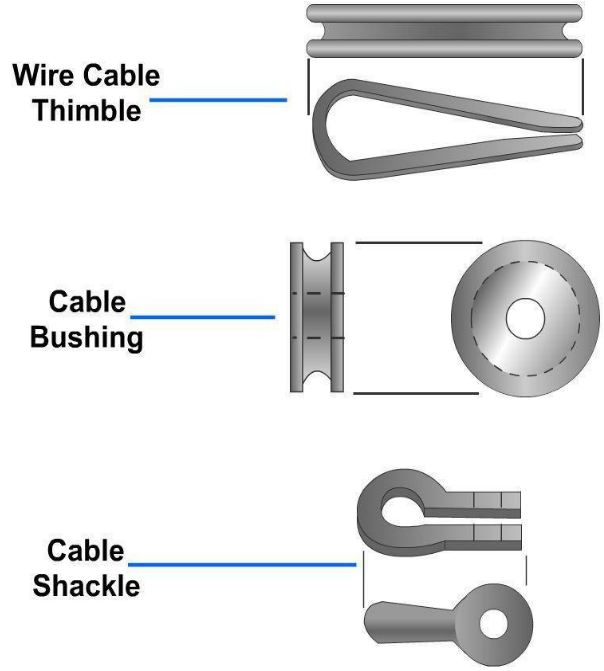


Figure 6-34 — Thimble, bushing, and shackle fittings.

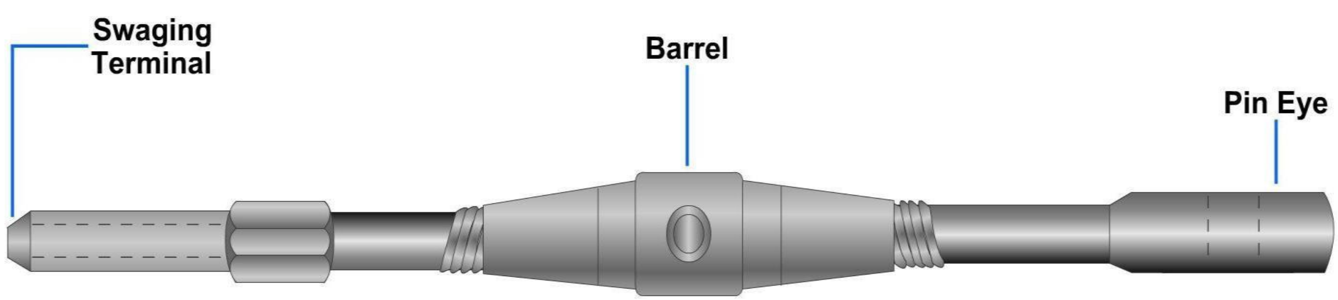


Figure 6-35 — Typical turnbuckle assembly.

exposed. On initial installation, the turnbuckle terminals should not be screwed inside the turnbuckle barrel more than four threads. *Figure 6- 36* shows turnbuckle thread tolerances.

After a turnbuckle is properly adjusted, it must be safetied. There are several methods of safetying turnbuckles. However, only two methods have been adopted as standard procedures by the services: the clip-locking (preferred) method and the wire- wrapping method.

Adjustable Connector Links

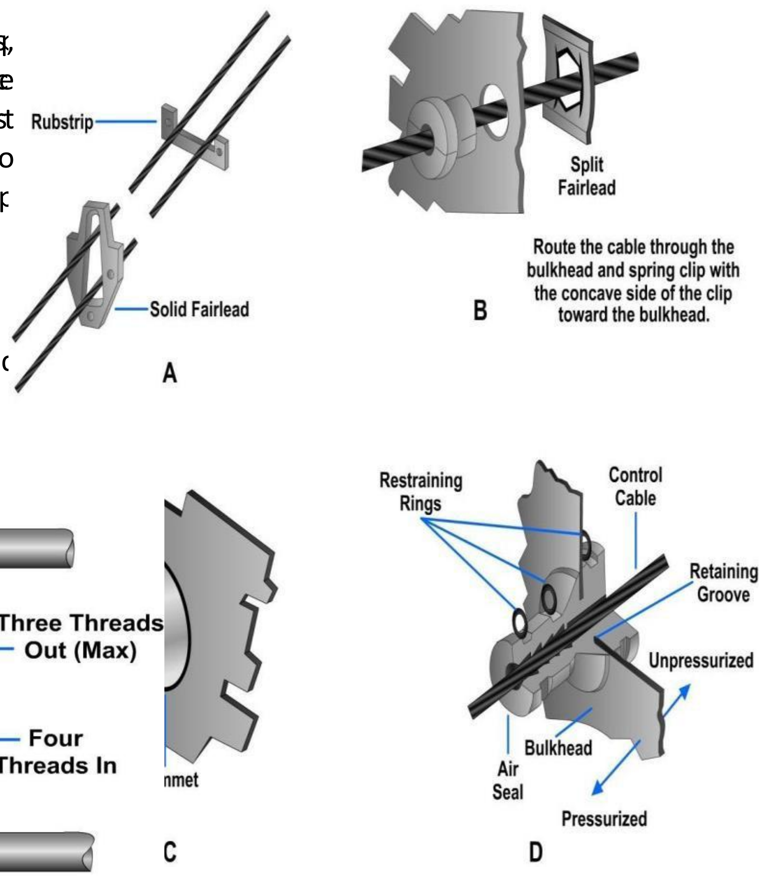
An adjustable connector link consists of two or three metal strips with holes arranged that they may be matched and secured with a clevis bolt to adjust the length of the connector. They are installed in cable assemblies to make major adjustments in cable length and to compensate for cable stretch. Adjustable connector links are usually used in very long cable assemblies.

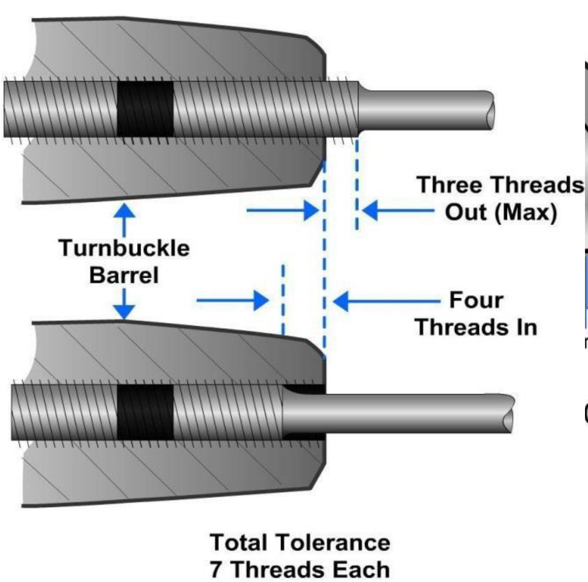
GUIDES

Fairleads (rubstrips), grommets, guides. They are used to prote rubbing against nearby metalpart They are also used as supports to cable. *Figure 6-37* shows some typ

Fairleads

Fairleads may be made of a solic when they pass through holes in





reduce cable whipping and vibration in long runs of cable. Split fairleads are made for easy installation around single cables to protect them from rubbing on the edges of holes.

Grommets

Grommets are made of rubber, and they are used on small openings where single cables pass through the walls of unpressurized compartments.

Pressure Seals

Pressure seals are used on cables or rods that must move through pressurized bulkheads. They fit tightly enough to prevent air pressure loss, but not so tightly as to hinder movement of the unit.

Pulleys

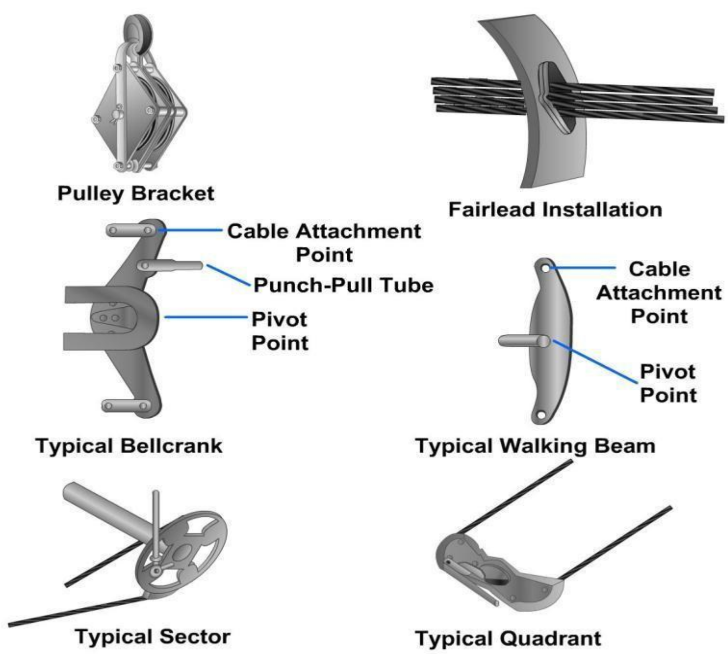
Pulleys (or sheaves) are grooved wheels used to change cable direction and to allow the cable to move with a minimum of friction. Most pulleys used on aircraft are made from layers of cloth impregnated with phenolic resin and fused together under high temperatures and pressures. Aircraft pulleys are extremely strong and durable and cause minimum wear on the cable passing over them. Pulleys are provided with grease-sealed bearings and usually do not require further lubrication.
However, pulley bearings may be pressed out, cleaned, and relubricated with special equipment. This is usually done by depot-level maintenance activities.

Pulley brackets made of sheet or cast aluminum are required with each pulley installed in the aircraft.

See *Figure 6-38*. Besides holding the pulley in the correct position and at the correct angle, the brackets prevent the cable from slipping out of the groove on the pulley wheel.

SECTORS AND QUADRANTS

These units are generally constructed in the form of an arc or in a complete circular form. They are grooved around the outer circumference to receive the cable, as shown in *Figure 6-38*. The terms *sector* and *quadrant* are used interchangeably. Sectors and quadrants are similar to bell cranks and walking beams, which are used for the same purpose in rigid control systems.



AIRCRAFT ELECTRICAL HARDWARE

An important part of aircraft electrical maintenance is determining the correct type of electrical hardware for a given job. These maintenance functions normally require a joint effort on the part of the AM and the Aviation

WIRE AND CABLE

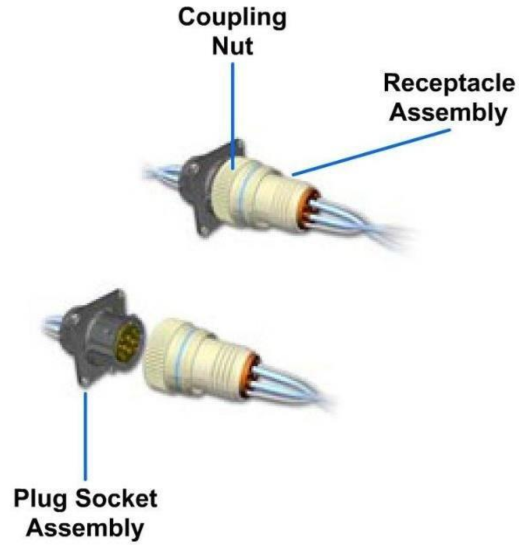
For purposes of electrical installations, a *wire* is defined as a stranded conductor covered with an insulating material. The term *cable*, as used in aircraft electrical installations, includes the following:

- Two or more insulated conductors contained in the same jacket (multiconductorcable)
- Two or more insulated conductors twisted together (twisted pair)
- One or more insulated conductors covered with a metallic braided shield (shieldedcable)
- A single insulated conductor with a metallic braided outer conductor (RFcable)

For wire replacement work, the aircraft MIM should be consulted first. The manual should list the wire used in a given aircraft.

CONNECTORS

Connectors are devices attached to the ends of cables and sets of wires to make them easier to connect and disconnect. Each connector consists of a plug assembly and a receptacle assembly. The two assemblies are coupled by means of a coupling nut. Each consists of an aluminum shell containing an insulating insert that holds the current-carrying contacts. The plug is usually attached to the cable end and is the part of the connector on which the coupling nut is mounted. The receptacle is the half of the connector to which the plug is connected. It is usually mounted on a part of the equipment. One type of connector assembly commonly used in aircraft electrical systems is shown in *Figure 6-39*.

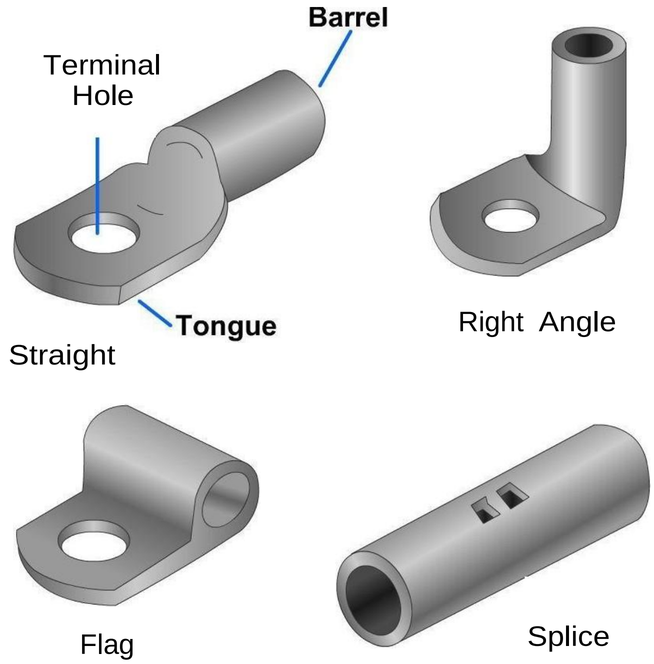


TERMINALS

Since most aircraft wires are stranded, it is necessary to use terminal lugs to hold the strands together. This allows a means of fastening the wires to terminal studs. The terminals used in electrical wiring are either of the soldered or crimped type. Terminals used in repair work must be of the size and type specified in the applicable MIM. The solderless crimped-type terminals are generally recommended for use on naval aircraft. Soldered-type terminals are usually used in emergencies only.

The basic types of solderless terminals are shown in *Figure 6-40*. They are the straight, right angle, flag, and splice types. There are variations of these types.

Figure 6-39 — C


Barrel
Terminal
Hole
Tongue
Straight
Right Angle
Flag
Splice

BONDING

An aircraft can become highly charged with static electricity while in flight. If the aircraft is improperly bonded, not all metal parts have the same amount of static charge. A difference of potential exists between the various metal surfaces. If the resistance between insulated metal surfaces is great enough, charges can accumulate. The potential difference could become high enough to cause a spark. This constitutes a fire hazard and also causes radio interference. If lighting strikes an aircraft, a good conducting path for heavy current is necessary to minimize severe arcing and sparks.

When all metal parts of an aircraft are connected to complete an electrical unit, the result is called *bonding*. Bonding connections are made of screws, nuts, washers, clamps, and bonding jumpers.

Figure 6-41 shows a typical bonding link installation.

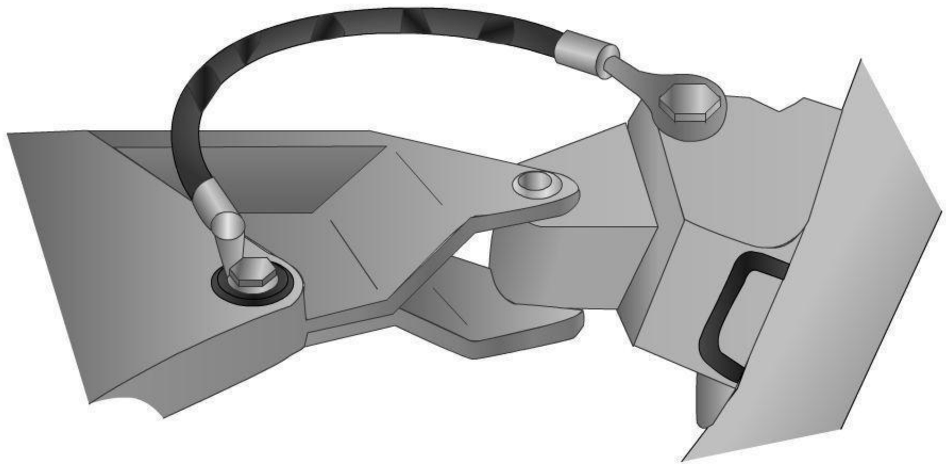


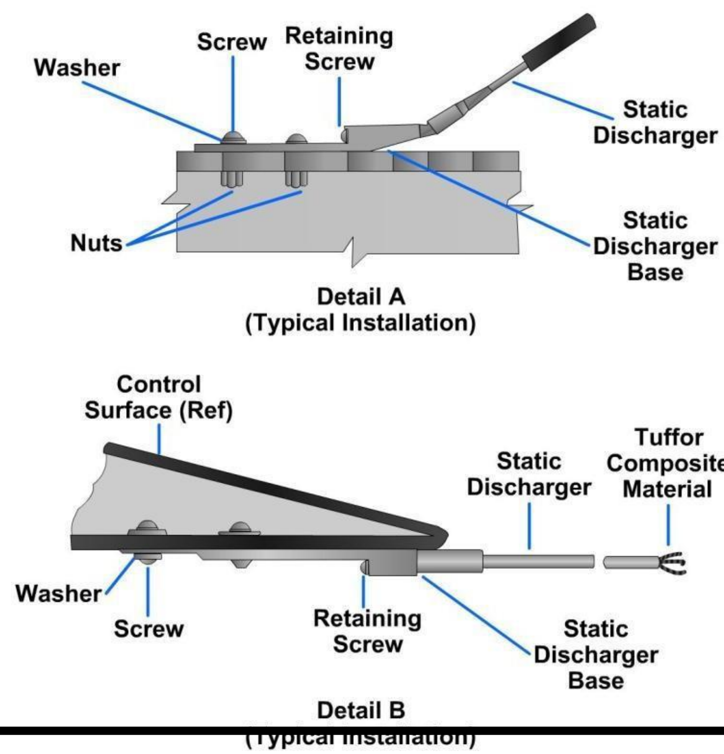
Figure 6-41 — Typical bonding link installation.

Bonding also provides the necessary low-resistance return path for single-wire electrical systems. This low-resistance path provides a means of bringing the entire aircraft to the earth's potential when it is grounded.

When an inspection is performed, both bonding connections and safetying devices must be inspected with great care.

STATIC DISCHARGERS

Static dischargers are commonly known as *static wicks* or *static discharge wicks*. They are used on aircraft to allow the continuous satisfactory operation of onboard navigation and radio communication systems. During adverse charging conditions, they limit the potential static buildup on the aircraft and control interference generated by static charge. Static dischargersare not lighting arrestors and do not reduce or increase the likelihood of an aircraft being struck by lightning. Static dischargers are subject to damage or significant changes in resistance characteristics as a result of lightning strike to the aircraft, and they should be inspected after a lightning strike to ensure proper static discharge operation.



Static dischargers are fabricated with a wick of wire or a conductive element on one end, which provides a high-resistance discharge path between the aircraft and the air. See *Figure 6-42.* They are attached on some aircraft tothe

Figure 6-42 -Typical static dischargers.

ailerons, elevators, rudder, wing, horizontal and vertical stabilizer tips, etc. Refer to your aircraft's MIM for maintenance procedures.

TORQUING OF FASTENERS

Fastener fatigue failure accounts for the majority of all fastener problems. Fatigue breaks are caused by insufficient tightening and the lack of proper preload or clamping force. This results in movement between the parts of the assembly and the bending back and forth or cyclic stressing of the fastener. Eventually, cracks will progress to the point that the fastener can no longer support its designed load. At this point the fastener fails with varying consequences.

TYPES OF TORQUE WRENCHES

The two most commonly used torque wrenches are the dial indicating type and the setting or click type.

Dial Indicating Type

This torque wrench measures change in applied torque through a deflecting member. A dial or digital readout is located below the handle to permit convenient and accurate reading. Indicating torque wrenches operate in clockwise and counterclockwise directions.

Setting or Click Type

This type of wrench compares the applied load to a self-contained standard. Reset is automatic upon release of applied load.

TORQUING PROCEDURES

For the nut to properly load the bolt and prevent premature failure, a designated amount of torque must be applied. Proper torque reduces the possibility of the fastener loosening while in service. The correct torque to apply when you are tightening an assembly is based on many variables. The fastener is subjected to two stresses when it is tightened. These stresses are tension and torsion.
Tension is the desired stress, while torsion is the undesirable stress caused by friction. A large percentage of applied torque is used to overcome this friction, so that only tension remains after tightening. Proper tension reduces the possibility of fluid leaks.

The recommended torque values provided in *Table 6-2* have been established for average dry, cadmium-plated nuts for both the fine and coarse thread series. Thread surface variations such as paint, lubrication, hardening, plating, and thread distortion may alter these values considerably. The torque values must be followed unless the MIM or structural repair manual for the specific aircraft requires a specific torque for a given nut. Torque values vary slightly among manufacturers. When the torque values are included in a technical manual, these values take precedence over the standard torque values provided in the *Structural Hardware Technical Manual*, NAVAIR01-1A-8.

Separate torque tables and torquing considerations are provided in NAVAIR 01-1A-8 for the large variety of nuts, bolts, and screws used in aircraft construction. This manual should be used when specific torque values are not provided as a part of the removal/replacement instructions.

ranges should be used only when materials and surfaces being joined are of sufficient thickness, area, and strength to resist breaking, warping, or other damage.

For corrosion-resistant steel nuts, the torque values given for shear-type nuts should be used. The use of any type of drive-end extension on a torque wrench changes the dial reading required to obtain the actual values indicated in the torque range tables. See *Figure 6-43*.

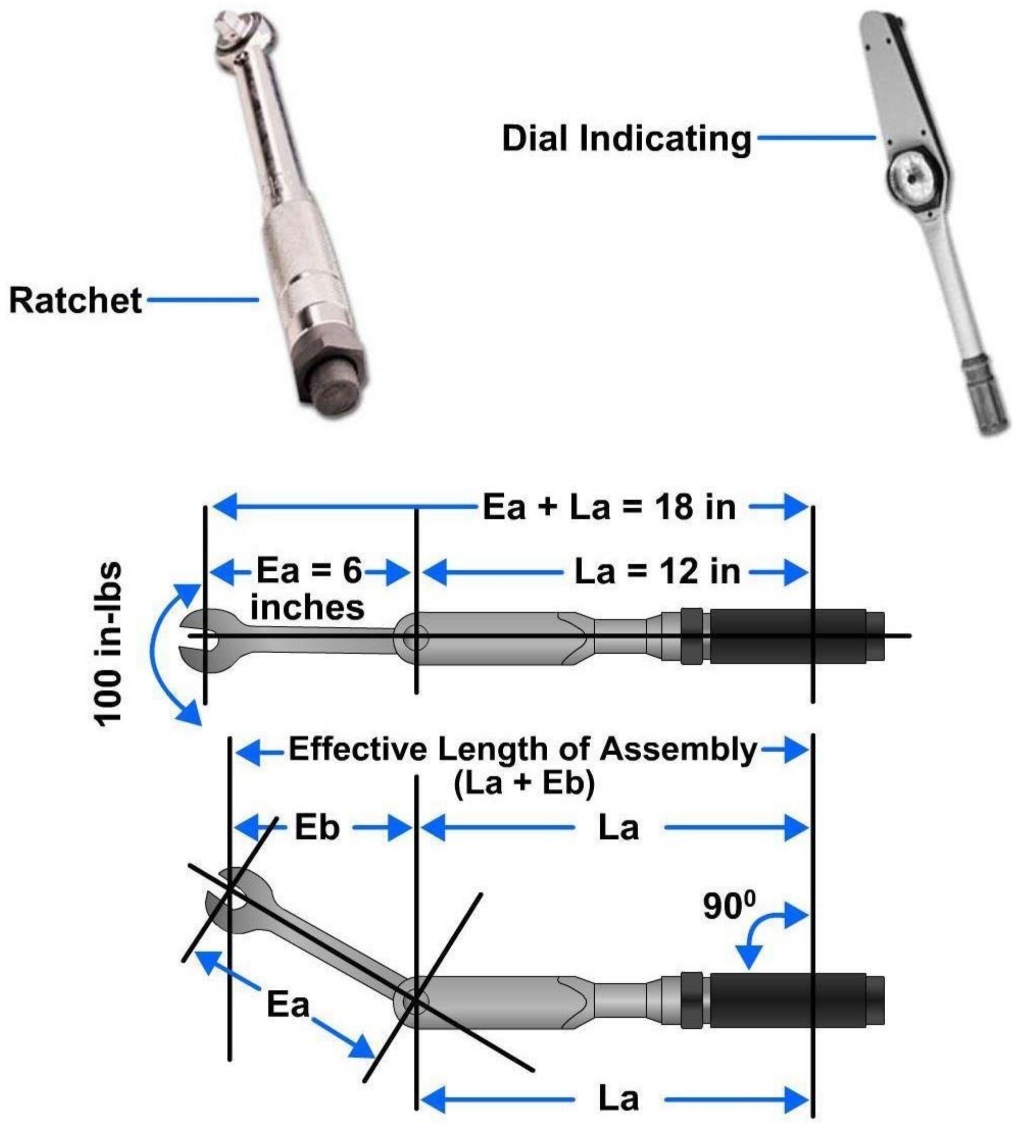


Figure 6-43 — Torque wrenches.

Table 6-2 — Recommended Torque Values (Inch-Pounds)

CAUTION
THE FOLLOWING TORQUE VALUES ARE DERIVED FROM OIL-FREE CADMIUM-PLATED THREADS.

TORQUE LIMITS RECOMMENDED FOR INSTALLATION (BOLTS LOADED PRIMARILY IN SHEAR)			MAXIMUM ALLOWABLE TIGHTENING TORQUE LIMITS	
Tap Size	Tension-type nuts MS20365 and AN310 (40,000 psi in bolts)	Shear-type nuts MS20364 and AN320 (24,000 psi in bolts)	Nuts MS20365 and AN310 (90,000 psi in bolts)	Nuts MS20364 and AN320 (54,000 psi in bolts)
FINE THREAD SERIES				
8-36	12-15	7-9	20	12
10-32	20-25	12-15	40	25
1/4-28	50-70	30-40	100	60
5/16-24	100-140	60-85	225	140
3/8-24	160-190	95-110	390	240
7/16-20	450-500	270-300	840	500
1/2-20	480-690	290-410	1,100	660
9/16-18	800-1000	480-600	1,600	960
5/8-18	1,100-1,300	600-780	2,400	1,400
3/4-16	2,300-2,500	1,300-1,500	5,000	3,000
7/8-14	2,500-3,000	1,500-1,800	7,000	4,200
1-14	3,700-5,500	2,200-3,300*	10,000	6,000
11/8-12	5,000-7,000	3,000-4,200*	15,000	9,000
11/4-12	9,000-11,000	5,400-6,600*	25,000	15,000
COARSE THREAD SERIES				
8-32	12-15	7-9	20	12
10-24	20-25	12-15	35	21
1/4-20	40-50	25-30	75	45
5/16-18	80-90	48-55	160	100
3/8-16	160-185	95-100	275	170
7/16-14	235-255	140-155	475	280
1/2-13	400-480	240-290	880	520
9/16-12	500-700	300-420	1,100	650
5/8-11	700-900	420-540	1,500	900
3/4-10	1,150-1600	700-950	2,500	1,500
7/8-9	2,200-3000	1,300-1,800	4,600	2,700

The above torque values may be used for all cadmium-plated steel nuts of the fine or coarse thread series, which have approximately equal number of threads and equal face bearing areas.
*Estimated corresponding values.

TORQUING COMPUTATION

When using a drive-end extension, you must compute the torque wrench reading using the formula in *Figure 6-44*:

$$S = \frac{T \times L_a}{L_a + E_a}$$

Figure 6-44 — Drive-end extension formula.

Where:

S = handle setting or reading

T = torque applied at end of adapter

L_a = length of handle in inches

E_a = length of extension in inches

To exert 100 inch-pounds at the end of the wrench and extension, when L_a equals 12 inches and E_a equals 6 inches, it is possible to determine the handle setting by making the calculation shown in *Figure 6-45*.

$$S = \frac{T \times L_a}{L_a + E_a}$$

$$S = \frac{100 \times 12}{12 + 6}$$

$$S = \frac{1200}{18}$$

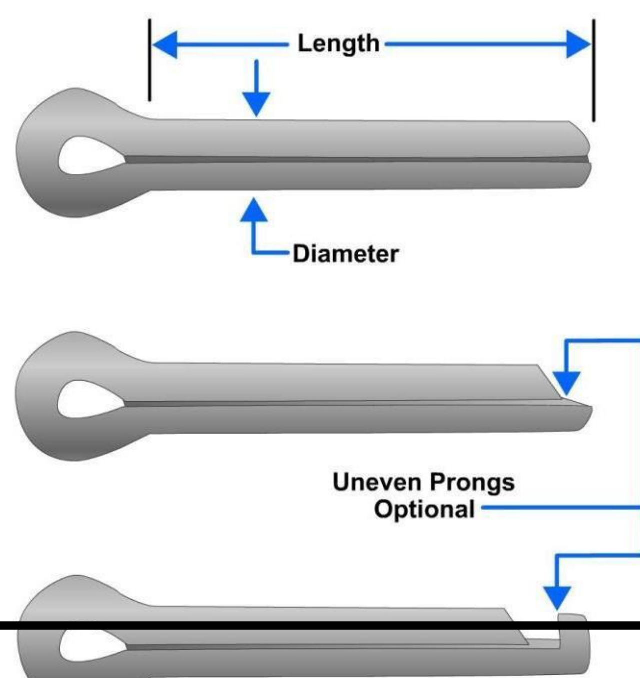
Figure 6-45 — Sample calculation.

Whenever possible, attach the extension in line with the torque wrench. When it is necessary to attach the extension at an angle to the torque wrench, the effective length of the assembly will be $L_a + E_a$, as shown in *Figure 6-43*. In this instance, length E_b must be substituted for length E_a in the formula.

AIRCRAFT SAFETYING METHODS

There are many different types of safetying materials used to stop rotation and other movement of fasteners. They are used to secure other equipment that may come loose due to vibration in the aircraft.

COTTER PINS



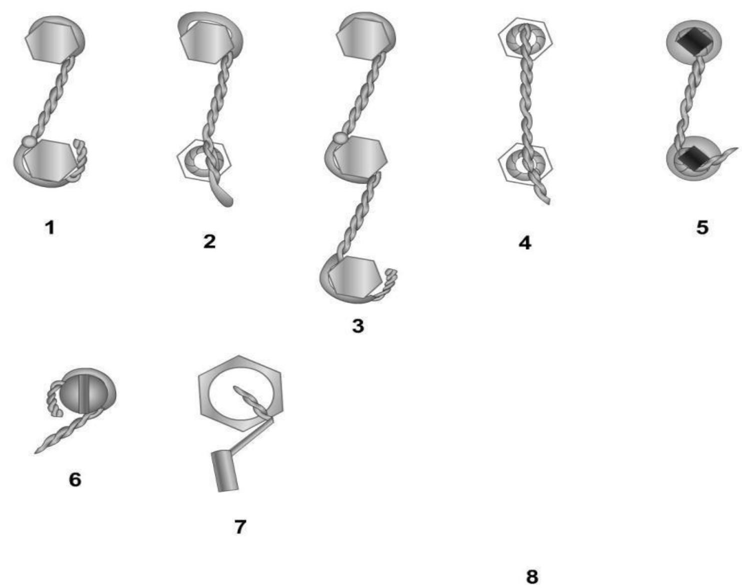
Cotter pins are used to secure bolts, screws, nuts, and pins. Some cotter pins are made of low-carbon steel, while others consist of stainless steel and are more resistant to corrosion. Also, stainless steel cotter pins may be used in locations where nonmagnetic material

is required. Regardless of shape or material, all cotter
p1ns are used Tor tne same general

NOTE

Whenever uneven prong cotter pins are used, the length measurement is to the end of the shortest

SAFETY WIRE



Safety wire comes in many types and sizes. First, the correct type and size of wire for the job must be selected. Annealed corrosion- resistant wire is used in high-temperature, electrical equipment and aircraft instrument applications. All nuts—except the self-locking types—must be safetied; the method used depends upon the particular installation.

Figure 6-47 shows various methodscommonly used to safety wire nuts, bolts, andscrews.
Examples 1, 2, and 5 in *Figure 6-47* show the proper method of safety wiring bolts, screws, square head plugs, and similar parts when wired in pairs. Examples 6 and 7 show a single-threaded component wired to a housing or lug. Example 3 shows several components wired in series. Example 4 shows the proper method of wiring castellated nuts and studs.

TURNBUCKLE SAFETYING

When adjustments and rigging on the cables are completed, the turnbuckles should be safetied as necessary. Only two methods of safetying turnbuckles have been adopted as standard procedures by the armed services: the clip-locking method (preferred) and the wire-wrapping method (*Figure 6-48*).

Lock clips must be examined after assembly for proper engagement of the hook lip in the turnbuckle barrel hole by the application of slight pressure in the disengaging direction. Lock clips must not be reused, as removal of the clips from the installed position will severely damage them.

Clip-Locking Turnbuckles

The clip-locking method of safetying uses a NAS lock clip. To safety the turnbuckle, the slot in the barrel must be aligned with the slot in the cable terminal by holding the lock clip between the thumb and forefinger at the end loop. The straight end of the clip should be inserted into the aperture formed by the aligned slots by bringing the hook end of the lock clip over the hole in the center of the turnbuckle barrel and seating the hook loop into the hole. Application of pressure to the hook shoulder at the hole will engage the hook lip in the turnbuckle barrel and complete the safety locking of one end. The above steps are then repeated on the opposite end of the turnbuckle barrel. Both locking clips may be inserted in the same turnbuckle barrel hole, or they may be inserted in oppositeholes.

Wire-Wrapping Turnbuckles

First, two safety wires are passed through the hole in the center of the turnbucklebarrel. The ends of the wires are bent 90 degrees toward the ends of the turnbuckle, as shown in *Figure6-48*.

Next, the ends of the wires are passed through the holes in the turnbuckle eye or between the jaws of the turnbuckle fork, as applicable. The wires are then bent toward the center of the turnbuckle, and each one wrapped four times around the shank. This secures the wires in place.

When a swaged turnbuckle terminal is being safetied, one wire must be passed through the hole provided for this purpose in the terminal. It is then looped over the free end of the other wire, and both ends wrapped around the shank.

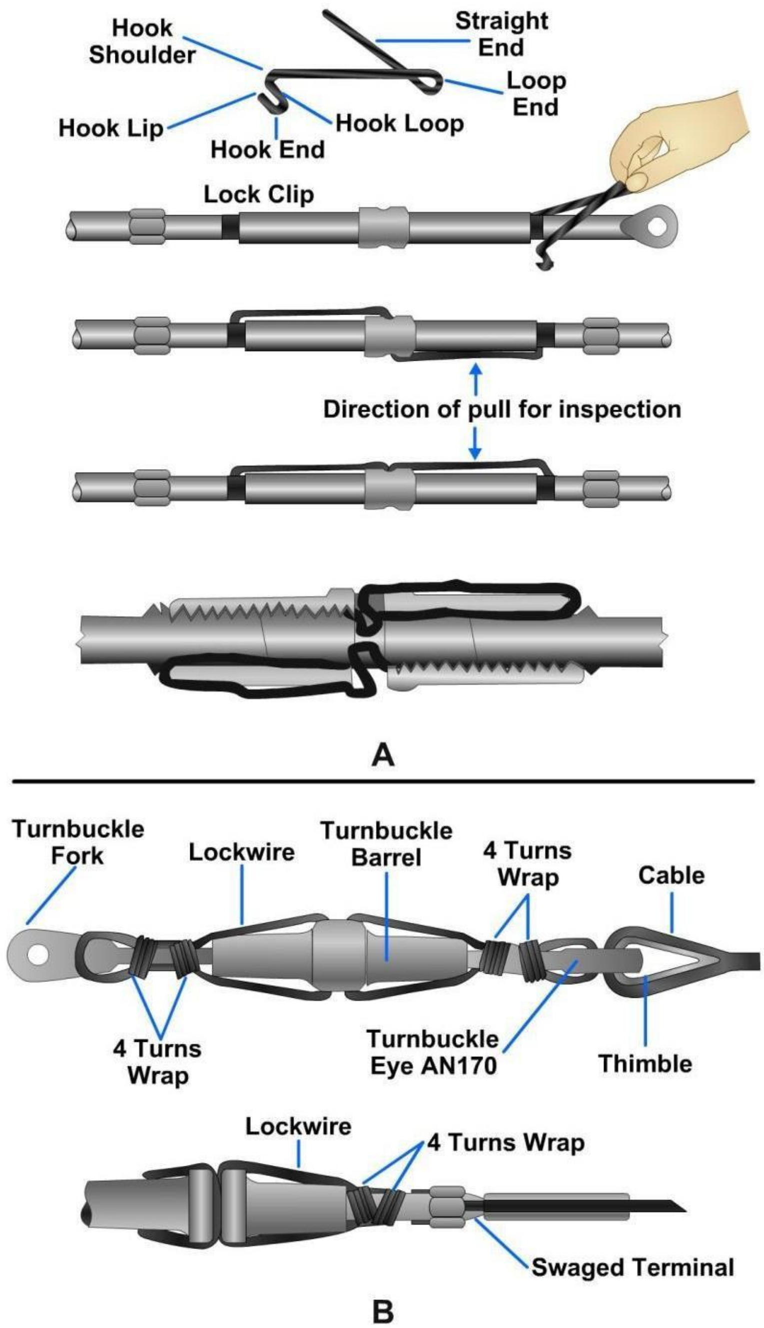


Figure 6-48 — Safetying turnbuckles: (A) Clip-locking method (preferred); (B) wire-wrapping method.

UNIT-3

DESIGN OF WINGAND TAIL STRUCTURES

1 Engineering for purpose

1.1 Safe design

The concepts and theories that underpin the field of engineering is known as *Structural integrity* – that is, the safe design and assessment of load-bearing structures in their entirety, including any individual components from which they may have been constructed. Aspects of structural integrity are implemented in almost every engineering design process, even if the engineer or designer does not necessarily think of it in that way. In this unit, we have separated the skills and knowledge associated with expertise in structural integrity under two headings: *Stress analysis*, which is the study of how applied forces lead to internal stresses in structures; and *Fracture mechanics*, which is the study of components and structures containing cracks.

1.2 Component failure

We have all experienced component failures in one form or another. In many cases this is because something has reached the end of its working life due to a slow-acting failure mechanism: car tyres wear slowly and will eventually burst if not replaced; the filament in a light bulb slowly loses material until it cannot sustain the applied voltage and melts. Failures where something has been so badly designed that it cannot withstand its intended loading during normal use are rarer, but they do occur nonetheless. Take a look at Figure 1, which shows the broken handle of a decorative cake knife, the sort that gets used only on 'special' occasions. In fact, this example of failure was caused by poor design. Note that a metal 'tang' extends from the blade into the handle as a means of reinforcement. In this case the tang was simply too short to strengthen the ceramic handle sufficiently against the bending loads that arose during cutting. The failure occurred while the knife was being used at a wedding reception and resulted in blood-soaked icing.

1.3 Environmental factors

As indicated earlier that many failures occur after a product has been in service for some time: such as the wear of a car tyre, or corrosion of the car body itself. It is also possible for components to fail because of a combination of a manufacturing defect with the applied loading or with the environmental conditions during use. Figure 6 illustrates the link from mechanisms such as corrosion, fatigue (repeated loading) and creep (continuous deformation under load) to failure in some form.

So in addition to knowing the stresses in a material arising from the applied loading, depending on the environment in which the component is used it may be necessary to consider the effects of corrosion, wear, creep and fatigue. The effects of any of these mechanisms can weaken a structure to the point where it can no longer bear the loads for which it was originally designed, as shown in Figure 7.

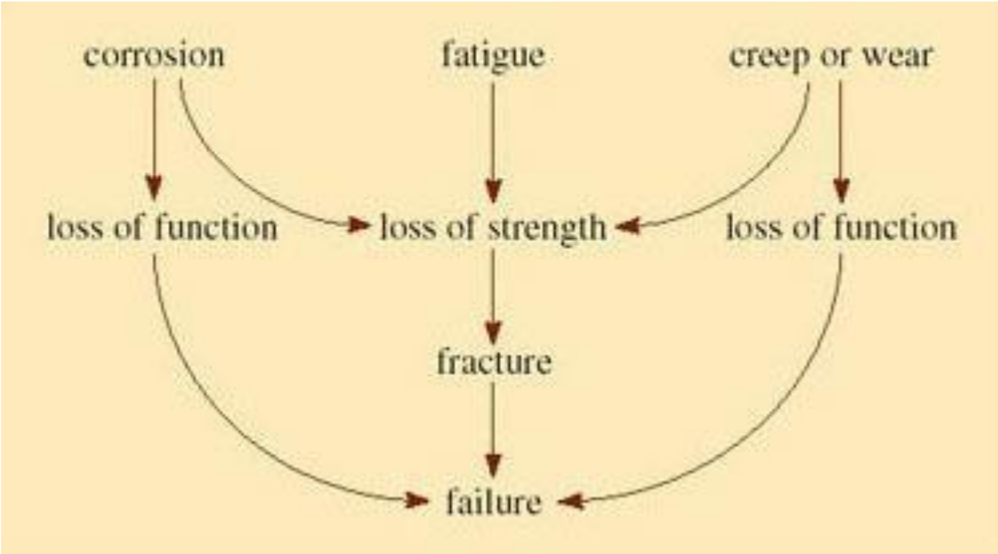
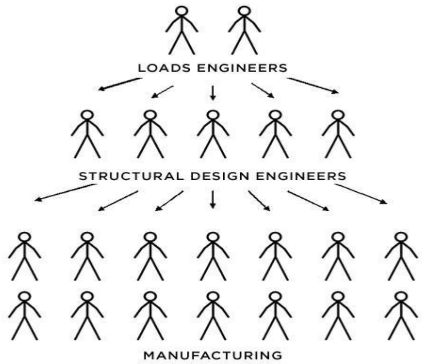


Figure 6 Routes to failure from different mechanisms

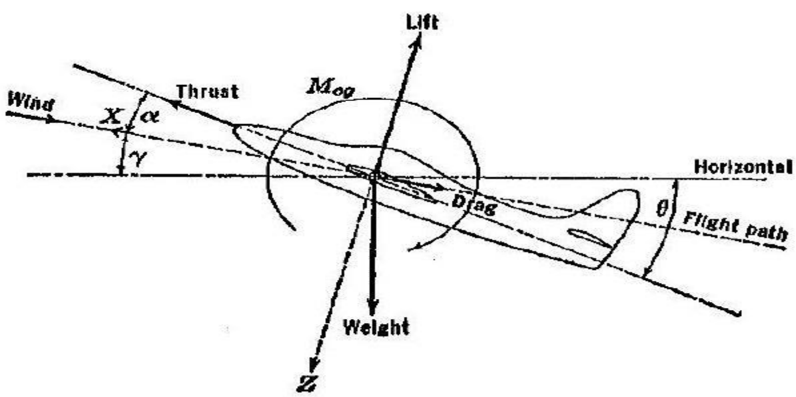
Day one, in order to avoid potential pitfalls in your aircraft design and certification process. Loads are needed at nearly all stages of a design program. Early in preliminary design, the structural designers need initial loads to size preliminary structure. As the design iterations progress, the detail and fidelity of the loads increases. The final step for an aircraft is a full set of certification loads for submission to the FAA and EASA. The work of the loads group drives
the stress and structural design groups; organizations that are often much larger than the loads group. As a result of this, changes to loads inputs after a substantial portion of detailed design is complete can necessitate a significant amount of rework. Let TLG work with your organization through the early stages of the design process in order to minimize the potential for expensive surprises in the design and certification of your project.

AIRCRAFT LOADS AFFECT MANY ENGINEERS IN THE DESIGN AND MANUFACTURING PROCESS



WHAT ARE AIRCRAFT LOADS?

Aircraft loads are those forces and moments, or loadings, applied to the airplane structural components (the wing, horizontal tail and the fuselage, for instance) to establish the required strength level of the complete airplane. These loadings may be caused by air pressure (lift), inertia (mass, weight) forces or ground reactions during take offs and landings. The determination of design loads involves a full aircraft analysis of the air pressure and inertia forces during certain prescribed maneuvers, either in the air or on the ground. TLG is capable of determining these loads for all kinds of aircraft in all manner of flight conditions.



WHAT INFORMATION IS NECESSARY TO CALCULATE LOADS?

One of the challenges in calculating loads is that the full aircraft must be accurately modeled. The primary basic data to the loads analysis are accurate airplane geometry, aerodynamic data,

weight (inertia) data, design speeds, stiffness data, miscellaneous systems data, operational data and regulations and requirements. This makes loads a multidisciplinary process. Early in a design program, these parameters can be estimated from various methods, but when using estimations, the engineer will err on being conservative, so the loads will be higher than certification level loads. As the design becomes more detailed and defined, the loads will become more concise to the final certification level.

HOW DO YOU CHOOSE THE FLIGHT CONDITIONS FOR LOADS?

The overall guidance on loads is the CFR 14 Part 23 and 25 regulations. These require the aircraft to be structurally sound for a specific set of flight conditions. The loads analysis needs to cover all possible combinations of speed, altitude, flap angle, airplane gross weight, airplane center of gravity, passenger and payload distribution, fuel quantities, engine thrust and airbrake positions for each of the required maneuver and load cases for each part of the airplane. An example of a load condition is for the pilot to suddenly roll the airplane while flying at 20,000 feet altitude, a speed of 300 knots, full engine thrust, airbrakes retracted, flaps and slats retracted, five passengers sitting in the forward most seats, half full wing fuel tanks and maximum allowed cargo. Adding up all of the possible combinations, there are thousands of unique load cases.

WHAT DIFFERS BETWEEN STATIC AND DYNAMIC LOADS?

Static loads are calculated for conditions in which the aircraft is assumed to be at steady state and range from high speed dives to low speed stalls. The dynamic loads are how the airplane responds to gusts and bumps, including landing. Dynamic loads are more prominent on large, flexible airplanes because of the structural response. For example, the gust may simply accelerate the aircraft up briefly, but the frequency of the gust will cause the wing to bend in a certain mode shape, and the resulting vibration then induces more loads on other parts of the structure due to both aerodynamic and inertial forces.

HOW MANY LOADS CASES NEED TO BE RUN?

For an aircraft, the combination of static and dynamic loads, with the range of the flight envelope, all payload loading conditions and flight maneuvers results in hundreds-of-thousands of load conditions. TLG has developed an extensive set of tools to facilitate quickly setting up and running thousands and thousands of static and dynamic loads cases and then quickly post-processing to obtain the most critical load cases to provide loads envelopes to the stress and structural engineers. TLG utilizes in-house loads programs and MSC.Nastran for loads determination.

Loads acting on a Wing

A wing is primarily designed to counteract the weight force produced by the aircraft as a consequence of its mass (the first post in this series deals with the fundamental forces acting on the aircraft). Thus during straight and level flight, the wing provides an upward lifting force equal to the weight of the aircraft plus the trim force generated at the horizontal tail to keep the aircraft balanced. The downward trim force comes about as a result of the need to balance the moment generated by the lift vector acting away from the center of gravity of the vehicle. In the conceptual design phase it is common to account for the additional force generated at the tail by multiplying the aircraft weight by a factor of 1.05 (5%) to account for the trim force; alternatively one can estimate the required force based on the estimated design weight of the aircraft and the approximate moment arm between the estimated location of the c.g. and the estimated location of the tail.

Lift is equal to weight plus horizontal tail trim force at 1g

Load Factor

An aircraft does not just fly straight and level during all phases of operation. If the pilot banks the aircraft at a 60 degree angle during a sharp turn, he needs to produce twice the lifting force to counteract the weight due to the angle of the lift vector relative to the weight (which always acts downward). In this instance, the wing is producing a lift force equal to twice the weight of the aircraft and the aircraft is said to be pulling 2g's (twice the gravitational force) or operating at a load factor of 2.

A 60 degree bank angle results in a 2g turn

The example above illustrates that there are many cases where the aircraft will exceed a loading of 1g. The Federal Aviation Administration (among other regulatory bodies) is responsible for ensuring that all certified aircraft comply to a basic standard of safety. Therefore a series of regulations are published, which among other regulations, detail the minimum load factor that a particular aircraft class should be designed to withstand.

The following extract comes from FAR 23.337:

Limit maneuvering load factors.

(a) The positive limit maneuvering load factor *n* may not be less than--

[(1) $2.1+\frac{24,000}{W+10,000}$ for normal and commuter category airplanes, where *W* = design maximum takeoff weight, except that *n* need not be more than 3.8;]

(2) 4.4 for utility category airplanes; or

(3) 6.0 for acrobatic category airplanes.

(b) The negative limit maneuvering load factor may not be less than--

(1) 0.4 times the positive load factor for the normal, utility, and commuter categories; or

(2) 0.5 times the positive load factor for the acrobatic category.

(c) Maneuvering load factors lower than those specified in this section may be used if the airplane has design features that make it impossible to exceed these values in flight.

Extract from FAR 23.337 describing the Limit Load Factor[/caption] -->

Sec. 23.337

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Amdt. 23-48, Eff. 03/11/96

Extract from FAR 23.337 describing the Limit Load Factor

The minimum design limit load factor is a function of the classification of the aircraft that is being designed. For example, it follows that an aerobatic aircraft will require a higher limit load factor than a commuter aircraft due to the difference in the severity of the maneuvers the two are expected to perform.

The extract shown above pertains to an aircraft that is to be FAR Part 23 certified which is the airworthiness standard for Normal, Utility, Acrobatic, and Commuter type aircraft. Airliners and larger commercial aircraft do not fall into the FAR 23 category and so are certified in accordance with FAR Part 25 which is the airworthiness standard for Transport Category Aircraft.

Further to specifying the maximum maneuvering load factor, the aircraft must also be designed to withstand a gust loading during level flight. Gust loading is outside of the scope of this tutorial but the reader is referred to FAR 23.341 for further information.

Limit and Ultimate Loading

The maximum maneuvering load factor specified for an aircraft design is known as the aircraft limit load.

A limit load is defined as the maximum expected load that the aircraft will see during normal operation.

It is not sufficient to design an aircraft's structure to be able to withstand a limit load as this leaves no margin of safety in the design. Limit loads are therefore multiplied by a factor of safety to arrive at a set of Ultimate Loads which provide for a safety margin in the design and manufacturing of the aircraft. The standard factor of safety for aircraft design is 1.5.

The ultimate load factor is therefore equal to 1.5 times the limit load specified in the FAR regulation.

FAR regulations stipulate that an aircraft must be able to withstand limit loads with neither any permanent deformation of the structure nor any detriment to safe operation of the aircraft.

Ultimate loads can result in plastic deformation of the structure but must be held for three seconds without failure.

Shear and Bending on a Wing

A wing is designed not only to produce a lifting force equal to the weight of the aircraft, but must produce sufficient lift equal to the maximum weight of the aircraft multiplied by the Ultimate Load Factor. So an aircraft that weighs 12 000 lbs and is designed to an ultimate load factor of 4.5 must thus be able to produce 54 000 lbs of lift up to a speed governed by the FAR regulations (dive speed). There will be a minimum speed below which the wing is incapable of producing the full 54 000 lbs of lift and this is governed by the maximum lift coefficient of the wing and resulting stall speed. In reality a V-n diagram is constructed which graphically illustrates the flight envelope of the aircraft. We wont' discuss the V-n diagram in this introductory post.

Once the maximum lifting force that wing is expected to produce has been established, the distribution of that lifting force over the span of the wing is estimated. The lift distribution over a conventional wing is parabolic in nature, rising from the tip and reaching a maximum at the root.

Lift Distribution and Bending Moment acting on a wing

This resulting vertical force distribution over the span of the wing causes the wing to flex and bend upward when it is loaded. If you look out of the window and at the wing of a modern airliner like the Boeing 787 during takeoff and landing you are sure to see a high degree of flexing. The lift produced by the wing results in a large bending moment at the wing root that must be transferred to the wingbox (the structure that connects the wing to the fuselage).

There are therefore two primary types of loading that the wing structure must be designed to withstand. A vertical shear force due to the lift generated.A bending moment arising from the lift distribution.

The wing is also subjected to torsional loads arising from the pitching moment formed by the offset between the centre of pressure and the attachment points of the wing, and horizontal (in-plane) shear forces as a result of the drag force acting on the wing. This introduction will concentrate on the vertical shear and bending moment as these loads drive the wing design.

An example of the distributed lift load and resulting shear and bending moment diagrams arising from this loading is shown below. In both cases it is clear that the location of the highest shear and bending is the wing root.

Lift distribution, Shear diagram and Bending Moment Diagram of a typical wing

Wing Structural Components

The primary objective of the wing's internal structure is to withstand the shear and bending moments acting on the wing at the Ultimate load factor. The secondary objective is to make the wing as light as possible without compromising the structural integrity of the design as described above.

An optimized wing design will fail just as the ultimate loading conditions are reached.

There is no need to make the wing any stronger than it needs to be, and any excess strength (wing weight due to extra material) will reduce the payload capacity of the aircraft making it uncompetitive or uneconomic to operate. In reality the wing will be analysed using computational methods for many different loading combinations that exist at the edge of the aircraft design envelope and then subjected to a static test at the ultimate load factor to show that failure will not occur below the ultimate load.

An aircraft wing is usually designed with a semi-monocoque approach where all the components making up the wing structure are load bearing. A typical semi-monocoque wing structure is shown below with the various components labelled:

Typical structural arrangement of a semi-monocoque wing showing the various components labelled

Spar Cap (flange):

These consist of the upper and lower flanges attached to the spar webs. The spar caps carry the bending moment generated by the wing in flight. The upper spar cap will be loaded in compression and the lower in tension for a positive load factor (wing bending upward). The spar caps also form a boundary onto which wing skin is attached and support the wing skin against buckling. Concentrated load points such as engine mounts or landing gear are attached to the main spar.

Spar web:

The spar web consists of the material between the spar caps and maintains a fixed spacing between the them. This allows the spar caps to act in pure tension and compression (bending) during flight. The spar web is responsible for carrying the vertical shear loads (lift) which arises from the aerodynamic loading of the wing. The spar webs and caps are collectively referred to as the wing spar.

Wing Ribs:

The ribs are spaced equidistant from one-another (as far as is practical) and help to maintain the aerodynamic profile of the wing. The ribs form part of the boundary onto which the skins are attached, and support the skins and stiffeners against buckling. Ribs also form a convenient structure onto which to introduce concentrated loads

Stringers/Stiffeners:

Stiffeners or stringers form a part of the boundary onto which the wing skin is attached and support the skin against buckling under load. The stiffeners also carry axial loads arising from bending moments in the wing.

Skin:

The wing skin transmits in-plane shear loads into the surrounding structure and gives the wing its aerodynamic shape.

Analysis Methods

What follows is a brief introduction into some methodologies and analyses typically carried out during the design of a new wing structure. We will not go so far as to look into the specifics of the mathematics used, but will discuss the preliminary structural layout of the wing and look at two analysis methods that drives the structural design: a shear flow analysis and a collapse moment analysis.

Preliminary Structural Layout

Before the structural layout of the wing is designed, a preliminary sizing of the wing planform should have been completed to size the wing for its required mission. If you have been following along from the start of this series then you'll be familiar with sizing a wing with respect to plan area and aspect ratio, sweep and supersonic flight, and selecting a suitable airfoil profile in order to complete the planform design of the wing.

Once the planform is frozen, a preliminary structural layout should be drawn up using the following rules of thumb:

Generally the main spar is located at or near the 25 % chord location. The aerodynamic center of the wing exists at approximately quarter chord which is the location on the wing where the moment coefficient is independent of angle of attack. It is good design practise to locate the main spar near the aerodynamic centre.

A rear spar is often required in order to attach the trailing edge flap and aileron surfaces to the main wing structure. If the surfaces have already been specified during the conceptual phase (before the structural design is started) then these surfaces will form a natural constraint and drive the placement of the rear spar.

Ribs will need to be placed at any points in the wing where concentrated loads are introduced. Common examples such as engine pylons, landing gear, and flap and aileron junctions should guide the placement of the first few ribs.

Additional ribs should be placed equidistant along the span of the wing such that the aspect ratio between the ribs and the skin remains close to one. This aids in unloading the shear in the skin and reduces the tendency for the skins to buckle.

Stringers can be added between the spars. This will aid the skin in resisting shear buckling.

Example of a preliminary structural layout for a rectangular untapered wing

Structural Idealization

In order to efficiently analyse the wing structure, a number of simplifying assumptions are typically made when working with a semi-monocoque structure.

The spar caps/flanges and stiffeners only carry axial (bending) loads.The skins and spar web only carry shear loads.

This is the classical approach to aircraft structural design and will result in an efficient structure that has been sized with conventional methods which are well accepted by the certification authorities. However, improvements in computing power along with the rise of composite materials in structural design means that there is a gradual movement away from the classical methods to analyzing the structure in such a way that seeks to further optimize the design to produce the lightest possible structure. We'll just focus on the classical methods for the sake of this tutorial.

Shear Flow Analysis

Examining the mathematics behind a shear flow analysis is outside of the scope of this introductory tutorial; rather the methodology and rationale will be discussed.

A shear flow analysis is used to size the thickness of the wing skin and shear webs. A shear force diagram is determined at the maximum load factor which then serves to specify the variation in shear force along the span of the wing. The variation in shear force along the span forms the input into the calculation as the shear at each spanwise location must be transferred into the wing structure.

Based on the assumption that the skin and web only transmits shear and no axial load, the shear stress within a skin panel will remain constant where ever the thickness of the skin is constant.

The product of the shear stress and the thickness is therefore constant along a skin and is termed shear flow.

$q=\tau\times tq=\tau\times t$

A panel section of the wing can therefore be modelled as a set of skins where thickness is a variable, and once the shear flows acting on each of the skins are known, the thickness of the skins can be varied until the shear stress in each skin is below the material allowable shear stress. In this way, the wing skins and web will not fail as a result of the shear loading induced when the aircraft operates at the edge of the design envelope. The final skin shear flows are also a function of the spar cap area, and this can also be varied to manipulate the final shear flows.Shear flow analysis on a simple box beam wing structure

Collapse Moment Analysis As with the shear flow analysis, the mathematics behind this calculation are complex and outside of the scope of this tutorial. Instead we briefly introduce the rationale behind a collapse moment analysis.

As described above, a shear flow analysis is used to size all the shear components of the wing structure (webs and skins). We now examine the bending components of the design; namely the spar cap areas and the propensity of the skins on the upper surface of the wing to buckle under compression at high load factors.

The spar caps are responsible for transferring the bending moment generated by the wing into the surrounding structure. When the wing is subjected to a positive load factor it will tend to deflect upward and load the upper spar caps and skin in compression, and the lower structure in tension.

Wing in bending (positive load factor) loads the upper skins in compression and lower skins in tension

A collapse moment analysis examines the interaction between the wing skin in compression (which will tend to buckle) and the ability of the spar caps to absorb the extra load transferred if the skins do buckle. Buckling of the skin does not necessarily result in failure of the whole wing structure as the buckled skin will transfer load into the spar caps and stiffeners that border the skin. This transfer is accomplished through shear flow.

The wing will fail when the stress in the stiffeners or spar caps reach their maximum crippling (failing) stress. The problem becomes an iterative one as the stress at which the skin first starts to buckle must be determined, which in turn affects how much additional load is transferred into the spar caps.

The critical bending moment at which the spar cap/stiffener will reach its critical stress and fail is a function of the cross-sectional area of the stiffener and also the distance that the stiffener lies from the neutral axis. The position of the neutral axis is in turn a function of the extent to which the skins have buckled on the application of the maximum load.

The moment at which the structure will collapse is determined once the crippling stress (critical stress in spar cap) and the moment of inertia (function of extent to which skins have buckled) is known.

This collapse moment is then compared to the bending moment diagram generated for the wing to ensure that the bending moment applied is lower than the collapse moment at all spanwise locations of the wing.

The wing will not fail in bending if the collapse moment is greater than the bending moment at all spanwise locations

Since the bending moment is a maximum at the root of the wing, the spar caps will need to be large enough (sufficient area) so as not to fail in bending. Using a constant sparcap area from root to tip would result in a situation where the applied bending moment is very much smaller than the collapse moment as one moves toward the tip. This would result in an inefficient structure which is overly heavy. One way to mitigate this is to taper the spar cap area as one moves toward the wing tip in such a manner that weight is reduced but the collapse moment is always greater than the applied moment at all points along the wing. An example of a structural layout where a tapered main spar flange is used

Wrapping Up

Completing the full structural design of a new wing is a complex and iterative process. The analysis described above just represents a small part of the design and stress analysis process. A wing structure would be modeled using a Finite Element (FE) package and tested for many different load combinations before a prototype is built and tested to the point of destruction as a means to validate the paper calculations and computer analysis. However, starting with some hand calculations, similar to those shown above is a good way to begin the design process as it ensures that the engineer understands the resulting load paths before creating an FE model.

UNIT IV
DESIGN OF FUSELAGE AND LANDING GEAR

DESIGN OF FUSELAGE AND LANDING GEAR

The main body section of an aircraft is called a fuselage. This forms the central body of the aircraft onto which wings, control surfaces and sometimes engines are connected. The fuselage houses the crew, any passengers, cargo, an array of aircraft systems and sometimes fuel.

A well designed fuselage will ensure that the following are met:

- The intended payload is adequately and efficiently housed.
- The fuselage is sized such that the various control and stabilization surfaces (typically the vertical and horizontal tail) are located such that the aircraft is stable in flight.
- Loading the aircraft with goods, fuel and passengers does not negatively impact on the stability of the aircraft for a range of payload configurations (center of gravity is adequately located).
- The fuselage structure will not fail due to excessive loading throughout the entire aircraft flight envelope.
- The mass of the fuselage is optimized to ensure safe operation without carrying any additional or excess weight.
- The aerodynamic shape of the fuselage is such that the minimum drag is produced during typical operation while still ensuring that the design payload is adequately housed.
- The fuselage design is versatile enough to offer the potential to stretch the aircraft if a number of aircraft configurations are desired.

Let’s start by examining three popular design methodologies for the structural design of a fuselage.

Structural Design Principles

Throughout the years a number of design principles have been adopted regarding the structural layout of a fuselage. Three common design methodologies are described below in chronological order leading up to the semi-monocoque design that is most prevalent today.

Space Frame (Truss)

The earliest aircraft fuselages were built with a space frame or truss like construction. Often wood was used as the primary structural material with a fabric covering providing the aerodynamic shape. **In this fuselage configuration the force members of the truss provide the structural stiffness, and the aerodynamic covering provides the shape, but does not add much to the overall stiffness of the structure.**

A space frame is a simple albeit inefficient way of building a fuselage structure as the fabric skins add weight without contributing to the rigidity of the structure. One popular aircraft designed with a space frame fuselage is the iconic PA-18 Piper Super Cub which is pictured below.

Monocoque

By the end of the First World War limitations in the the use of wooden truss configurations were being identified. As the flight speed and wing loading of newer designs increased, the variation of the structural properties of the wood and its susceptibility to environmental degradation meant that wooden structures were no longer an efficient means of production. New methods were sought and steel was investigated as a replacement for wood. Steel is stiff and strong (both prerequisites in the design of an efficient structure) but its high density makes it very heavy (density of wood approximately 500 – 800 kg/m3kg/m3 vs steel 7800 kg/m3kg/m3). To efficiently design with steel, engineers had to make use of very thin sections which were intricately curved and shaped to prevent buckling of the thin structure. **The term monocoque structure refers to a structural arrangement where the skins take all of the loading and contribute to all of the structural rigidity of the design.** One major downfall when designing a pure monocoque structure is the difficulty of incorporating concentrated loads into the structure such as engine mountings or the wing-fuselage interface. The distribution of these point loads into the skin structure becomes very difficult to efficiently achieve. Interestingly, in recent times the introduction of composites as a material from which to build aircraft structures has seen a move back towards designing a pure monocoque structure, although typically a hybrid design of a metallic substructure with composite skin panels is typically used on larger composite aircraft.

Semi-Monocoque

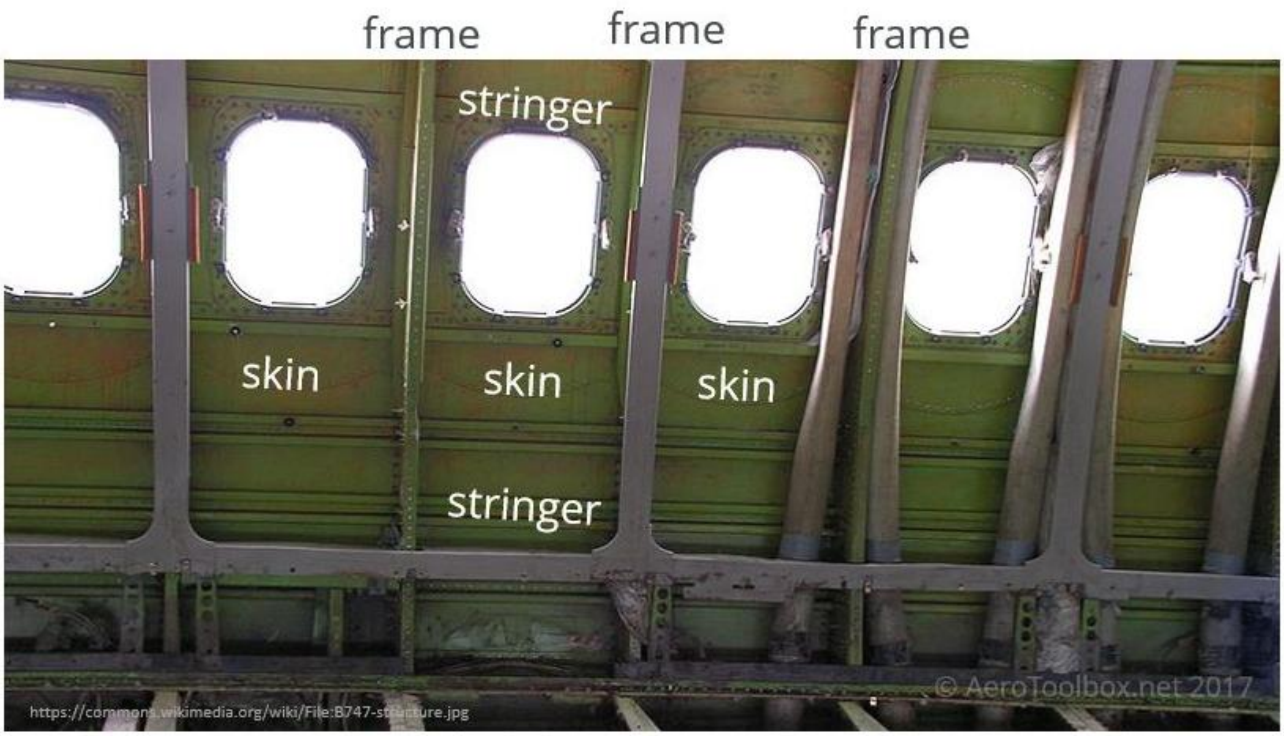
Somewhere between the space frame arrangement (skin takes no load) and pure monocoque arrangement (skin takes all the load) lies the semi-monocoque design which is the most common method of constructing aircraft structure today. **In a semi-monocoque structure both the skin and set of frames are load carrying and contribute to the overall stiffness of the structure.** This design methodology was born out of the use of aluminium rather than steel as the primary structural material used in the design of aircraft structures. Aluminium has many advantages over steel, principally its density is approximately one-third that of steel. For a constant structural mass, the aluminium sections can be thicker which reduces the susceptibility of those skins to buckling, which in turn produces a more efficient structure.

The internal structure of an ATR-72 showing a semi-monocoque construction

A semi-monocoque fuselage therefore typically consists of the following structural components:

- **Frames** – these transverse elements are built in the shape of the fuselage cross-section and are typically spaced approximately 20 inches or 50 cm apart.
- **Stiffeners/stringers** – the frames are joined to one-another by longitudinal stiffeners or stringers.
- **Skins** – the skin is load-bearing and gives the fuselage its form and shape. The skin is attached to the sub-structure on an aluminium aircraft by riveting it to the frames and stringers.

Semi-monocoque structures are the predominant way in which aircraft are designed and so the rest of this tutorial will focus on the application of semi-monocoque structures.



Semi-monocoque fuselage structure as seen on a Boeing 747-230

Fuselage Loading

A fuselage structure is loaded in a number of ways. These include:

- Aerodynamic loads as a result of the aircraft maneuvering through the air.
- The distribution of the mass of the fuselage induces bending
- Inertial loads created by point masses connected to the fuselage (engines attached to fuselage by a pylon is an example).
- Concentrated point loads: for example the interface between the fuselage and the tail.
- Internal pressurization loads (if the aircraft is pressurized).
- Shock loading: for example the nose landing gear impacting the runway on landing.

The loading experienced by the fuselage is likely a combination of each of these at a given moment in time. How then do each of the structural elements present in a semi-monocoque fuselage structure work together to distribute and transfer the resulting loading?

The **frames** work to support the skins and stiffeners against buckling while retaining the aerodynamic shape of the fuselage. Frames are also used wherever concentrated loads are introduced into the structure, for example at the wing-to-fuselage interface, and the tail-to-fuselage interface. Finally, frames are also used in conjunction with the skin to resist the internal pressure formed when an aircraft is pressurised.

Stiffeners and stringers are responsible for transmitting the axial loading (both tension and compression) that arise out of the bending moments induced through the fuselage structure. A good example would be the bending moment generated through the fuselage when applying a rudder input during flight. The stiffeners also assist in preventing the fuselage skin from buckling.

Finally the **skins** transmit shear loads and work to introduce load into the stiffeners. The skins also resist the internal pressure which is present in a pressurized aircraft.

To summarize:

Axial loads are carried by the longitudinal stiffeners and stringers

Shear loads are carried by the skin

The basic methodology behind the structural design of a fuselage is to ensure that the skin/stiffener combination does not buckle between transverse frames. Therefore the frames must be stiff enough that they do not buckle globally, and the skin and stiffeners, which form a series of segments on the fuselage, must not buckle locally. An optimized fuselage design results when these conditions are met for the lightest possible structure.

Fuselage Sizing

Let’s move on from the various structural elements that are required to design a fuselage onto how you determine the size and shape of the fuselage required for your aircraft design.

A good starting point is to thoroughly understand the requirements of the aircraft that you are designing; here are some questions you should ask yourself:

- What does my payload look like? Am I designing a passenger aircraft, carrying cargo or munitions?
- How is my aircraft powered? Do I need to include space in the fuselage for an engine or will the engines sit externally on the wing or towards the rear of the fuselage?
- What does a typical mission for my aircraft look like? Am I more interested in achieving a high cruise speed at the expense of payload or is the size and extent of the payload the driving requirement?
- Is the aircraft to be pressurized or unpressurized? Pressurized aircraft generally have cylindrical fuselage cross-sections as this is the most efficient shape to resist the internal pressure.

Once you are clear which factors will drive the fuselage design you can start to sketch a preliminary outline of your fuselage. It is useful to start by first placing all the components that you know your fuselage will need to house e.g. engines, passengers, cargo etc, and then shaping the fuselage around these. Generally it is good to start by creating a number of cross-sections of your proposed fuselage over the critical components and then begin to join them up to form your preliminary design. Of course it is also very important to consider the location of the center of gravity of your fuselage and internal components, as the location of the aircraft C.G relative to the center of lift of the wing is a critical stability criterion.

The placement of the wing and tail surfaces will also drive the total length of the fuselage both from a stability and a controllability standpoint. A longer fuselage means that the tail surfaces can be made smaller since the moment arm between the aircraft C.G and the aerodynamic center of the horizontal and vertical tail surfaces is increased, which increases the effectiveness of the control surfaces. This is very nicely illustrated when comparing the size of the vertical tail on the shorter Boeing 747SP to that of the 747-400.

Comparison of the size of vertical tail on Boeing 747SP and 747-400

Typically the fuselage contributes somewhere between 20 and 35 % of the total drag produced by the aircraft at cruise and this is a function of three key variables:

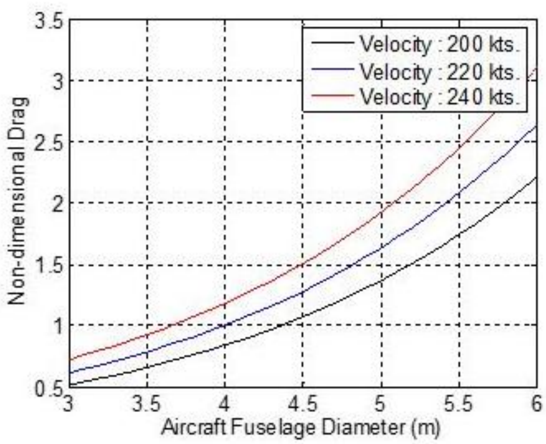
- The maximum cross-sectional area of the fuselage.
- The fuselage slenderness ratio (ratio of length-to-diameter).

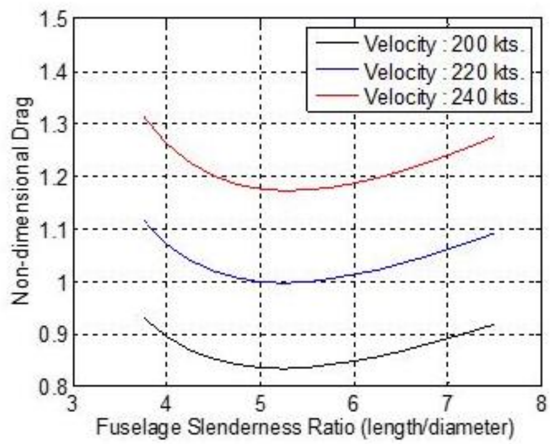
- The total wetted area of the fuselage.

The effect that these variables have on the total profile drag of the fuselage is shown in the plots generated below:

An increase in fuselage diameter from 4 m to 5 m produces an increase in fuselage profile drag of 60 %. This illustrates just how important it is to size your fuselage in order to fit your intended payload but not make it unnecessarily larger.

The length of your fuselage should be sized according to the maximum cross-sectional area. A slenderness ratio (length-to-diameter) of between 5 and 6 produces the minimum drag case.





Plots of Fuselage Diameter and Slenderness Ratio against Drag (Normalized)

The location of doors and windows on the fuselage is another important consideration that must be carefully studied. The locations of the windows for example will affect the locations of the transverse frames through the fuselage. Anybody who has flown on a commercial airliner would argue that the locations of the windows are driven by the placement of the frames and not the other way around! The type, size, and minimum number of doors and emergency exits placed on the fuselage is specified by the regulations published by the Federal Aviation Authority. This is driven by the need to quickly and efficiently evacuate passengers in the event of an emergency. Doors and windows form cut-outs on the fuselage structure which requires additional reinforcement of the structure around these openings. This is turn results in a heavier structure and so the size and number of cut-outs should be kept to a minimum.

One additional but important consideration is the design of the cockpit. Pilot visibility is a major consideration (at least while commercial aircraft remain piloted) and the cockpit must be sized in order to allow the pilots to safely operate the aircraft at all times during flight. The approach and landing phase of the flight are the most critical from a pilot visibility perspective. During landing the pilot will pitch the aircraft nose up to increase the angle of attack of the wing in order to fly at a slower speed. Visibility of the runway at this attitude is an important factor that must be considered. Delta winged aircraft like the Concorde land at very high angles of attack, which is why the Concorde nose rotates downward during landing to allow the pilots to see the runway over the aircraft's nose.

Concorde just prior to touchdown with forward nose rotated downward

Introduction

Another aircraft major component that is needed to be designed is landing gear (undercarriage). The landing gear is the structure that supports an aircraft on the ground and allows it to taxi, take-off, and land. In fact, landing gear design tends to have several interferences with the aircraft structural design. In this book, the structural design aspects of landing gear are not addressed; but, those design parameters which strongly impact the aircraft configuration design and aircraft aerodynamics will be discussed. In addition, some aspects of landing gear such as shock absorber, retraction mechanism and brakes are assumed as non-aeronautical issues and may be determined by a mechanical engineer. Thus, those pure mechanical parameters will not be considered in this chapter either. In general, the followings are the landing gear parameters which are to be determined in this chapter:

1. Type (e.g. nose gear (tricycle), tail gear, bicycle)
2. Fixed (faired, or un-faired), or retractable, partially retractable
3. Height
4. Wheel base
5. Wheel track
6. The distance between main gear and aircraft cg
7. Strut diameter
8. Tire sizing (diameter, width)
9. Landing gear compartment if retracted
10. Load on each strut

Landing gear usually includes wheels, but some aircraft are equipped with skis for snow or float for water. In the case of a vertical take-off and landing aircraft such as a helicopter, wheels may be replaced with skids. Figure 9.1 illustrates landing gear primary parameters. The descriptions of primary parameters are as follows. Landing gear height is the distance between the lowest point of the landing gear (i.e. bottom of the tire) and the attachment point to the aircraft. Since, landing gear may be attached to the fuselage or to the wing; the term height has different meaning. Furthermore, the landing gear height is a function of shock absorber and the landing gear deflection. The height is usually measured when the aircraft is on the ground; it has maximum take-off weight; and landing gear has the maximum deflection (i.e. lowest height).

Thus, the landing gear when it has the maximum extension is still height, but is less important and application. The distance between the lowest point of the landing gear (i.e. ground) to the aircraft cg is also of significant importance and will be employed during calculations. Wheel base is the distance between main gear and other gear (from side view). The landing gear is divided into two sections: 1. Main gear or main wheel[1], 2. Secondary gear or secondary wheel. Main gear is the gear which is the closest to the aircraft center of gravity (cg). During the landing operation, the main wheel touches first with the point of contact to the ground. Furthermore, during the take-off operation, the main wheel leaves the ground last. On the other hand, main gear is carrying great portion of the aircraft load on the ground.

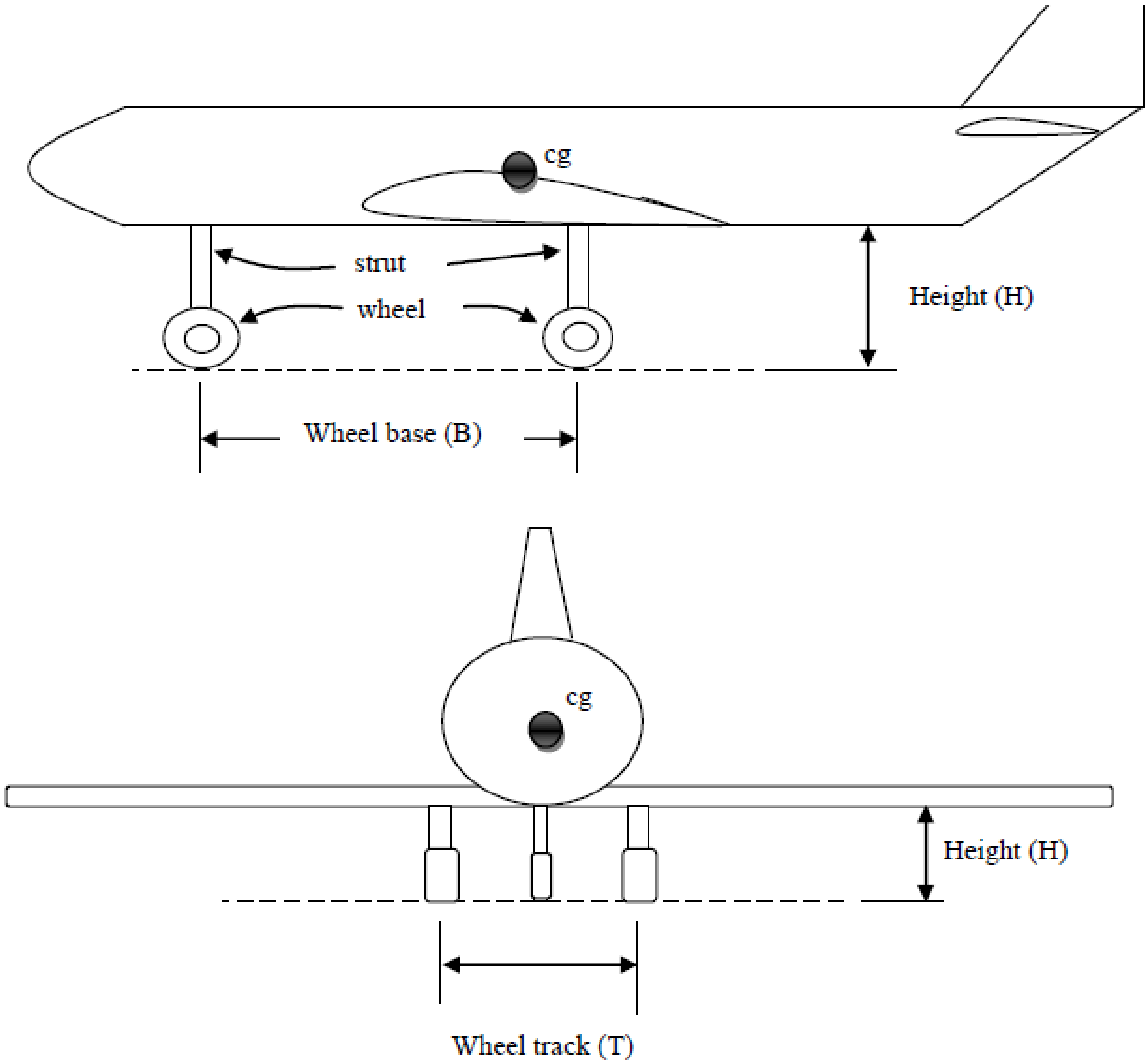


Figure 9.1. *Landing gear primary parameters*

In order to allow for a landing gear to function effectively, the following design requirements are established:

1. Ground clearance requirement
2. Steering requirement
3. Take-off rotation requirement
4. Tip back prevention requirement
5. Overturn prevention requirement
6. Touch-down requirement
7. Landing requirement
8. Static and dynamic load requirement

UNIT V

FATIGUE LIFE, DAMAGE TOLERANCE, FAIL SAFE- SAFE DESIGN-WEIGHT CONTROL AND BALANCE

Fail-Safe and Safe-Life Designs

FATIGUE LIFE, DAMAGE TOLERANCE, FAIL SAFE- SAFE DESIGN-WEIGHT CONTROL AND BALANCE

Fail-Safe and Safe-Life Designs

And Factor of Safety

Factors of Safety (a.k.a. Safety Factor)

The factor of safety is usually expressed as a ratio of the "load carrying capability" of the structure to the expected loading. Loading may be static, impact, fatigue, wear, et cetera. The purpose of using a safety factor is to assure that the design does not fail in the event of unexpectedly high loads or the presence of material/design defects. Factors of safety are applied to decrease the probability of failure, or in more positive terms, they increase the probability of success. They are applied in part due to inherent ignorance present in all designs. Ignorance stems from natural variability in materials and manufacturing processes, maintenance, and what the design really experiences in its lifetime. Lower factors of safety may be required if the following are true, larger ones are justified if these are less true:

High quality and consistency of materials, manufacturing, maintenance and inspection

Good control or knowledge of the actual loads and environment

Highly reliable analysis and/or experimental data

The commercial airplane business has extremely rigorous control over airplane structures and systems from fabrication and assembly through inspection and maintenance. The environmental effects and maximum loads airplanes experience are also well understood. Extensive fatigue and static testing is conducted on components and systems. Therefore, relatively low factors of safety are applied (around 1.3) even though safety is at stake. The degree of ignorance is not the only element that the engineer should use to determine appropriate factors of safety. The potential harm that failure can produce is also important. If failure would result in a mere inconvenience, then a small factor of safety may be acceptable. If failure would be expensive or even life threatening, then a larger factor of safety is justified.

How does an engineer determine an appropriate factor of safety? In some instances, such as pressure vessels, minimum factors of safety are mandated by codes and standards. But this is not often the case. Experience with similar designs is often the best method. Typically, factors of safety range from a low of 1.3 to around 5.

Fail-Safe and Safe-Life Designs

Aerospace engineers, for designs involving fatigue loading, developed safe-life and failsafe philosophies. The concept of fail-safe designs is extended here to include all designs that mitigate the harm caused by failure.

What is meant by "Fail-Safe"?

Fail-safe designs are designs that incorporate various techniques to mitigate losses due to system or component failures. The design assumption is that failure will eventually occur but when it does the device, system or process will fail in a safe manner.

What is meant by "Safe-Life"?

Safe-life refers to the philosophy that the component or system is designed to not fail within a certain, defined period. It is assumed that testing and analysis can provide an adequate estimate for the expected lifetime of the component or system. At the end of this expected life, the part is removed from service.

When should either of these philosophies be employed?

The benefit of safe-life designs includes reducing the likelihood of unplanned maintenance and reducing the likelihood of any failure. Benefits of fail-safe designs include being able to manage the unexpected and mitigating damage if failure occurs. There is no method to help determine which if either of these philosophies should be employed. Engineers must use their judgment on a case-by-case basis. The decision to use either of these philosophies is justified whenever the "cost" and likelihood of failure outweighs the "cost" of implementing either fail-safe or safe-life designs.

"Cost" of failure may include:

Physical harm to people or the environment

Loss or destruction of property or equipment

Loss of productivity or use of the failed "system" or device

Damaged reputation

Likelihood of failure

The engineer should always consider how likely a certain failure will be. In so doing, it is important to consider all potential loading conditions – even abusive loads.

"Cost" of implementing can include:

Increased expense and time for design and testing

Increased production costs

Decrease in product performance

There are no formulas to help determine when fail-safe or safe-life designs should be employed. Airplane designs employ both of these concepts, making air travel one of the safest modes of transportation. Yet, it is not possible to make aircraft completely safe. There are always conditions that are prohibitive to guard against.

Techniques for Safe-Life Design

Since it is imperative that the component or system not fail within the predicted life time, extensive testing and analysis is required. Safe-life designs involve a testing and analysis (typically fatigue analysis) to estimate how long the component can be in service before it will likely fail. Since no amount of analysis and testing can assure how long a particular part will perform without failure, a generous factor of safety should be included to prevent catastrophic failure. The product should be designed so that it can be easily inspected in service.

Techniques for Fail-Safe Design

Redundancies (avoid single point failures)

Back-up systems –If failure of a critical subsystem will cause severe losses, backup systems are often employed. For example, commercial aircraft have a minimum of two engines. They are designed such that fully loaded airplanes can takeoff even if one engine fails.

Multiple load paths – if a structural element fails, the load it was carrying will be transferred to other members. Obviously, it is essential that the fracture be detected before multiple members fail.

Intentional "Weak Link"

An inexpensive and easy to replace component may be used to prevent damage to expensive or difficult to repair component. Fuses in electrical circuits are an example of this for electrical systems. Shear pins are used on boat propellers are a mechanical example. These are inexpensive and easy to replace pins that transmit power from the shaft to the propeller. If the propeller strikes an object, the shear pin is designed to fail before the propeller or shaft are damaged.

Physical Law

Designing a system in such a way that failure cannot be catastrophic based on how failure will occur. For example, nature gas pipelines are produced from sufficiently tough material so that it will fail in a ductile manner, rather than brittle. Ductile fractures propagate at about 600 ft/sec. Brittle fractures propagate at about 1500-2500 ft/sec. When a crack forms in a pipe, the gas will immediately begin to decompress. The decompression wave will travel down the pipe at about the speed of sound (1300 ft/sec). If the crack speed is faster than the decompression speed, the crack front will always remain under high pressure and the crack will grow indefinitely. Otherwise, the decompression wave will out run the crack, and the crack will stop growing.

Early Detection

When a structure is designed such that cracks will easily be detected before they reach critical length, it may be considered a fail-safe design. A critical element of this is the detection of the crack before it reaches critical length. It is very important that proper materials (high fracture toughness) be selected that can withstand large cracks before fracturing.

Fracture mechanics must be used:

Determine minimum detectable crack length (how small of crack can nondestructive testing detect)

Determine critical crack length for the maximum load

Create a crack growth curve showing crack length as a function of number of cyclic loads

Determine how much time is required from the crack to grow from the minimum detectable length to critical length.

Leak-before-break – pressure vessels use this method to prevent explosive failures. Pressure vessels are designed such that a crack will propagate completely through the vessel before it reaches critical length. Generally, the cracks will start at the internal wall and progress outward, radially. Leaks are generally easy to detect, and therefore, should be detected before the crack grows to critical length. See Figure 1

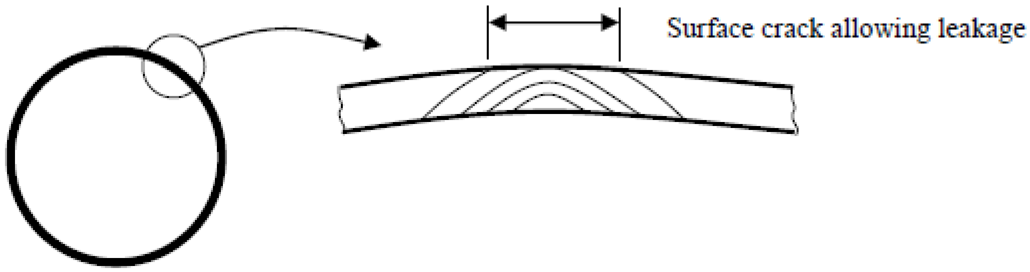


Figure 1 – Leak-before-break in a pressure vessel.

Figure 1 – Leak-before-break in a pressure vessel.

Crack Arresters – to prevent cracks that exceed critical length from fracturing the entire part, crack arresters may be added to the structure. In aircraft these are in the form of riveted straps added to the skin. This will contain the crack to a small area of the structure. See Figure 2.

Effectively, what is occurring is the crack tip stress intensity decreases as it approaches the arresters. The arresters start to carry more and more load, thus decreasing the load near the crack tip.

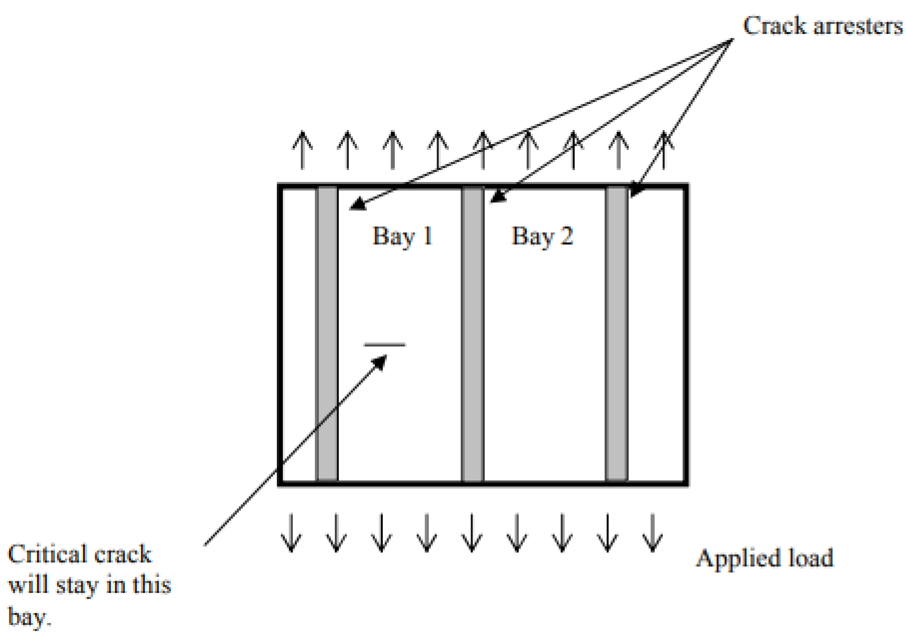


Figure 2 – crack arresters preventing extensive crack growth in a panel with axial loads.

Figure 2 – crack arresters preventing extensive crack growth in a panel with axial loads.