021 AIRFRAME & SYSTEMS

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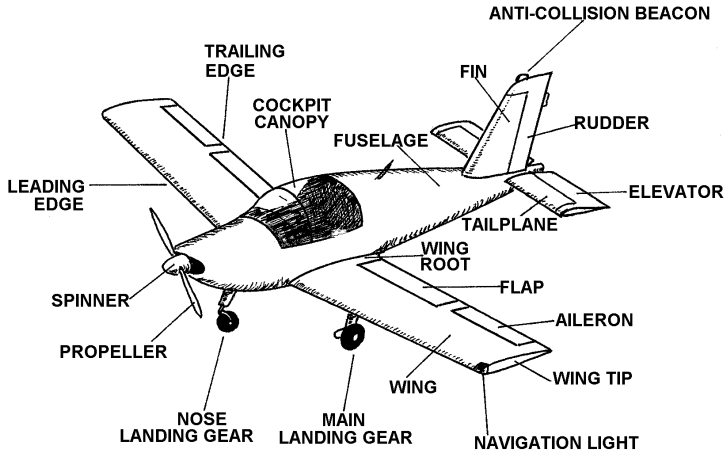
Aircraft Structures

1. The structure of the aircraft is known as the airframe and usually comprises five major units. These are the fuselage, the wings, the stabilising surfaces (fin and tailplane), the landing gear and the flying control surfaces. These major components, plus many others, are illustrated at Figure 1-1.

FIGURE 1-1
Aircraft Structure Major Components



2. Each of these airframe components must be sufficiently strong to withstand the forces acting upon it during all stages of flight without distortion or failure. Also, they must be joined together by bolts, screws, rivets, welding and so forth. Whichever method or combination of methods is used, it must be of sufficient strength to withstand the loads to which these junctions will be subjected.

3. The wings support the aircraft in flight so they must be made of materials that are strong enough to withstand the aerodynamic forces, without bending excessively or twisting. However, as these forces vary at different flight speeds or during turbulence, the wings must be able to flex. The same applies to the junction between wings and fuselage.

4. When the rudder or elevators are used, the forces acting upon them tend to twist or bend the fuselage, which must be strong enough to resist this. Similarly, it is important that the wings, whilst able to flex up and down, do not twist when the ailerons are used.

5. When the elevators are deflected up or down there is a twisting force (torque) applied to the horizontal stabiliser and its attachment to the fuselage. Both must be strong enough to resist this twisting force, but the stabiliser must be supple enough to flex, or bend, otherwise it might snap like a dry twig. The same requirements exist for the fin, when the rudder is deflected left or right.

6. The landing gear must be strong enough not only to support the weight of the aircraft on the ground, but also to withstand the shock of landing, the twisting loads when the aircraft turns during taxiing and the bending loads at touchdown. All this applies equally, of course, to the points of attachment of the landing gear to the airframe.

7. The aircraft designer must consider all these factors and produce an aircraft constructed of materials strong enough to withstand all of the loads to which the airframe will be subjected. The aircraft must be flexible where necessary to absorb changing loads and rigid where necessary to prevent twisting.

8. Having calculated the maximum anticipated loads the designer arrives at a compromise, which gives sufficient structural strength but keeps airframe weight to a minimum. This normally ensures that each of the various parts of the structure is designed to fail at an ultimate load that is 1½ times greater than the maximum applied load. The ratio of ultimate load to maximum applied load (1.5:1) is known as the safety factor.

Stress

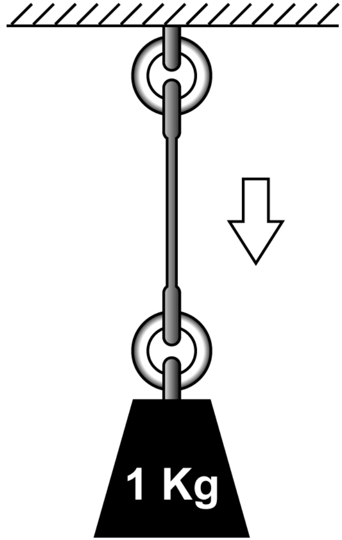
9. The application of force to a given area of material induces stress within that material. This stress will cause the material to change its shape, or deform, and this is called strain.

10. The stresses that act upon the component parts of the airframe are tension, compression, bending, torsion and shear.

Tension

11. Tension is the stress that resists the forces tending to pull a material apart. This is illustrated at Figure 1-2. The cable supporting the weight is in tension, or is being subjected to tensile stress.The pylon from which an under-wing engine is slung is in tension when the aircraft is stationary on the ground with the engine stopped.

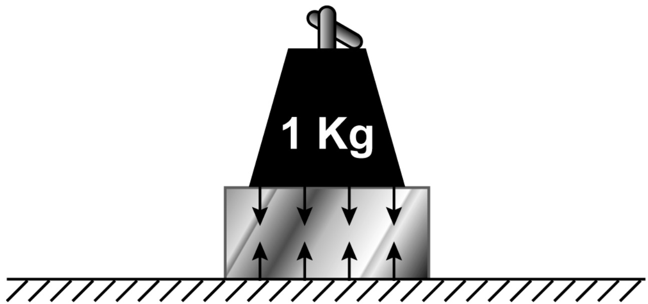
FIGURE 1-2
Tension



Compression

12. Compression is the stress that resists a crushing or squeezing force, as illustrated at Figure 1-3. The material beneath the weight is in compression. When an aircraft is standing on the ground the landing gear struts are subjected to compression stress.

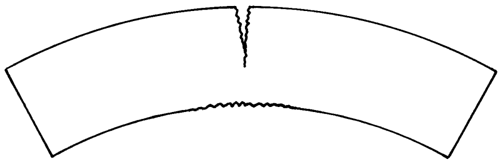
FIGURE 1-3
Compression



Bending

13. Bending involves both tension and compression. When a material is bent it is subjected to both tension and compression stress. This is because one side of the material is being increased in length, or stretched and the other is being shortened, or compressed. This is illustrated at Figure 1-4. When an aircraft wing is bent upwards due to increased loading the upper surface is in compression and the lower surface is in tension.

FIGURE 1-4
Bending



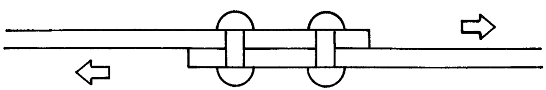
Torsion

14. Torsion is the stress that resists twisting. Thus, the twisting force applied to a wing when the aileron is deflected sets up torsional stress in the wing structure. This twisting force is known as torque.

Shear

15. Shear is the stress that resists a force tending to cause one layer of material to slide over an adjacent layer. Suppose two metal panels were joined by a lap joint with rivets. If the assembly is placed in tension a shearing stress will be set up in the rivets. This is illustrated at Figure 1-5. Were the two components joined by an adhesive bond, the bonding material would be in direct shear.

FIGURE 1-5
Shear



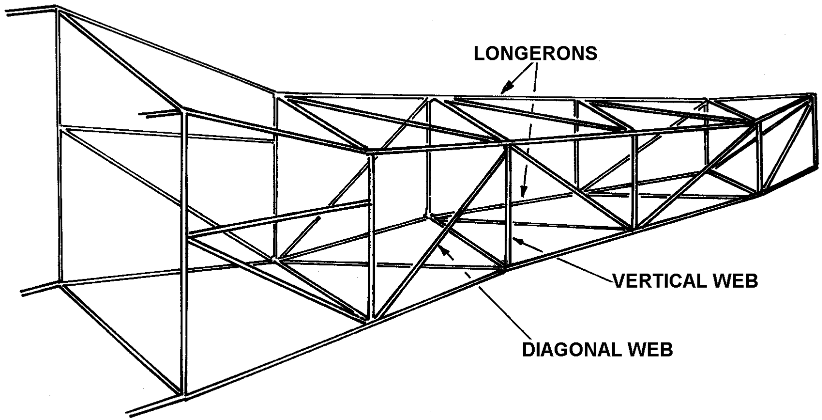
16. The components of the airframe are constructed so as to spread the loads such that stresses are not concentrated at any particular point, which would otherwise be subject to failure. The major flight loads are borne by the aircraft wings and fuselage.

Fuselage Construction

17. Besides providing the accommodation for crew, passengers, freight, systems and equipment, the fuselage must be able to withstand the stresses of flight. These are, typically, the torsion from the empennage (rudder and elevators) and the propeller (in a single-engine aircraft), bending on touchdown and tension and compression transmitted from the wings in flight. There are three common forms of fuselage construction known as steel tube (or truss), monocoque and semi-monocoque.

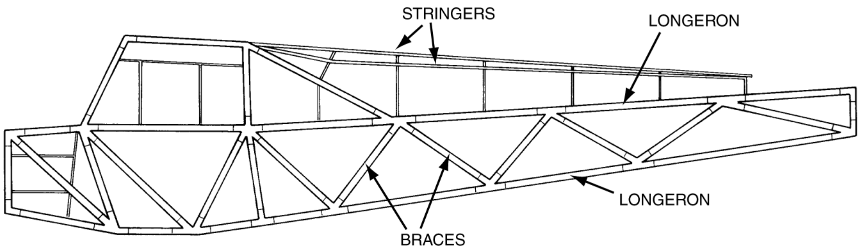
18. The truss type of fuselage comprises a framework made, in modern aircraft, of steel tubes. The principal components are longitudinal tubes called longerons, joined together by lateral braces. The lateral members may be perpendicular to the longerons, with intermediate diagonal braces as shown at Figure 1-6, in which case the construction is known as a Pratt truss.

FIGURE 1-6
Steel Tube or Truss Light Aircraft Construction



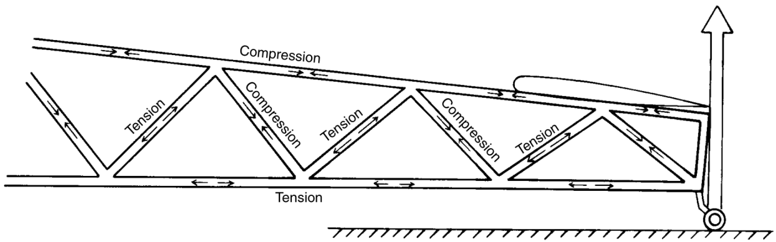
19. In many aircraft an alternative type of truss, known as the Warren truss is used, which employs only diagonal braces between the longerons, as illustrated at Figure 1-7.

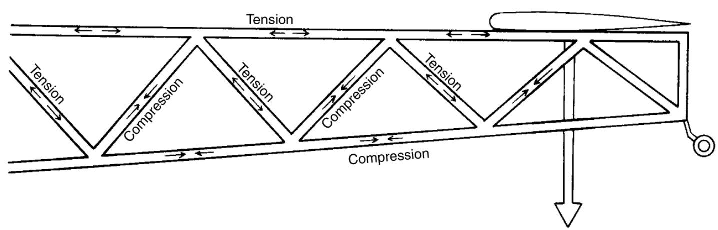
FIGURE 1-7
Warren Truss



20. The basic concept of truss construction is that the compression and tension stresses, due to the bending that a fuselage is primarily subjected to, are alternately carried by the truss components as shown at Figure 1-8. When bending loads are reversed the loading of the truss members is reversed and so stresses are spread evenly over the whole structure, avoiding concentration at any one point.

FIGURE 1-8
Avoiding Stress Concentration



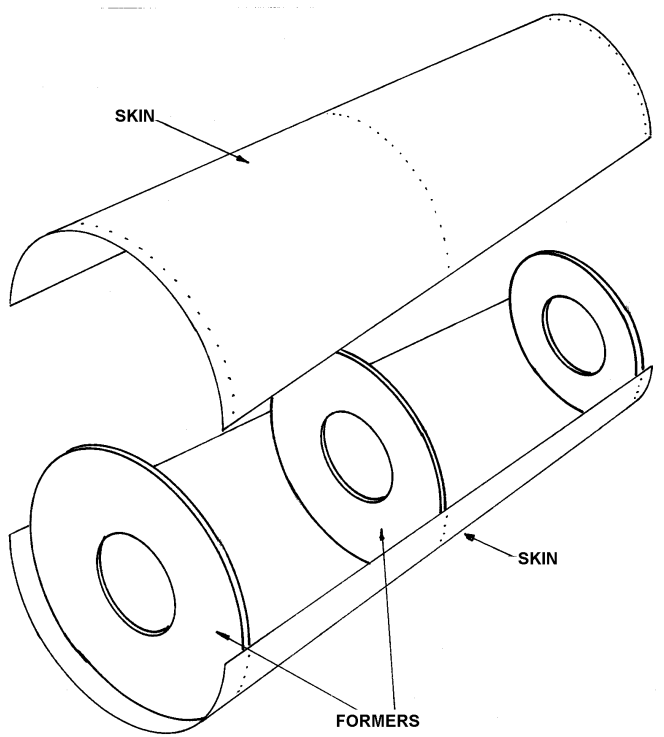


21. Although steel tubing is the material most commonly used in truss construction nowadays, wood and aluminium have both been extensively used in the past, often with steel wire forming some of the bracing members. As a general rule, truss type construction is limited to light aircraft fuselages. The fuselage skin is usually made of thin gauge aluminium, since it carries no load. In earlier aircraft types the skin was often fabric or plywood.

Monocoque Type

22. The name means ‘single shell’ and in this type of construction the strength to maintain fuselage rigidity and withstand stress is all in the fuselage skin. There are no bracing members, only formers to maintain the desired shape of the fuselage. Since the skin must take all the loads this type of construction is unsuited to large diameter fuselages because the skin thickness necessary would incur a high weight penalty. Hence, monocoque construction is limited to small, narrow fuselages. An example is shown at Figure 1-9.

FIGURE 1-9
Monocoque Type
Construction



23. The material most commonly used for monocoque construction is high strength aluminium alloy, 2024 duralumin being a typical example.

Semi-Monocoque

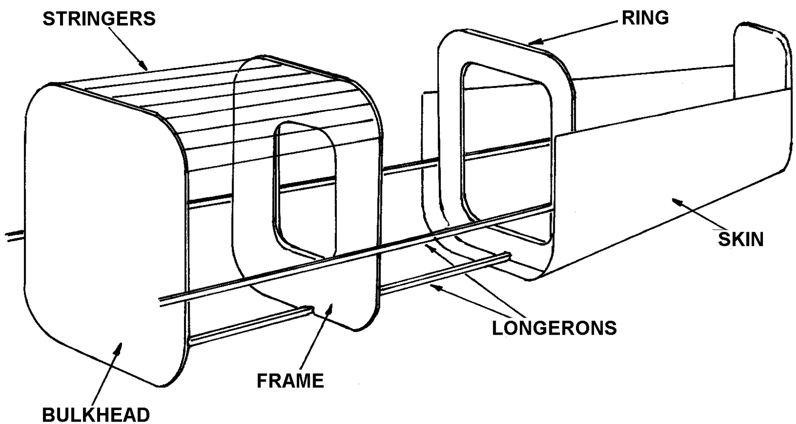
24. Neither truss nor monocoque construction is suitable for most aircraft fuselages, especially where large, pressurised aircraft are concerned. Because of this a form of semi-monocoque construction is used which employs longerons to brace the load-bearing skin material and take some of the loads.

25. Shorter longitudinal members call stringers supplement the longerons. Formers called frames, rings and bulkheads maintain fuselage shape. The main advantage of this form of construction is that it is capable of maintaining its structural integrity even in the event of considerable damage, since loads and stresses are spread over the whole structure rather than being concentrated in the frames or skin. An example of semi-monocoque construction is shown at Figure 1-10.

26. The longerons and stringers absorb the tensile and compression stress due to bending whilst torsional stress is taken up by the skin. The longerons and stringers are also the attachment points for the skin.

27. The materials used in semi-monocoque construction are principally metal, with high strength aluminium alloy being the commonest, especially in smaller aircraft. In larger aircraft steel and titanium alloys are often used for major load-bearing components. Secondary and non load-bearing components are increasingly made from fibreglass, kevlar, graphite-based compounds and composite materials. Cabin floors, for example, are often made from aluminium and fibreglass honeycomb sandwiched between aluminium sheeting.

FIGURE 1-10
Semi Monocoque Construction

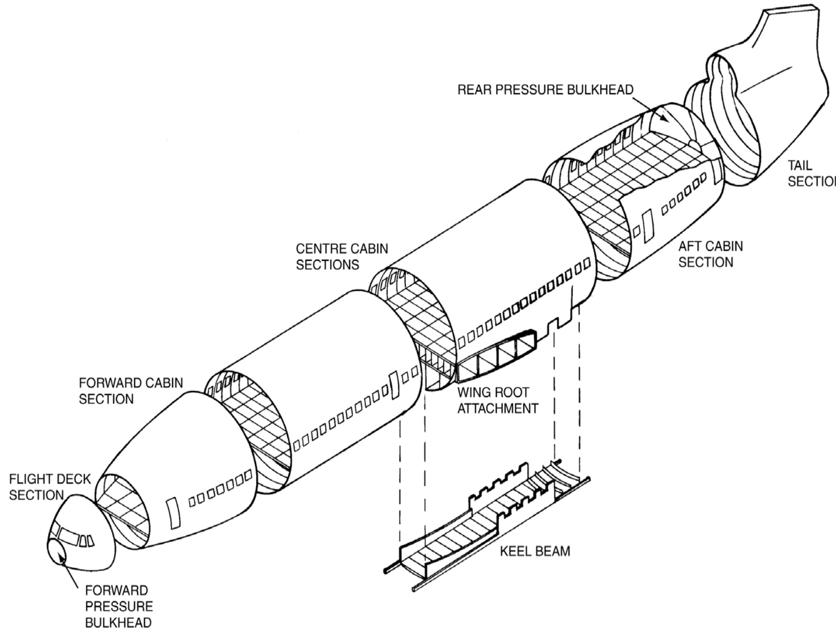


28. In many aircraft fuselages, especially smaller types, a combination of structural methods may be used. Some Cessna designs, for example, use steel truss construction for the forward fuselage and cockpit area and semi-monocoque for the rear fuselage and tail cone.

29. Large transport aircraft fuselages are usually of semi-monocoque construction and formed of a number of sections joined end-to-end. The simplest format comprises a streamlined nose section including the flight deck, a parallel-sided cylindrical cabin section to which the wings are attached and a tapered tail section carrying the empennage.

30. Strong circular frames are spaced at regular intervals along the length of the fuselage, some of which are reinforced to form bulkheads. The frames are joined together by many longitudinal stringers, to which the load-bearing outer skin is attached. In most cases longitudinal strength and rigidity is supplemented by a stout beam extending along the inside base of the structure, known as a keel beam. The keel beam runs along the fuselage centreline through the wing centre section area. The general concept is illustrated at Figure 1-11, showing the fuselage construction principle of the Lockheed L-1011.

FIGURE 1-11
Keel Beam



31. The major stresses to which the aircraft fuselage is subject are bending stress, since all the weight is borne at the wing centre section and, in pressurised aircraft, hoop stress. Hoop stress is due to the tendency of the fuselage to expand because of the internal pressure and places the frames and skin in tension.

Flight Deck and Cabin Windows

32. The cockpit windows must be strong enough to withstand impact damage, such as bird-strike, and must remain clear to afford the pilots uninterrupted forward vision. To achieve these requirements they are usually of laminated construction, especially in larger aircraft. By assembling the pre-stressed laminations so that the directions of principal strength lie perpendicular to each other much greater strength is achieved than with a single transparency of similar thickness. Thin, transparent electrical heating mats are layered between the laminations to maintain the windshields free of frost or condensation.

33. As with all other windows in pressurised aircraft, the cockpit windows must be strong enough to withstand the force due to differential pressure at altitude and must be recessed into a strong framework to prevent them being blown outwards by the internal pressure.

34. Cabin window openings are centred between the fuselage frames and are strengthened by aluminium doublers, or reinforcing plates, around the strong aluminium alloy frames. The latter are recessed so that the window panel is fitted from inside the fuselage to withstand pressurisation forces. The openings have well-rounded corners to avoid stress concentration, which could lead to stress cracking and fatigue failure under the repetitive expansion and contraction of successive pressurisation cycles. For the same reason the size of cabin window openings is limited in pressurised hulls.

35. Cabin windows are usually made of strong plastic material, such as Perspex, and this is also used for the cockpit windows of un-pressurised aircraft. The cockpit windows of pressurised aircraft are usually made from strengthened glass.

Wing Construction

36. The wings generate virtually all of the lift that keeps the aircraft airborne. The wings, therefore, support the remainder of the aircraft. Thus, in flight, there is considerable upward bending force acting upon the wings and this is largely concentrated at the point of attachment to the fuselage. In addition the ailerons, when deflected, apply a twisting force about the lateral centreline of the wings. Consequently, the wing structure must be strong enough to withstand the bending and torsional stresses, which are trying to deform the wing. The fuselage attachment points must be able to withstand the stresses imposed by the upward bending forces acting on the wings and by the twisting forces applied by the ailerons, both of which are trying to separate the wings from the fuselage.

37. In some aircraft, where the wings are necessarily of light construction, the loads are in part taken by bracing struts and wires. In most cases, however, the wings are designed on what is known as the cantilever principle, where structural rigidity is provided entirely by the wing structural members.

38. The bending stresses to which the wing is subjected may be carried by one or more transverse beams, known as spars, or by building the wing as a box structure in which almost all the stresses are carried by the external skin. The latter is known as stressed-skin construction.

39. Torsional stress, due largely to the effects of movement of the centre of pressure, is taken up by chordwise ribs that give greater rigidity. The ribs also provide the aerofoil shape. Stringers run spanwise, between the spars, to provide attachment points for the skin and to provide additional span-wise rigidity.

40. Wings of spar construction are either monospar, having a single spar as the name suggests, two-spar or multi-spar. A monospar wing is illustrated at Figure 1-12 and a two-spar wing at Figure 1-13. Multi-spar wings, having more than two span-wise spars are uncommon.

FIGURE 1-12
Mono Spar

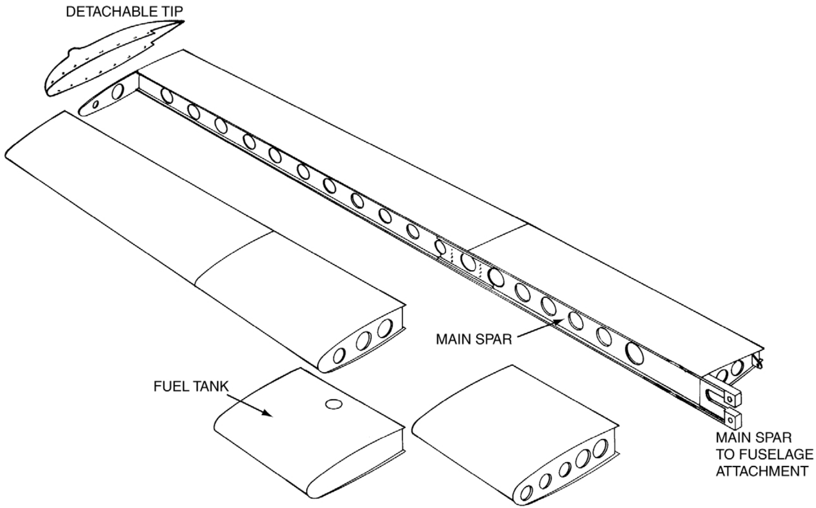
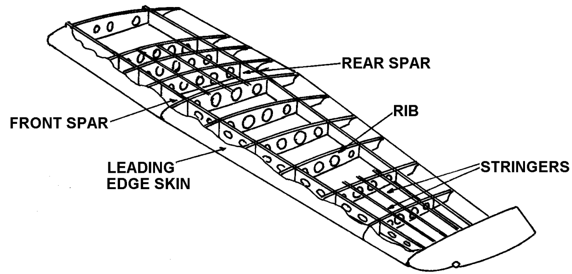
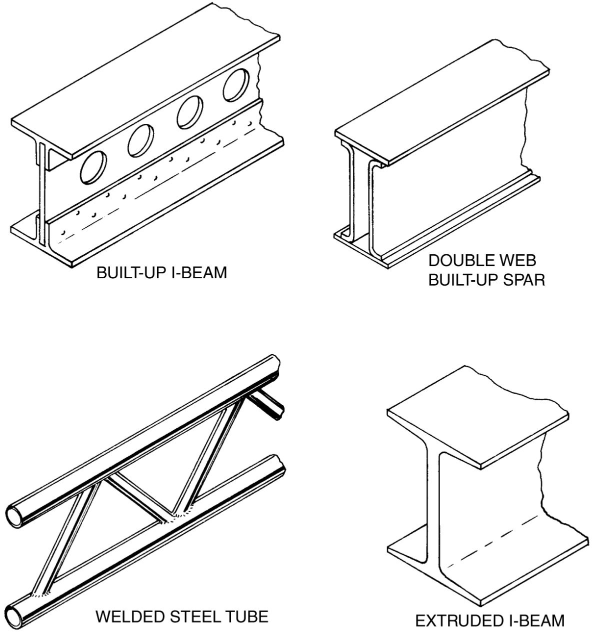


FIGURE 1-13
Two Spar



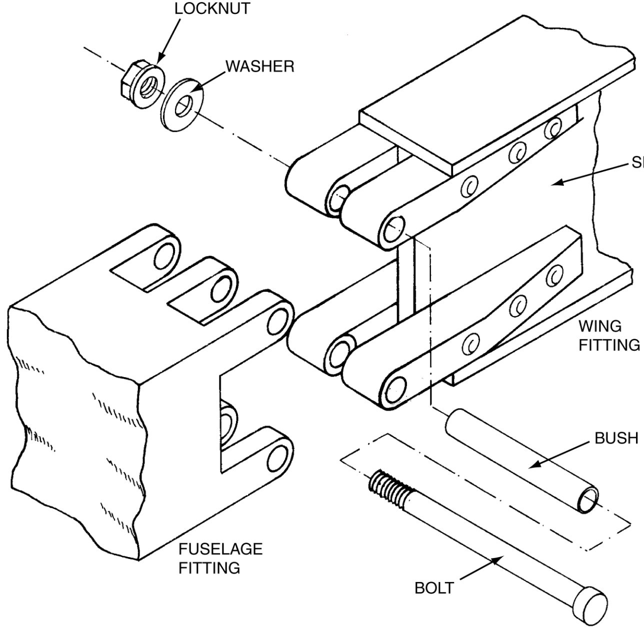
41. The wing spars of modern aircraft are made of metal, formed into a beam either by extrusion or by built-up construction. Some examples of spar beam construction are shown at Figure 1-14.

FIGURE 1-14
Types of Metal
Spar Construction



42. The material most commonly used is high strength aluminium alloy, although a few highly stressed aircraft, particularly military types, use titanium spar webs. The attachment points between wing spar and fuselage centre section are often of titanium or steel alloy in large aircraft. Typical attachment point design is illustrated at Figure 1-15.

FIGURE 1-15
Typical Attachment Point Design

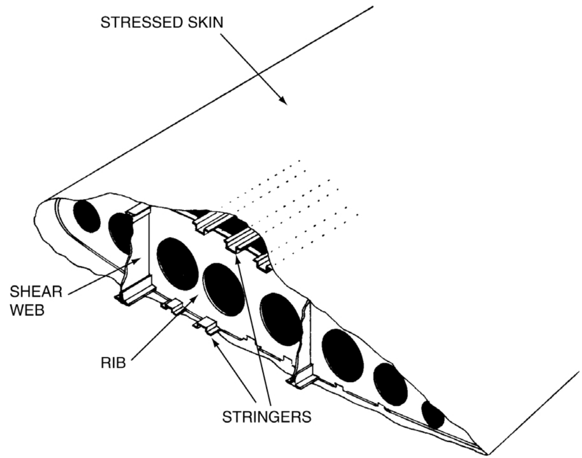


43. The wing ribs are the formers that maintain the aerofoil section of the wing and have to be strong enough to resist the torsional stress tending to twist the wing. These forces are much less than the bending loads carried by the spars and the ribs are consequently of relatively light construction. They are typically either pressed as one piece, or built up from light aluminium alloy sheet.

44. Spanwise stiffness is complemented by the use of stringers running parallel to the spars, to which the wing skin is attached. In spar constructed wings this skin is usually of light aluminium alloy sheet.

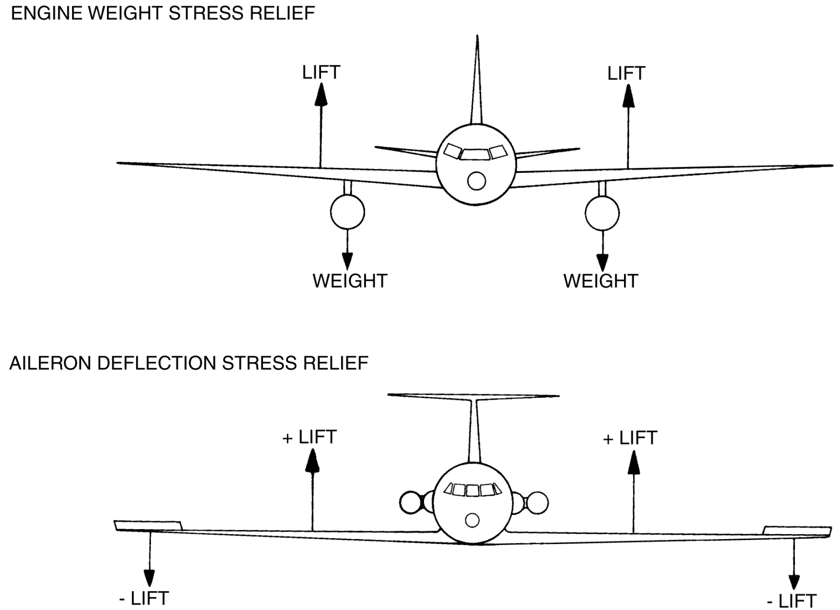
45. Stressed-skin wings have no spars as such, but shear webs running span-wise to which the ribs are attached. In turn, stringers are attached to the ribs and the stressed skin metal is riveted to the stringers to form a load bearing 'box' as illustrated at Figure 1-16. The stressed skin material is usually high tensile aluminium, the thickness ranging from about half a millimetre in small aircraft to as much as 16 millimetres in a large transport aircraft.

FIGURE 1-16
Stressed-Skin Wing Construction



46. The bending stress acting upon a wing in flight can be alleviated to some extent by applying downward forces to oppose the upward force of lift. This can be achieved by wing-mounted engines and by the weight of fuel in outboard fuel tanks. In aircraft with fuselage-mounted engines, or to compensate when fuel in the outboard tanks has been used, some aircraft are fitted with ailerons biased toward the up position to provide a stress-relieving downward force at the outer wings. These concepts are illustrated at Figure 1-17.

FIGURE 1-17
Wing-Bending Stress Relief



Maximum Zero-Fuel Weight

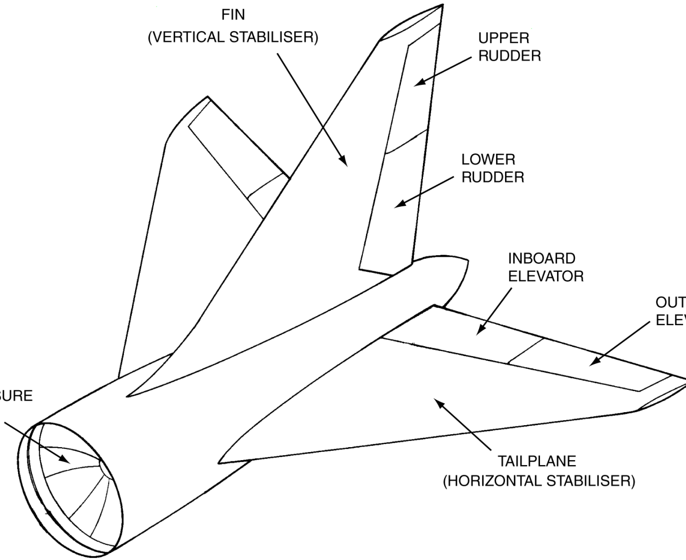
47. The maximum zero-fuel weight (MZFW) is defined as the maximum permissible weight of an aeroplane with no usuable fuel. If an aeroplane were to be airborne with no fuel in the wings, the upward bending stress on the wing structure would exceed its design limiations. Since there will always be fuel in the wings, then these limiations will not be exceeded.

48. In a heavy landing for example, not only is the landing gear likely to sustain damage but the wing spar attachment points are likely to sustain damage due to the large forces as the wings move rapidly downwards. In addition, the fuselage might be subjected to excessive loading stress.

Stabilisers

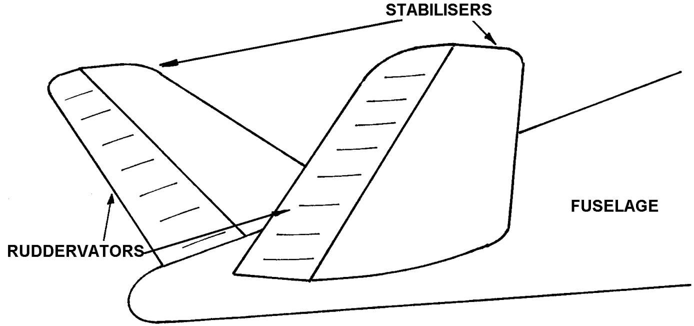
49. The stabilising surfaces of most aircraft are located at the tail and are known as the empennage. Typically they comprise the horizontal surfaces of stabiliser (tailplane) and elevator, for longitudinal stability and control, and the vertical surfaces of fin and rudder for directional stability and control. These are illustrated at Figure 1-18.

FIGURE 1-18
Stabilising Surfaces



50. In a few light aircraft the horizontal and vertical stabilisers and associated control surfaces are combined in a Vee tail configuration. Instead of the conventional fin and stabiliser layout, with directional control provided by a rudder and longitudinal control by the elevators, the fin and rudder are omitted and the stabilisers are set at an angle as shown at Figure 1-19.

FIGURE 1-19
Vee Tail



51. The control surfaces attached to the angled stabilisers are known as ruddervators, since they perform the functions of both rudder and elevators.

52. When the pilot's control column is moved forward, both ruddervators move downward in unison and when the control column is moved rearward both ruddervators move upward in unison, thus providing longitudinal (pitch) control.

53. When the rudder pedals are moved, the ruddervators move in opposite directions to give directional (yaw) control. For example, pushing on the right rudder pedal, for right yaw, causes the right ruddervator to move down and the left ruddervator to move up. This will apply a force to the left on the tail assembly, causing the aircraft to yaw to the right.

54. When the left rudder pedal is pushed, the left ruddervator will move down and the right ruddervator will move up.

55. When both controls (control column and rudder pedals) are operated together the control surfaces movement is differentially biased to achieve the desired effect. To initiate a pitch up and yaw to the right, for example, (control column back, right rudder pedal forward) the left ruddervator will move up (yaw right, pitch up) whilst the movement of the right surface will depend upon the bias of the yaw/pitch requirement. If more pitch than yaw is required the right surface will move up also, but less than the left. If more yaw than pitch is required the right surface will remain stationary or only move up a very small amount.

56. The construction of the stabilising surfaces is basically the same as that for the aircraft wings, but on a considerably smaller scale, since far lower lifting forces and their associated twisting and bending stresses are involved. Typically the horizontal and vertical stabilisers (fin and tailplane) are of twin-spar construction, with the associated flying controls (elevator and rudder) hinged to the rear spar. Because of this the rear spar is usually much stronger than the front spar, which is the opposite of the usual wing construction. The material used in stabiliser structures is primarily aluminium alloy.

57. The horizontal stabilising surface (the tailplane) is required to exert little effort in steady level flight, usually developing a relatively small negative lift to balance, or stabilise, the lift/weight couple. The effort exerted by the tailplane only changes significantly when the elevators are deflected. The vertical stabiliser (the fin) exerts no effort until the rudder is deflected.

58. Horizontal stabilisers are susceptible to aerodynamic flutter, a form of structural vibration in which the surface twists under load. The twisting of the aerofoil leads to bending due to aerodynamic force, which leads to twist in the opposite direction, a reversal of aerodynamic bending and so on. The cycle of twisting and bending usually increases with time, which ultimately results in structural failure. The solution is to increase the torsional stiffness of the structure, to prevent the initial twisting. A full explanation of the various flutter modes can be found in the Principles of Flight book.

59. During flight at high Mach numbers the centre of pressure rapidly moves longitudinally, resulting in large changes in the aerodynamic force generated by the horizontal stabiliser. In large aircraft this could cause rapid changes in the stabilising effort and therefore stress on the stabiliser structure. To compensate for the centre of pressure movement, aircraft required to operate at high Mach number are equipped with a Mach trimming device. This automatically adjusts horizontal stabiliser effort, by adjusting either the elevators or the tailplane incidence, to maintain longitudinal trim and prevent rapid reversals of loading on the stabiliser structure.

Design Philosophies

60. Components of an aircraft structure are known as structural or non-structural. Structural components transfer loads and forces from one location to another, or absorb forces. Such components are those already described such as wing spas, ribs, fuselage, bulkheads etc. Non-structural components do not transfer loads or absorb forces, and include such items as landing gear doors, cabin doors and nose radar radomes.

61. The life of an airframe is limited by fatigue, caused by the load cycles imposed during take-of, landing and pressurisation. This lift has been calculated over the years by using different design philosophies, these being safe-life, fail-safe, and damage-tolerant.

Safe-Life

62. This original design philosophy involved testing to failure and to allocate the structure, a safe life of 25% of the average life when tested to destructive failure.

Fail-Safe

63. In this construction, components were designed to accept the loads of adjacent components should one of the latter fail. The philosophy anticipated failure. However, precise inspection techniques were not specified, and there were, for example, no studies of crack growth. In addition, testing was done with new components under laboratory conditions (as was fail-safe) and did not take into consideration the deterioration caused by such things as climate conditions and aircraft utilisation.

Damage Tolerant

64. This design philosophy accepts that production components will have minor flaws and anticipates their growth. Precise inspection procedures are then laid down in order that these flaws may be identified before they become critical. It is damage-tolerant design which is now used for large transport aeroplanes.

Station Numbers

65. During operational service, an aeroplane will require servicing to various levels for general maintenance, and will also require the replacement of parts. It will thus be necessary to establish a method of locating various components by establishing reference lines and station numbers for various parts of the aeroplane structure. For example, points along the fuselage are referenced to a zero datum which is usually at the front of, or close to, the fuselage. The station numbers are given as distances forward (or aft) of the datum line. Wing stations are measured from the fuselage centreline.

Primary Controls

66. The function of the primary flying control surfaces is to provide control about one of the three primary axes of roll, pitch and yaw. These are usually the ailerons for control in roll, the elevators for control in pitch and the rudder for control in yaw.

67. The ailerons, for roll control, are mounted on the outboard trailing edge of the wings and move differentially when deflected, one hinging upward to decrease lift at that point and the other hinging downward to increase lift. The resulting differential aerodynamic forces produce a rolling moment about the longitudinal axis of the aircraft. They are operated in some cases by sideways movement of the pilot's control column (or side stick), but more often by rotation of a control wheel attached to the top of the control column. Rotating the control wheel to the right deflects the right aileron up and the left aileron down, for roll to the right. Rotating the control wheel to the left has the opposite effect.

68. The elevators, for pitch control, are hinged to the trailing edge of the horizontal tail surfaces. When deflected downward the positive camber of the surface is increased, creating positive lift aft of the aircraft CG to produce a nose-down pitching moment about the lateral axis of the aircraft. Upward deflection of the elevators produces the opposite effect. They are operated by fore and aft movement of the pilot's control column. Forward movement deflects the elevators down for pitch down; aft movement deflects the elevators up for pitch up.

69. The rudder, for directional control, is hinged to the trailing edge of the vertical stabiliser, or fin. When deflected it produces camber on an otherwise symmetrical aerofoil, creating lift in a sideways direction. Left deflection of the rudder produces sideways lift to the right, aft of the aircraft CG, creating a yawing moment about the aircraft normal axis, which yaws the nose to the left. Right deflection of the rudder has the opposite effect. The rudder is operated by the pilot's rudder foot pedals, attached to the rudder bar. Pushing the left pedal forward deflects the rudder to the left and vice versa.

70. In all cases the degree of movement of the control surfaces is directly proportional to the degree of movement of the pilot's controls.

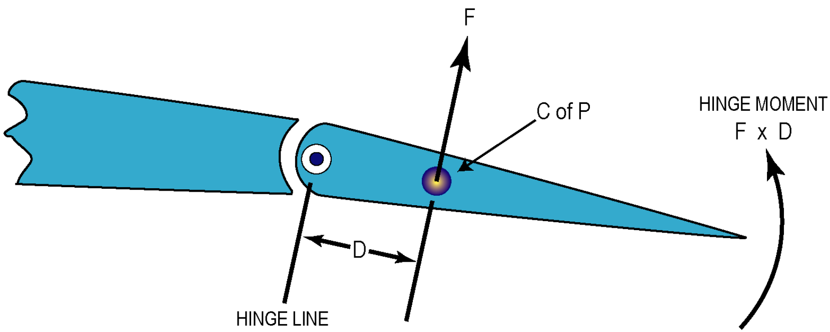
71. On some aircraft the effect of two of the above is combined in a single set of control surfaces. Examples are elevons, which combine the effects of elevator and ailerons and which are mounted at the wing tips of some swept-wing aircraft. They move differentially for roll control and up or down together for pitch control. Tailerons are essentially the same as elevons, except that they are tail-mounted rather than wing-mounted. Ruddervators are used on Vee, or butterfly tail aircraft and combine the effects of rudder and elevator, as explained earlier under stabilisers.

72. The method of actuation between the pilot's controls and the primary control surfaces is, in many aircraft, direct mechanical. In this type of system, movement of the pilot's controls is directly transmitted to the appropriate control surfaces by means of rods or wires passing through guiding pulley wheels. The force exerted by the pilot in moving the controls depends entirely upon the aerodynamic forces acting upon the control surfaces, which increase as the square of the true airspeed. Individual pilot strengths will clearly vary and consequently airworthiness requirements stipulate specific load limits for conventional wheel type controls which should not have to be exceeded. To assist the pilot a variety of control balancing devices may be used, which reduce the aerodynamic force to be overcome.

Aerodynamic Balance

73. When a control surface is deflected the airflow acting over it will try to return the control to the neutral position. The total force trying to return the control surface to the neutral position is the product of the lift force on the control surface and the perpendicular distance of the centre of pressure of the control surface to the hinge line. This is called the hinge moment and is illustrated at Figure 1-20.

FIGURE 1-20
Hinge Moment



74. The magnitude of the lift force generated by any control surface will vary directly as the square of the EAS. The pilot is required to provide the force to overcome the hinge moment and deflect the control surface (in a manual system). At all but the lowest airspeeds he or she could do with some form of assistance. This assistance is supplied in the form of control balance devices.

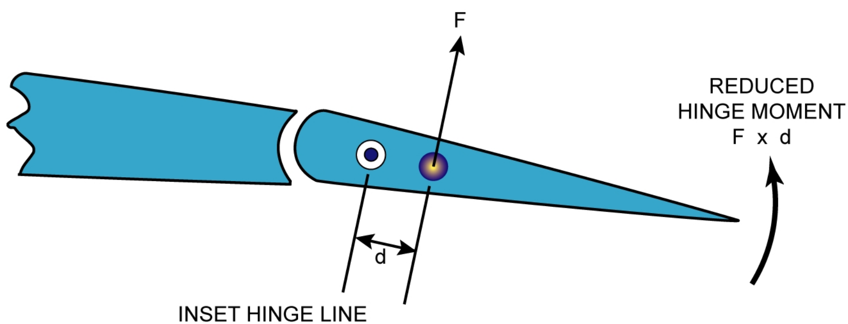
75. Control balancing is achieved either by reducing the hinge moment, or by setting up a force that acts against the hinge moment.

Nose Balance or Inset Hinge

76. This is perhaps the simplest form of aerodynamic balance. The hinge is set back towards the CP of the control surface so that, when it is deflected, air strikes the surface forward of the hinge and reduces the force needed to move the control by partially balancing the aerodynamic force aft of the hinge.

77. With this type of balance, care must be taken in the design to ensure that the centre of pressure is not too near the hinge line. When a control surface is operated its centre of pressure moves forward. If the margin between the centre of pressure and the inset hinge is too small there is a possibility that the CP will move forward of the inset hinge, reversing the direction of the hinge moment and is known as overbalance. An inset hinge is illustrated at Figure 1-21.

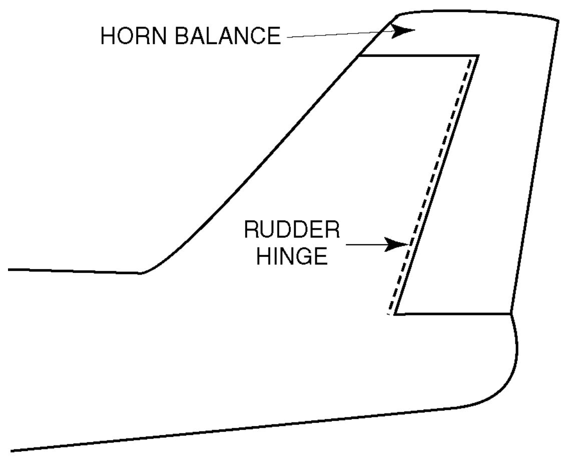
FIGURE 1-21
Inset Hinge



Horn Balance

78. An example of a horn balance is shown at Figure 1-22. Although it is shown here on a rudder, horn balances can equally well be used on ailerons or elevators. In this system a portion of the control surface acts ahead of the hinge line, and therefore produces a moment in opposition to the hinge moment.

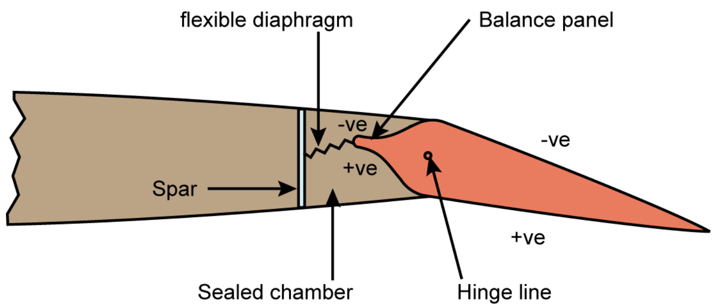
FIGURE 1-22
Horn Balance



Internal Balance

79. A projection of the control surface in the form of a balance panel, often referred to as a 'beak', is connected by a flexible diaphragm within a sealed chamber to a fixed structure (eg. spar). See Figure 1-23. Control surface movement produces a pressure differential between upper and lower surfaces and these upper and lower surface pressure changes are fed to the chamber to provide a partial balancing moment. Internal balance is therefore achieved with no increase in exterior drag.

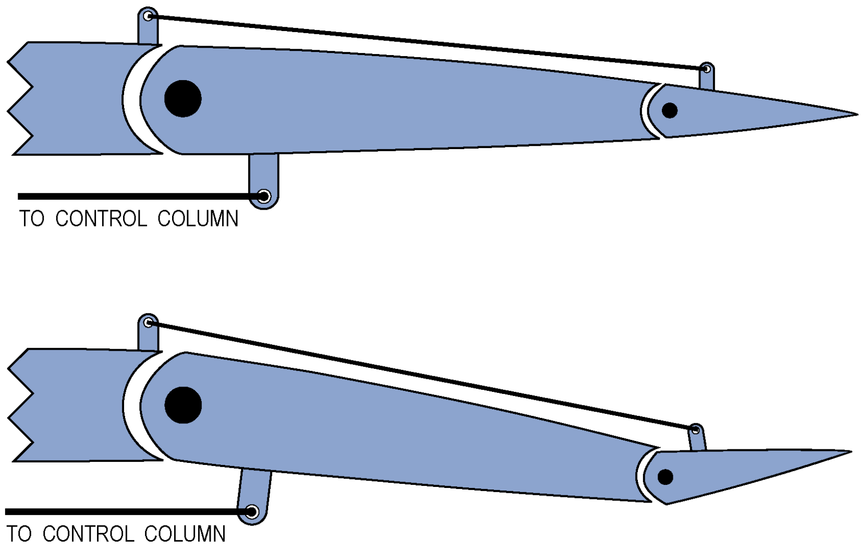
FIGURE 1-23
Internal Balance



Balance Tab

80. Like the inset hinge and the horn balance the balance tab serves the purpose of reducing the stick forces involved in moving a primary control surface, at a given airspeed. A balance tab is shown at Figure 1-24. As the primary control surface moves one way, so the balance tab moves the other. Since the tab is a considerable distance from the hinge line of the primary control, the moment produced by it is large. The balance tab imposes a small penalty in terms of drag, and diminishes slightly the effectiveness of the primary control surface to which it is fitted.

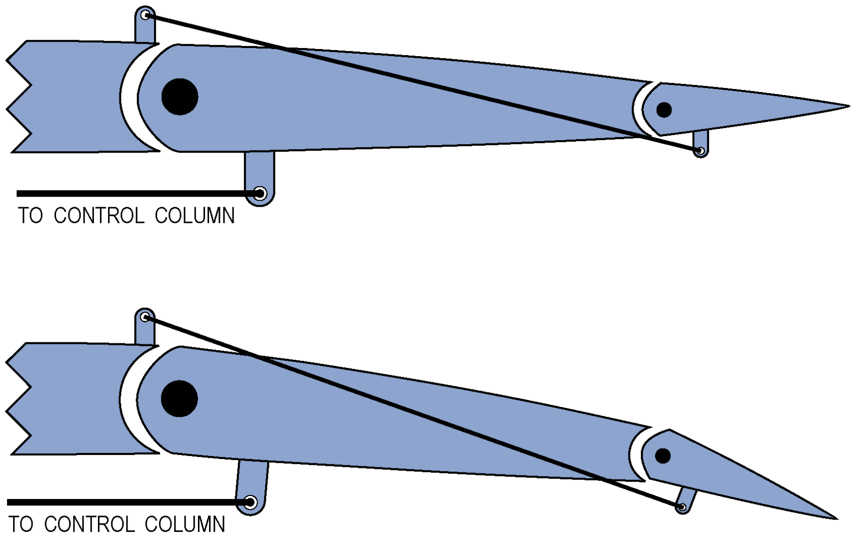
FIGURE 1-24
Balance Tab



Anti-balance Tab

81. In some aircraft, far from requiring assistance in moving a control surface against the aerodynamic loads, the hinge moment is too small. This results in very low control column loads, a lack of feel and the possibility of over-stressing the airframe due to excessive deflection of the control surface. This often occurs because of the hinge being too close to the centre of pressure of the control surface. In order to improve the situation an anti-balance tab is fitted which operates in the same direction as the control surface. An anti-balance tab is illustrated at Figure 1-25.

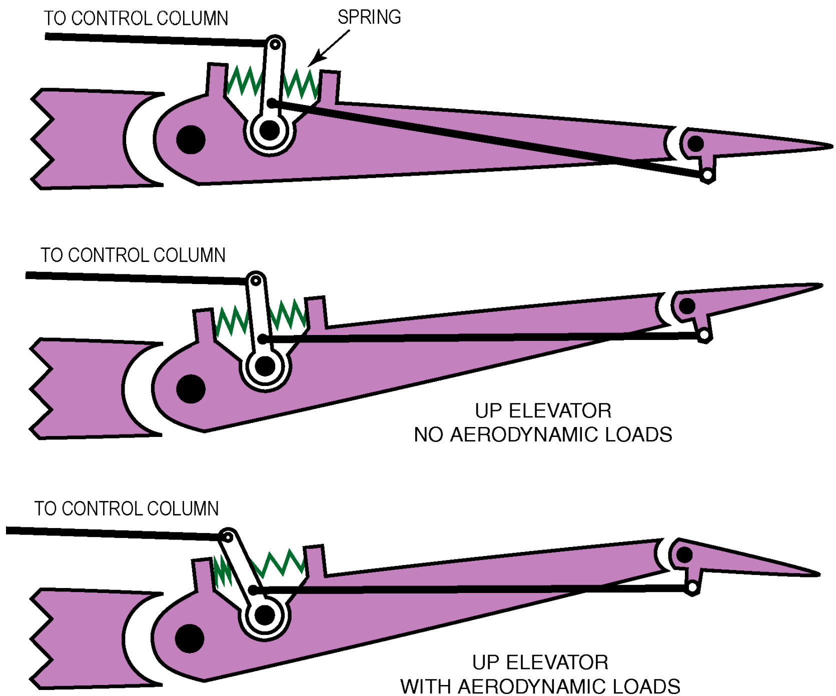
FIGURE 1-25
Anti-Balance Tab



Spring Tab

82. With many aircraft aerodynamic balancing is not considered necessary at low airspeeds, but is progressively required as airspeed increases, and with it the aerodynamic loads. The spring tab system may then be fitted to deal with this situation. A spring balance tab is shown at Figure 1-26.

FIGURE 1-26
Spring Tab

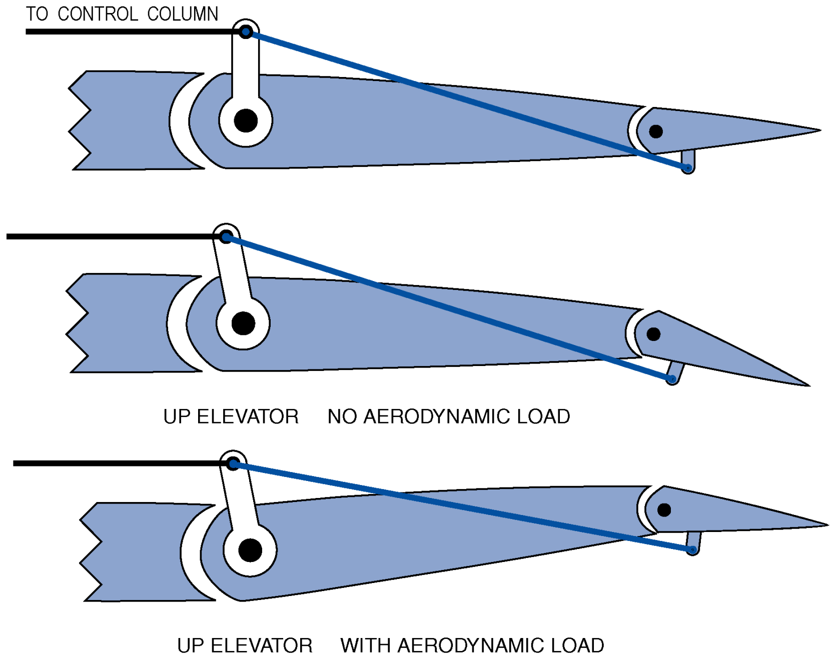


83. The movement of the control column is transmitted to a lever pivoted on the main control surface but not directly operating it. Operation of this surface is through springs, and with low aerodynamic loads the movement of this pivot arm is transmitted to the main control surface through the springs; consequently there is no alteration in the geometry between the primary control surface and the balance tab. When the aerodynamic loads increase at high speed, in order to transmit the control column movement via the pivot arm to the control surface, the spring becomes compressed. This upsets the geometry of the system and brings into operation the balance tab on the trailing edge, which moves in the opposite direction to the primary control surface, thus assisting the pilot by reducing the stick forces involved.

Servo Tab

84. When manual controls are used to operate very large control surfaces the loads involved, even with balance tab assistance, may be unacceptable. Under these circumstances servo tabs are used to operate the control surfaces. A servo tab is a small aerofoil section, once again attached to the trailing edge of the main control surface. The servo tab is operated directly by the control column, with no direct connection between the control column and the main control surfaces. As with the balance tab and the trim tab (described later), the servo tab moves in the opposite direction to the primary control surface. A servo tab is illustrated at Figure 1-27.

FIGURE 1-27
Servo Tab



Powered Controls

85. The force needed to move the control surfaces of a large aircraft flying at high speeds makes it virtually impossible for satisfactory control to be exercised by manual controls. It is therefore necessary that the primary control surfaces be power operated.

86. Powered controls may be divided into two categories, power-assisted controls and fully power operated controls.

Power Assisted Controls

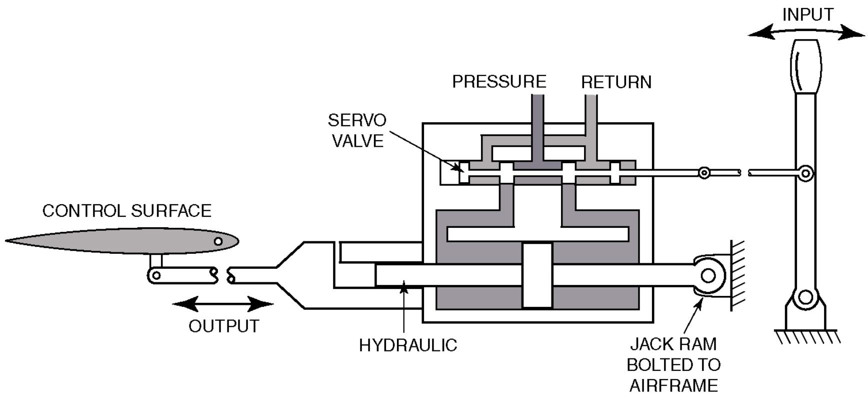
87. With power-assisted controls, the force needed for movement of the control surface is provided partly by the physical force produced by the pilot and partly by the power system. Here then the pilot will have 'feel' which is provided by the control surface loads. Should a fault or power failure occur in the control system, a disconnect system will be available, and control will continue to be maintained by manual means alone, although control loading will be relatively high. Trim control for a power-assisted control is provided in the same way as that for manually operated flying controls.

Fully Powered Controls

88. Where fully power operated controls are installed, power systems are provided which, while independent of each other, operate in parallel and provide all the force necessary for operation of control surfaces. Movement of the pilot's controls is transmitted to actuators, which provide the force necessary to move the control surfaces.

89. Safeguards against faults or power failure may be provided by manual reversion, or by more than one system, each with its individual hydraulic circuit. Thus conventional trailing edge tabs will not be included and trim would be obtained by altering the zero positions of the artificial feel mechanism. Occasionally, where necessary, balance tabs are fitted for the maintenance of servo and hinge loads. An example of a power operated control system is shown at Figure 1-28.

FIGURE 1-28
Powered Flying Control System

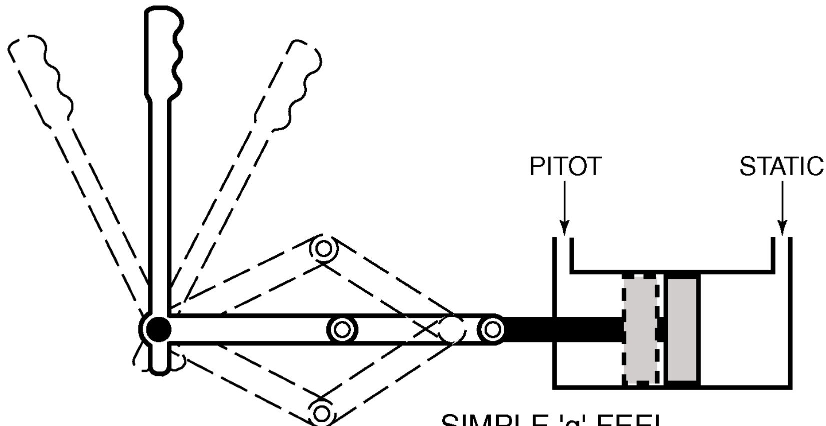


90. Power operated controls are irreversible, which means that there is no feedback of aerodynamic forces from the control surface. Consequently, the pilot has no feel through the controls for the aerodynamic loading on the control surfaces.

91. Feel is provided by artificial methods and, in fact, the artificial feel rarely has any direct relationship to the forces working on the control surface. Feel can be provided by a spring that exerts a constant load for a given control position, so that the more the control is moved the greater the spring force to be overcome. The disadvantage of this system is that the resistance to control movement is the same irrespective of airspeed. A proportionate force at low airspeed would be inadequate at high speed, alternatively a proportionate force at high airspeed would be too great at low speed.

92. A much more satisfactory arrangement is to provide loading which varies in direct relationship to airspeed. This loading is a force proportional to dynamic pressure (q). The arrangement is therefore commonly referred to as q feel. Pitot and static pressure is sensed and applied to the pilot's controls to produce a force resisting movement proportional to airspeed and therefore representative of the force the pilot would feel with manual controls. In its simplest form a q feel system comprises a large piston with pitot pressure applied to one side and static pressure to the other. The overall force acting on the piston is thus due to dynamic pressure. The piston is connected to the pilot's controls as shown at Figure 1-29.

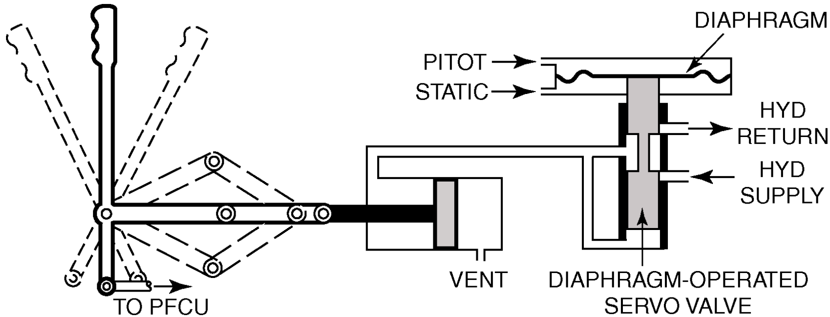
FIGURE 1-29
Simple ‘q’ Feel



SIMPLE 'q' FEEL

93. The bulkiness of the simple ‘q’ feel system can be overcome by the utilisation of a hydraulically enhanced system in which the pitot and static pressures are fed to either side of a diaghragm attached to a hydraulic servo-valve. The servo-valve provides a metered hydraulic pressure, which is an amplified value of dynamic pressure, to a small jack which opposes control column movement. An example of a hydraulic ‘q’ feel system is shown at Figure 1-30.

FIGURE 1-30
Hydraulic 'q' Feel



Fly-By-Wire

94. In many present-generation aircraft signals from the pilot's primary controls are transmitted electrically to actuators which move the control surfaces. This is known as fly-by-wire. The system lends itself to the incorporation of sophisticated electronic processing which alters the response to control inputs by the pilots to avoid stalling, over-rapid or excess control surface movement, or unstable flight regimes. Fly-by-wire not only has the capability to improve aircraft performance, efficiency and safety, it can also incorporate co-ordination of control surface movements too complex for a pilot to achieve unaided.

95. Indication of the aircraft response to the pilot's control inputs is by means of the primary flight instruments; artificial horizon for roll and pitch, turn and slip indicator for yaw. Angle of attack sensors may give aural stall warning, or stick shaking, in the case of excessive pitch up.

96. To prevent control surface damage due to strong gusts of wind when the aircraft is stationary on the ground, control locks are fitted. These can be either external mechanical locking devices to prevent control surface movement, usually fitted to aircraft with manual controls, or internal hydraulic locks incorporated with powered flying controls. Visual warning that such locks have been fitted are provided externally at the control surface and in the cockpit at the operating control. On some aircraft, warning lights indicate when control surface locks are in place.

Secondary Controls

97. The principal function of the secondary control surfaces is not to provide control about one of the primary axes, but to adjust the lift or trim in specific flight circumstances. Examples are flaps, slats, spoilers (except when used for roll control), airbrakes, trimming tail-plane and trim tabs. All of these are described in detail in Principles of Flight.

98. Flaps and slats are attached to the trailing and leading edges of the wings and are extended symmetrically to increase wing lift at low flight speeds, such as approach and take-off. Spoilers are hinged to the wing upper surface and, when extended, destroy some of the lift generated. They are used during and after touch down to prevent 'floating' and extension of the landing roll and can be selected manually or automatically on landing after touchdown and wheel spin-up, provided the lever is in the armed position, or for a rejected take-off, when reverse thrust is selected above a particular speed (typically 60kt).

99. Airbrakes may be extended symmetrically from the wing upper or lower surfaces (or both) or from the fuselage to create drag and slow the aircraft in flight, during descent for example. A trimming tailplane is a horizontal stabiliser with variable incidence, adjusted to trim the aircraft longitudinally so that aircraft attitude can be maintained with minimal stick force. Trim tabs are attached to the primary control surfaces and adjusted to elminate the stick force needed to hold a given control position.

100. The mode of actuation of the secondary control surfaces may be direct mechanical, hydraulic or electric powered, or fly-by-wire, depending upon the aircraft size and operating speed. The pilot's controls usually take the form of levers for flap, slat, airbrake and spoiler operation and trimming wheels or electrical trim switch on the control wheel for trimming surfaces.

101. The position of flaps, slats, airbrakes and spoilers is indicated by the operating control position and, on larger aircraft, by position gauges and/or electronic indicating systems.

102. It is essential that wing-mounted lift augmentation devices (flaps and slats) should always be extended symmetrically, since asymmetric deployment would create a strong rolling moment difficult or impossible to counteract with the primary controls. Interconnecting actuators are designed to prevent asymmetric deployment and many aircraft are equipped with warning systems to alert the pilot to this potential hazard. Asymmetric deployment of wing-mounted airbrakes is similarly undesirable and warning devices are often incorporated with these controls.

In addition, a flap load limiter system protects the trailing edge flaps from excessive air loads by automatically retracting flaps from the fully extended landing flap position whenever the airspeed exceeds a predetermined speed. When airspeed is reduced, the flaps automatically return to the fully extended position.

Landing Gear Systems

Landing Gear Systems

1. The landing gear of a fixed wing aircraft supports the weight of the aircraft while it is on the ground and is made up of the main and auxiliary landing gear. The main gear provides the principal support. The auxiliary gear is in the form of a nose or tail wheel installation, almost invariably the former, especially in large aircraft.

2. The advantages of tricycle landing gear during ground operation are:

(i) No risk of the aircraft 'nosing over' during hard braking.

(ii) Better visibility for the pilot during taxiing.

(iii) Less likelihood of ground looping during take-off or landing in cross-wind conditions.

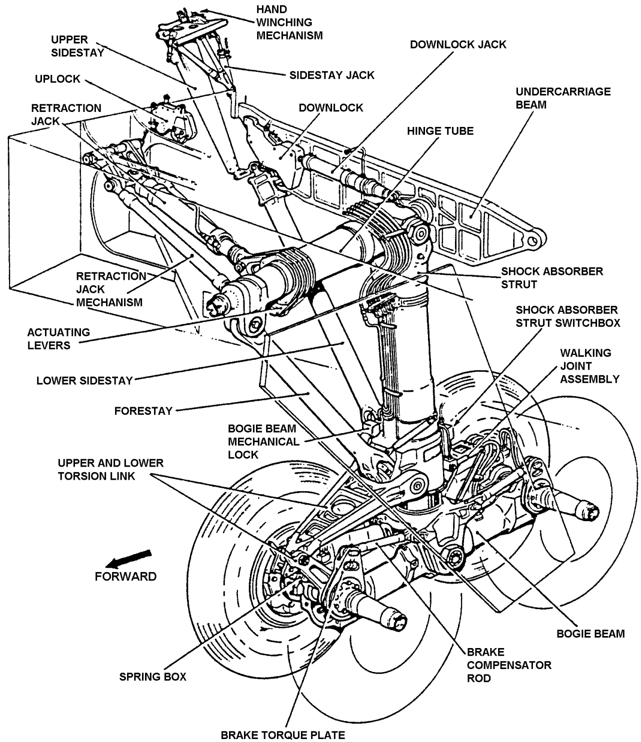
3. The landing gear contains shock absorbers to withstand the landing forces and the effects of taxiing over uneven surfaces, and brakes to stop and assist in controlling the aircraft on the ground. Most modern aircraft with tricycle (nose-wheel) layouts are equipped with nose-wheel steering for ground manoeuvring, some very large aircraft also have main wheel steering (known as body steering). In aircraft designed for flight at high speeds/altitudes the landing gear is retracted into the fuselage or wings in flight, to reduce profile drag. In low performance aircraft this is less important and the landing gear is often fixed in the extended configuration.

Main Components of Landing Gear

4. Figure 2-1 shows typical main landing gear components in a large aircraft.

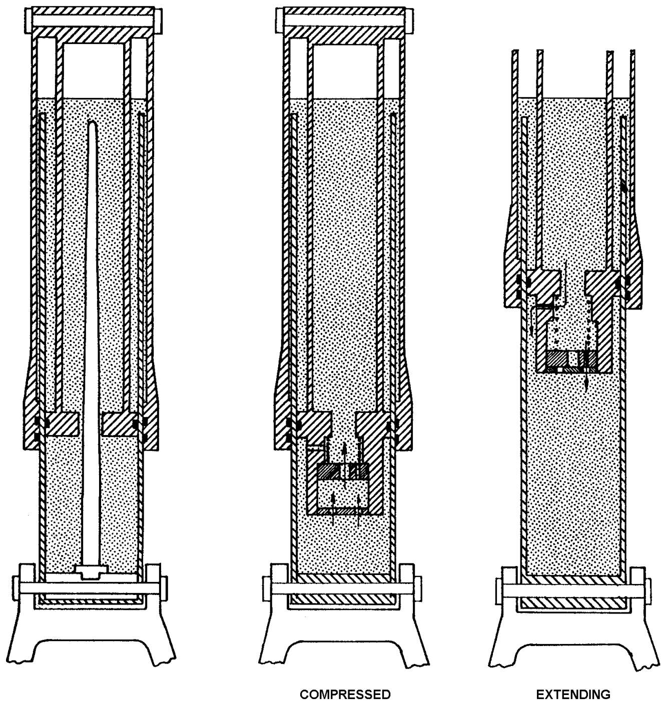
FIGURE 2-1

Main Landing Gear Components



Oleo-Pneumatic Shock Struts The landing gear shock struts are designed to absorb the shock loads of landing and taxiing over uneven ground, preventing them from being transmitted to the airframe. The upper (outer) cylinder of the strut is attached to the airframe and contains a lower (inner) cylinder which is free to slide up or down (and rotate in the case of the nose strut) within the outer cylinder. The cylinders are partially filled with oil and pressurised with compressed air or nitrogen. This compressed gas absorbs the shocks of normal taxiing and balances the weight of the aircraft when it is stationary on the ground, so that the inner cylinder takes up an approximate mid-stroke position. Under the increased shock of landing the inner cylinder moves up, shortening the strut length. To prevent excessive upward movement, transference of oil from lower to upper cylinder is progressively restricted by either a metering pin or a snubber valve as shown at Figure 2-2. As the volume of the gas space in the upper cylinder decreases with the upward movement, the gas pressure increases to balance the upward force. To prevent aircraft bounce on landing, the shock absorber damping on rebound is greater than the damping on compression.

FIGURE 2-2
Landing Gear
Shock Struts



METERING VALVE TYPE

SNUBBER VALVE TYPE

5. It is important that the struts are inflated to the correct gas pressure. Too high a pressure and the shock absorption is reduced, too low a pressure and the strut extension will be inadequate, leading to 'bottoming' and complete loss of shock absorption under shock loading. Checking the strut gas pressures is a job for an engineer, however a check of the amount of the inner cylinder which is visible (the amount of extension) is a good indication to the pilot on a walkaround inspection that the gas pressure is approximately right. Tables or graphs may be available which enable the pilot to determine the appropriate extension for a given weight and loading configuration.

Landing Gear Bracing

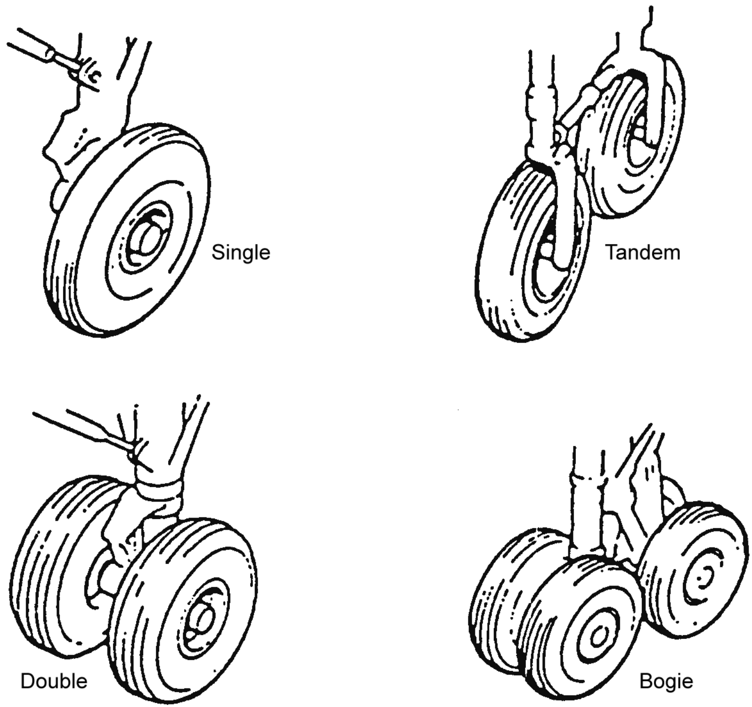
6. During take-off, landing and taxiing, the landing gear is subject to considerable side-for/aft loads. To prevent damage, breaking or possible collapsing of the leg, additional support is provided by fitting side load struts/stays and drag struts to the gear assembly. See Figure 2-1.

7. Torsion links maintain the wheel alignment with the longitudinal axis of the aircraft. They join the inner and outer cylinders but allow the shock strut to compress for damping.

Main Landing Gear Wheel Configuration

8. Heavy aircraft need to spread the weight of the aircraft over a wide area to achieve an acceptable pavement loading. The four basic configurations of landing gear wheel arrangement - single, double, tandem and bogie are illustrated at Figure 2-3.

FIGURE 2-3
Main Landing Gear Wheel Configuration



Landing Gear Locks

9. Landing gear up locks and down locks are provided. The locks are engaged by spring force and broken during retraction/extension by hydraulic pressure.

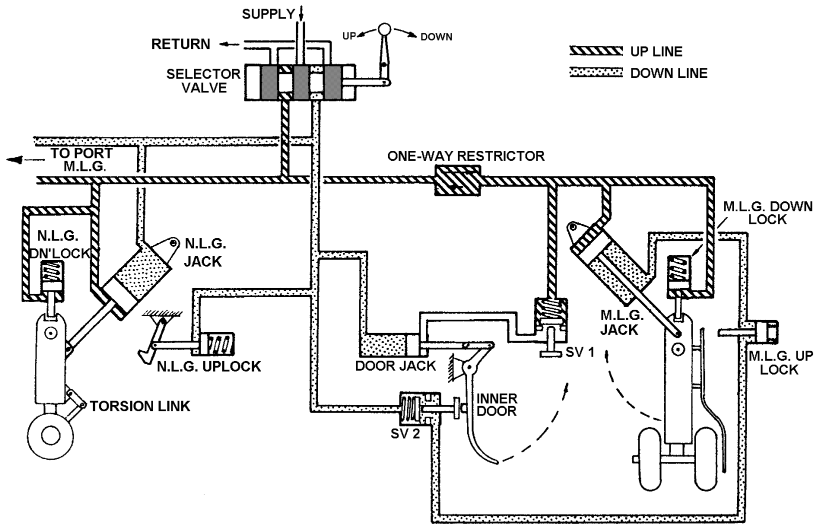
Landing Gear Doors

10. Landing gear doors are used to enclose the retracted landing gear to reduce drag in flight. The doors may be mechanically operated by gear movement or hydraulically operated and sequenced with the landing gear.

Extension and Retraction

11. Raising and lowering the retractable landing gear of an aircraft is, almost invariably, achieved by hydraulic systems. Since the landing gear is essential to safe landing of the aircraft, it is of vital importance that there should be an alternate means of extending the gear in the event of hydraulic system failure. In many aircraft an emergency pneumatic system can be selected to actuate the landing gear. An alternative method of lowering the landing gear in an emergency is a gravity extension or free-fall system. In this type of system, hydraulic pressure to the retraction system is shut off, the raise and lower lines are open to return and the gear and door up-locks are mechanically released. The gear extends under the influence of gravity and the down locks are engaged by spring action. Figure 2-4 shows a landing gear system and a description of its operation follows.

FIGURE 2-4
Landing Gear System



12. A selector valve directs hydraulic system pressure to the appropriate side of the operating jacks (up or down), at the same time connecting the other side of the jacks to the reservoir return line. Figure 2-4 shows the gear extended, let us first follow the sequence of operations when UP is selected.

Up selection. The selector valve spool moves to the right, directing hydraulic pressure to the up lines, releasing the main landing gear (MLG) and nose landing gear (NLG) down locks. Pressure is also applied to the MLG and NLG operating jacks to retract the gears. As the NLG reaches the fully-retracted position the spring-loaded NLG up lock engages with a spigot on the torsion links. As the MLG reaches the fully-retracted position the spring-loaded MLG up lock engages the detent on the MLG shock strut and a pintle on the main gear opens the sequence valve (SV1). This directs up line pressure to the inner door jack, which operates to close the inner door.

Down selection. The selector valve spool moves to the left, directing hydraulic pressure to the down lines, releasing the NLG up lock and pressurising the NLG operating jack to extend the nose gear. Before the main gear is extended the inner doors must be opened by the door jack. When the door reaches the full open position a pintle contacts sequence valve 2 (SV2), opening it and allowing hydraulic pressure to release the MLG up lock and extend the main gear. It will be noted that a one-way restrictor valve is fitted in the up line between the selector valve and the MLG jack. This restricts the flow of fluid returning from the jack to the reservoir during MLG extension, thus limiting the rate of travel of the heavy main gear which, with gravity added to hydraulic force, would otherwise be excessive. During retraction the one-way restrictor permits full fluid flow.

13. Accidental retraction of the landing gear, due to inadvertent UP selection with the aircraft on the ground, is prevented by weight-on safety switches, commonly called squat switches, (which isolate the selector switch when aircraft weight is on the wheels) or by mechanical locking devices inserted by ground maintenance staff.

Landing Gear Control and Indication

14. The landing gear control panel contains a selector handle with a knob in the shape of a wheel for UP or DOWN and indications of landing gear position - down and locked, travelling (unlocked) and up and locked. The gear operating handle may be locked when the gear is down and the aircraft weight is on the wheels. It will only be unlocked when the aircraft is airborne and its weight is off the wheels. In order to compare operator demands with aircraft conditions (airborne or on the ground) a logic circuit is provided. Three-position selector handles (UP/OFF/DOWN) are often used for additional safety. The OFF position permits depressurisation of the landing gear retraction system in flight.

15. Landing gear position is displayed by means of one indicator for each gear (nose, left main, right main) and often takes the form of three lights which illuminate green when the gears are down, and locked in the down position. Whilst the gears are travelling from UP to DOWN, or vice versa, a red light illuminates to indicate 'gear unlocked'. This sometimes takes the form of a flashing red light in the gear operating handle. When the gear is locked up, all lights are extinguished. An alternative type of indication is shown at Figure 2-5. The three gear indicators show:

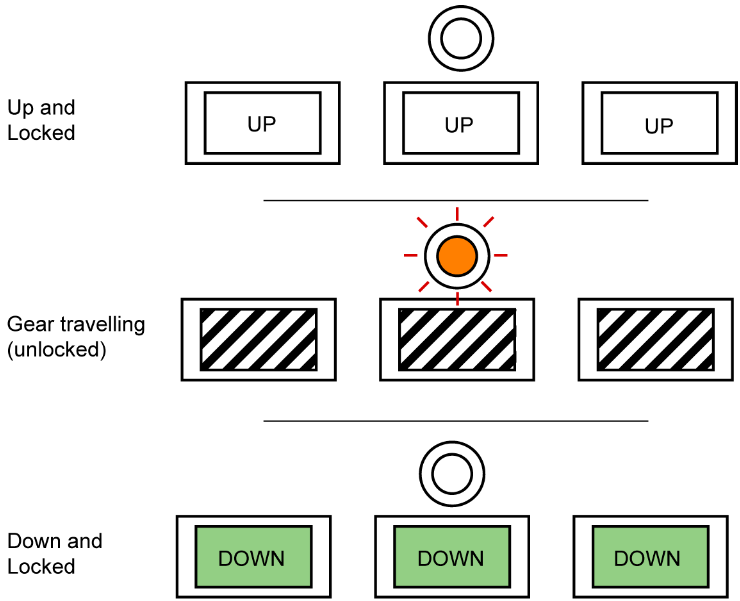
(a) up and locked

(b) unlocked (travelling)

(c) down and locked

16. The unlocked condition is accompanied by a red warning light.

FIGURE 2-5
Landing Gear Control & Indication



Landing Gear Warning System

17. The warning system provides visual and aural warning that an unsafe landing condition exists. The red landing gear warning lights come on and a warning horn sounds whenever the gear is not down and locked and the aircraft is not safe to land. Under certain conditions the warning horn can be cancelled.

18. The landing gear red lights illuminate when the throttles are retarded. The warning horn sounds with a combination of flap extension and throttles retarded.

Nosewheel Steering

19. Practically all tricycle undercarriage aircraft incorporate a steering mechanism for the nose-wheel. Larger aircraft invariably do, and some very large ones have main (body) gear steering also. The latter is necessary when the main gear uses several wheels in tandem.

20. In all but the lightest aircraft the nose-wheel steering commands are transmitted hydraulically to a yoke, or steering arm, attached to the nose-wheel shock strut. Light aircraft are steered by push pull rods connected to the rudder pedals.

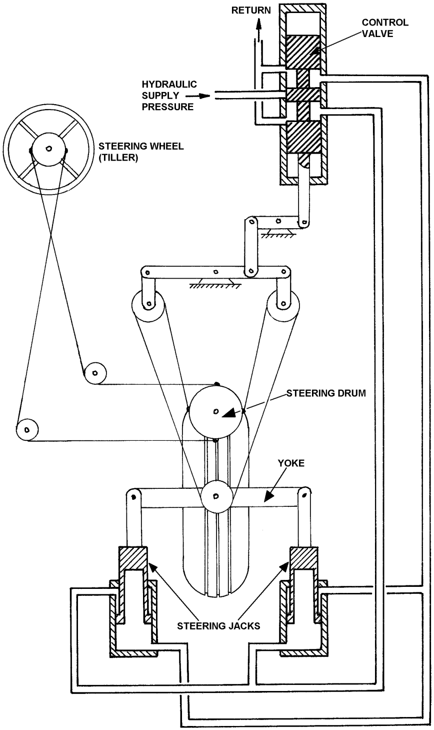
21. A schematic diagram of a nose wheel steering system is shown at Figure 2-6. Rotation of the pilot's steering wheel is transmitted by cables to a steering drum which, through a system of pulleys and linkages, moves the hydraulic control valve. This directs hydraulic pressure to the steering jacks which rotate the yoke attached to the lower cylinder of the NLG shock strut. As the yoke rotates, its motion is transmitted through the pulley system to return the control valve to the neutral position when the desired degree of turn has been reached by the nose wheel. In many transport aircraft the steering wheel or tiller is only used for large steering inputs, small inputs being transmitted through the rudder pedals. For aircraft ground towing, when the nosewheel must be able to caster freely through wide angles, the connection between upper and lower shock strut cylinders, provided by the torsion links, is broken by the removal of a quick-release pin. It is an essential pre-taxiing check to ensure that this pin is engaged.

22. Nose-wheels, especially single wheels, have a natural tendency to 'caster', that is a natural self-centring stability. Damping is required however to prevent oscillation about the centre line, a condition known as 'shimmy'.

23. Nose-wheel shimmy would exert considerable force on the steering mechanism and make steering the aircraft difficult. The castering tendency is snubbed by means of a shimmy damper. This is usually in the form of a hydraulic cylinder containing a piston and filled with hydraulic fluid. The piston rod is connected to a fixed part of the airframe and the cylinder is attached to the nose-wheel leg. A small orifice in the piston allows restricted flow of fluid from one side to the other, dampening piston movement. More complex dampers are sometimes used in large transport aircraft, consisting of a system of rotary vanes and known as a vane-type damper.

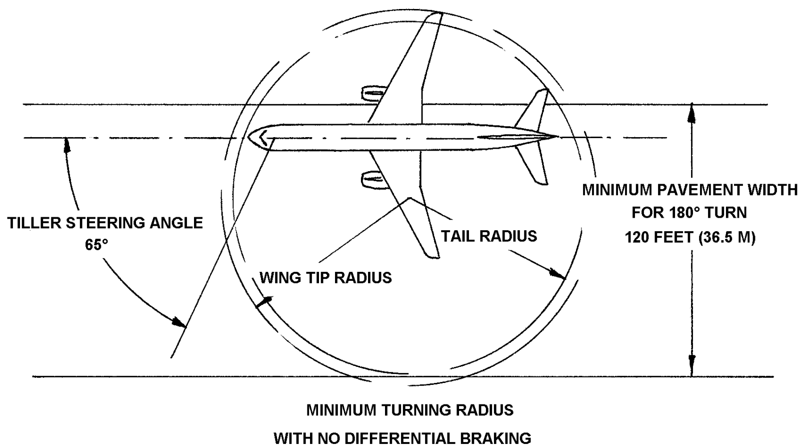
24. Note from Figure 2-6 that when the control valve is in the neutral (mid) position as shown, the steering jacks are connected to the return line and therefore the nosewheel is free to caster. This is necessary to permit directional control by rudder at high speeds whilst on the ground.

FIGURE 2-6
Nosewheel Steering



Turning Radius An important feature of nose-wheel steering is its effect upon aircraft turning radius, and particularly wing tip clearance. When the aircraft is being turned by means of the nose-wheel, the path followed by the outer wing tip will pass well ahead of the track of the nose-wheel, and therefore of the pilot, whose seat position is also forward of the nose-wheel. Consequently the pilot must allow sufficient clearance from ground obstructions to accommodate this. In the Boeing 757 for example, both outer wing and tailplane tip radius are greater than nose radius. This is shown at Figure 2-7.

FIGURE 2-7
Aircraft Turning Radius

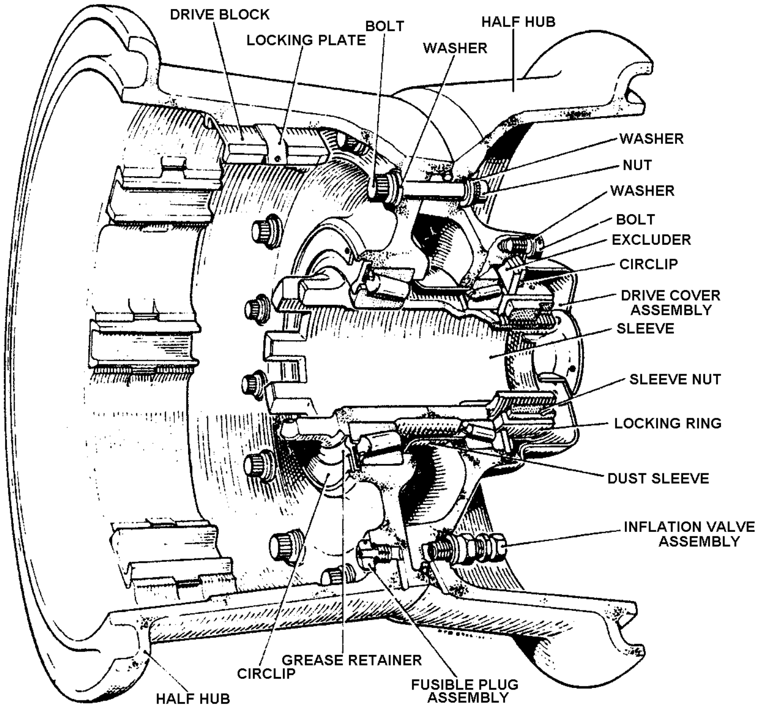


25. Factors which will affect the turning radius of an aircraft are taxi speed, gross weight, centre of gravity position, nosewheel steering angle, distance between nose and main wheel centres and track width between main wheels.

Wheels and Tyres

26. Aircraft wheels are usually constructed from forgings or castings made from aluminium or magnesium alloy to minimise aircraft weight. The most critical part of a wheel is the bead seat area which is rolled, pre-stressing its area, thus increasing its strength to protect from tensile loads applied by the tyre. Most large aircraft use tubeless tyres mounted on split-hub wheels. An example of such a wheel is shown at Figure 2-8.

FIGURE 2-8
Split-Hub Wheels

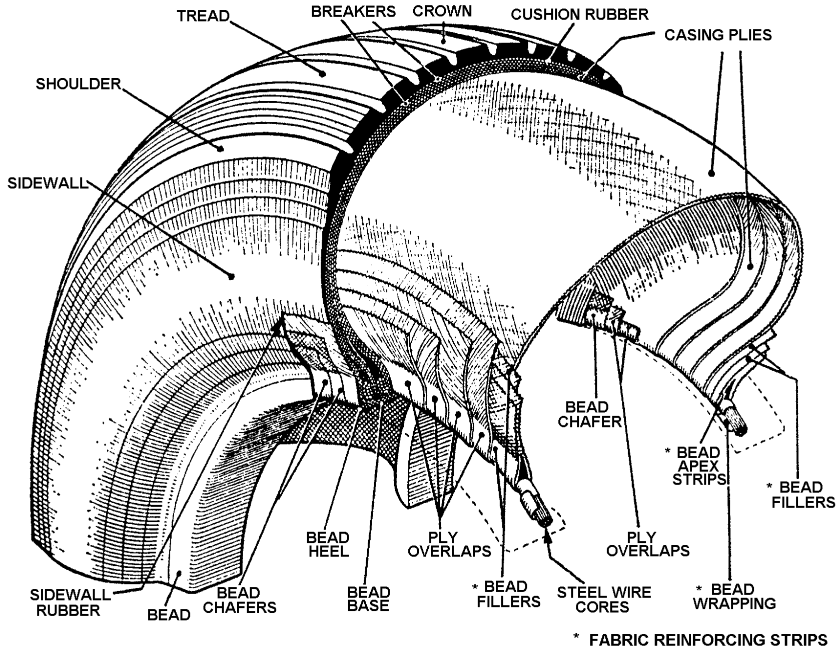


27. The two halves of the wheel are separated for installation and removal of the tyre. Obviously when assembled they make an airtight seal to contain the tyre pressure. The surface condition of the wheel flanges is also a vital factor in preventing air escaping from the tyre. The high pressures used to inflate large aircraft tyres makes the structural integrity of the wheels extremely important. Corrosion and cracking are conditions to be guarded against. Aircraft wheel hubs are made from aluminium alloy, but in some cases magnesium alloy, a material prone to rapid corrosion.

28. Aircraft tyres must withstand aircraft loads of many tons at speeds up to 250 mph. Consequently their construction is designed to withstand these loads whilst the tyre is constantly flexing during wheel rotation. Typical aircraft tyre construction is shown at Figure 2-9.

FIGURE 2-9

Aircraft Tyre Construction



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29. Tyres are classified by load, ply and speed rating. The term 'ply rating' is used to identify a tyre with its maximum recommended load and pressure. It is the index of the tyre strength and does not necessarily represent the number of cord piles used in its construction. The marking may be imprinted in full, e.g. 10 PLY RATING or abbreviated, e.g. 10PR. The speed rating is included for tyres used above 160 mph.

30. The tyre tread rubber is the abradable material in contact with the ground and may be patterned to achieve particular characteristics. The most common pattern is the ribbed tread. The ribs provide directional stability and the grooves between the ribs enable the tyre to disperse water, reducing the risk of aqua-planing.

31. Tyre tread pattern is generally limited to a ribbed or patterned variety for aircraft tyres. The tread of the tyre refers to the area forming the crown and shoulder. The most popular tread pattern is the ribbed variety which is formed from circumferential grooves around the tyre. A ribbed tread provides good traction, long tread wear and directional stability mostly suited for hard surface runways.

32. Patterned (diamond) tread tyres are particularly suitable for unpaved airfields.

33. Some nose wheels are fitted with a water deflector (or **chine**) on the upper sidewall to deflect water away from rear mounted engines.

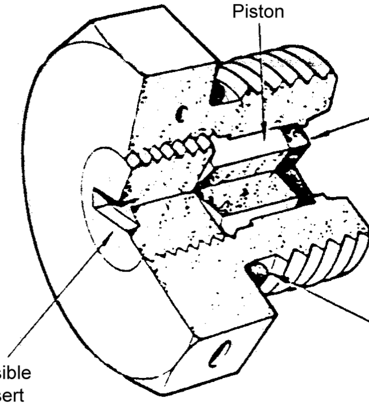
34. Twin contact tyres are used on nose wheels or tail wheels to prevent shimmy.

35. Tyre flexure, and friction due to contact with the ground, are causes of tyre wear and deterioration. The condition of the tyre must be sufficient to withstand the dynamic and static loads of supporting the aircraft. Damage to aircraft tyres, reducing their usable life, is caused by:

Creep. During braking and when the wheels spin up on touch-down the tyres may 'creep' around the wheel, which leads to wear of the tyre bead and can damage the inner tube in tubed tyres (fitted to some light aircraft). White creep indication marks are painted on tubed and tubeless tyres and the wheel rim (flange) in line with each other. These marks are one inch in width for tyres of up to 24 inches outside diameter and one and a half inches in width for tyres over 24 inches outside diameter. The tendency of the tyre to creep is greater when the tyre is newly fitted, and/or when the tyre pressure is too low.

Temperature. Build up of high wheel temperature during prolonged braking could result in overheating of the tyre bead, and an increase in tyre inflation pressure which could result in explosive fracture of the wheel. Fusible plugs are fitted in high performance aircraft wheels. These plugs melt at a predetermined temperature and release tyre pressure. See Figure 2-10.

FIGURE 2-10
Fuseable Plug



Wear. If a tyre is over-inflated it suffers excess wear on the crown. If the tyre is under-inflated it wears on its shoulders (rim of the crown). Locked wheels and the spin up on touch-down cause scuffing of the tread. It is recommended that tyres be removed when wear has reached the limits defined below:

(i) Patterned tread tyres may be used until the tread is worn to the depth of the pattern.

(ii) Ribbed tyres with marker tie bars may be used until worn to the top of the tie bars.

(iii) Ribbed tyres without marker tie bars may be worn to within 2mm of the bottom of the wear indicator groove.

(iv) Twin contact tyres may be used until the centre of the crown shows sign of contacting the ground.

Cuts. Foreign objects on the runway/taxiway can cut into the tyre tread.

Contamination. Leakage of oil (especially some hydraulic oils) and solvents onto tyres will destroy the rubber casing.

Tyre Checking Procedures

Tyre Inflation

36. Tyre inflation pressure is given in the aircraft operating manual. This is always the inflation pressure with the wheel not supporting the aircraft weight. When tyre pressure is adjusted with aircraft weight on wheels an allowance of 4% should be added to the rated inflation pressure. A tolerance of 5% to 10% above this loaded inflation pressure is generally specified and tyre pressures up to this maximum are permitted. Tyres should be inflated with nitrogen for safety.

37. If tyre pressures increase as a result of heating, due to prolonged taxying or heavy braking, the excess pressure should not be released as this could result in under-inflation at normal temperatures.

38. When checking the pressure of cold tyres which are at ambient temperature, any tyre which is more than 10% below loaded inflation pressure should be rejected, together with the companion tyre on the same axle. Any tyre which is between 5% and 10% below loaded inflation pressure should be re-inflated to the correct pressure and checked at the next daily check; if the pressure is again more than 5% low the tyre should be rejected.

39. When it is necessary to check the pressure of tyres that are still hot following a landing the pressure of each tyre should be checked and noted and compared with the pressures of the other tyres on the same undercarriage leg. Any tyre with a pressure 10% or more below the maximum recorded on the same leg should be re-inflated to that maximum pressure, but should be rejected if a similar loss is apparent at the next check. A typical tyre pressure for a commercial aircraft is in the range 150-250 PSI. Aircraft with electronic instrument systems (glass cockpit) may include a tyre pressure indication system.

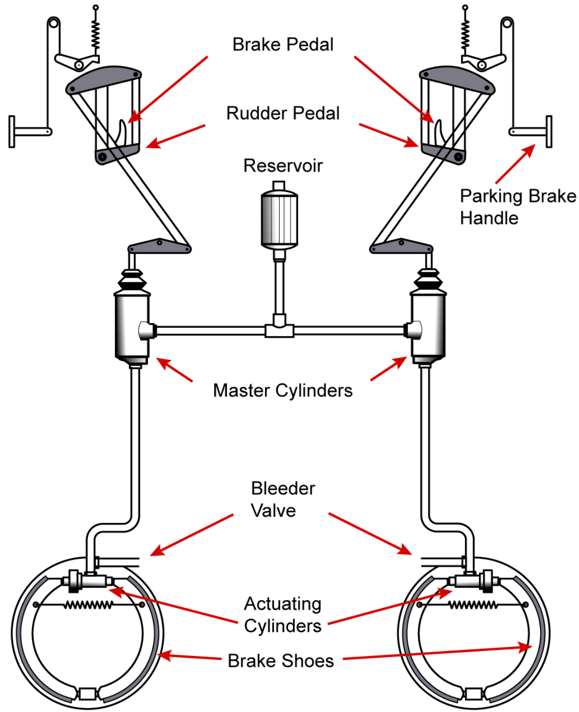
Tyre Venting

40. Tubeless tyres are vented to release air trapped in the casing during manufacture or by normal permeation through the inner liner. The awl hole vent positions are marked by green or grey dots on the lower side wall.

Wheel Braking Systems

41. Almost without exception, aircraft wheel brakes are of the disc type, only a few light aircraft remain fitted with drum brakes. Larger aircraft use multiple disc brakes, whereas light general aviation aeroplanes often require only single disc brakes. To avoid disc distortion due to heat, very large aircraft often use a variation of the multiple disc system known as a segmented rotor brake.

FIGURE 2-11
Unboosted Brake System



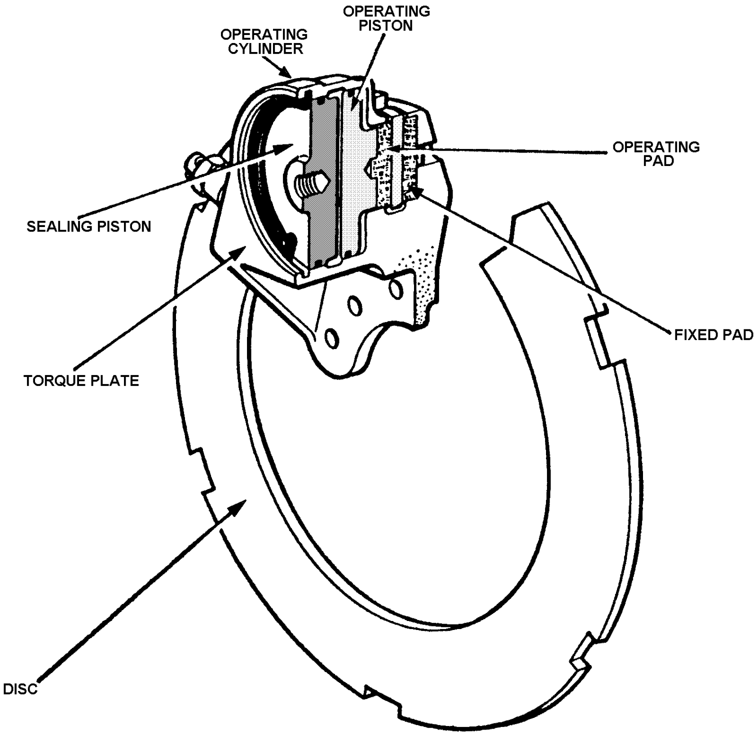
42. On light aircraft an independent or unboosted brake system may be used. A brake pedal attached to each rudder pedal permits the application of differential braking. The applied brake pressure is proportional to pedal pressure. The brakes may be locked on by applying a parking brake. See Figure 2-11.

43. The single disc brake comprises a polished steel disc which is keyed to the landing gear wheel. When the brake is applied, hydraulic pressure on a piston forces high friction pads to clamp on either side of the disc. The greater the force applied, the greater the braking friction.

44. Multiple disc brakes use a number of discs mounted parallel to each other with more hydraulically operated pads, to increase the braking friction. This is necessary when stopping a large, heavy aircraft.

45. Figure 2-12 shows a disc brake with a single disc, or brake plate. The brake plate is splined to the aircraft wheel and rotates with it. The torque plate carrying the brake operating pistons is attached to the axle and is stationary.

FIGURE 2-12
Single Disc Brake



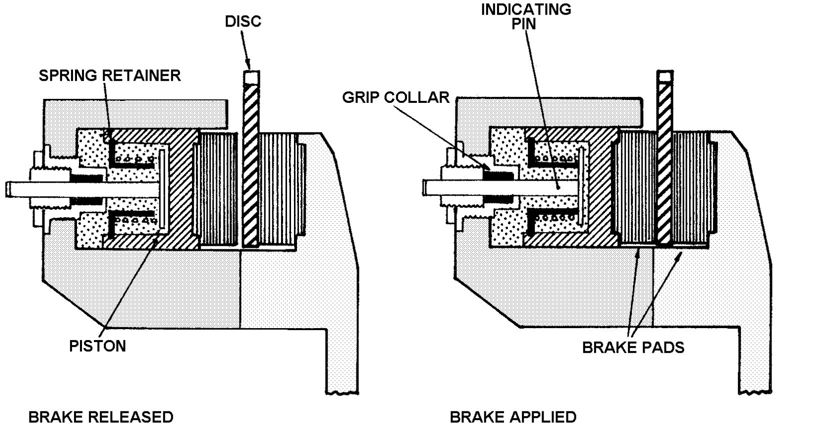
46. Application of braking causes hydraulic pressure to move the piston to the right. The brake pad attached to the piston pushes against the brake plate, which in turn moves on its splines to contact the fixed pad, effectively clamping the disc between the stationary brake pads. Release of the hydraulic pressure allows the return springs to move the brake pads axially out of contact with the disc.

Automatic Brake Adjusters

47. As the friction linings of the brake pads wear it is important that the clearance between the lining and the disc, with brakes released, is maintained constant, otherwise brake control authority would be progressively reduced.

48. An example of a single-disc brake adjuster is shown at Figure 2-13. When brake hydraulic pressure is applied to the piston it moves to the right, forcing the brake pads to grip the brake disc. The piston movement compresses the return spring until the spring retainer contacts the head of the indicator pin and pulls the pin through the grip collar in the brake cylinder housing. When brake hydraulic pressure is released the return spring moves the piston back until it contacts the head of the indicating pin. The return spring is not strong enough to overcome the friction of the grip collar, so the pin limits the return travel of the piston. The more the brake pad linings wear, the more the pin is pulled through the grip collar, thus maintaining a constant clearance between pad lining and disc. The amount of pin external protrusion is an indication of brake pad wear.

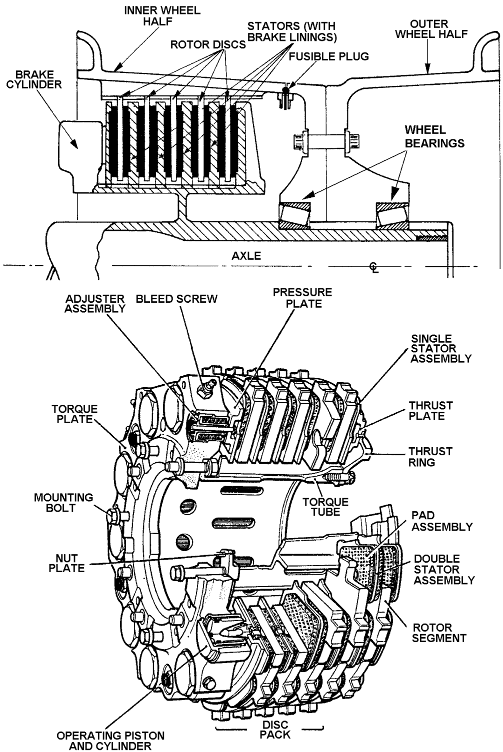
FIGURE 2-13
Single-Disc Brake Adjuster



Multiple Disc Brakes

49. The heavier the aircraft the greater the kinetic energy that must be converted into heat energy by the wheel brakes. This conversion takes place through friction between brake pads and wheel discs, thus more pads and more discs are necessary with heavy aircraft. A multiple disc brake is shown in the diagrams at Figure 2-14.

FIGURE 2-14
Multiple Disc-Brake



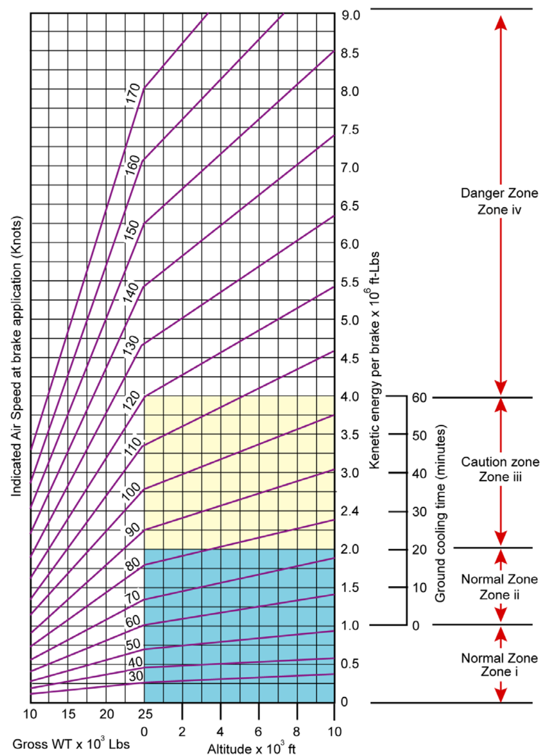
50. The majority of large aircraft have main landing gears that retract inwards towards the aircraft centre-line. To reduce the stress on the wheel spinning before gear retraction takes place, the main wheels are braked. This braking function is achieved by an autoretract brake system. Nose gears that retract fore/aft usually have de-spin friction pads contacted by the tyres when the gear reaches the up position.

Brake Energy Capacity

51. Stopping a high speed aircraft either after landing or on a rejected take-off involves the conversion of considerable kinetic energy into heat at the brake units and main wheels. This energy may be expressed in foot-pounds or joules. An aircraft may have a brake limitation chart to provide flight crew and maintenance personnel a means of determining how to deal with hot brakes safely and effectively.

FIGURE 2-15

Brake Limitation Chart



52. The specific purpose of the chart is to avoid in-flight fires and to ensure adequate brake capacity at all times for a rejected take-off. The chart determines the amount of energy to be absorbed by the brakes by considering the aircraft gross weight, indicated airspeed and density altitude at the time the brakes are applied. A condition zone for a braking event may be classified as normal, caution or danger.

The zones may be summarised as follows:

(a) Normal zone, zone I: Below 1.0 million ft.lb, no special requirements.

(b) Normal zone, zone II: 1.0 million to 2.05 million ft.lb. Allow a brake cooling time as shown before attempting a take-off.

(c) Caution zone, zone III: 2.05 million to 4.0 million ft.lb.

(i) Move aircraft clear of active runway.

(ii) Use brakes sparingly to manoeuvre.

(iii) Do not set parking brake.

(iv) Allow brake cooling time as shown.

(d) Danger zone, zone IV: over 4.0 million ft.lb.

(i) Clear runway immediately, as fusible plugs will blow 2 - 30 minutes after stop.

(ii) Do not apply dry chemical or quench until fusible plugs have released tyre pressure.

(iii) Do not approach for 30 mins or until fusible plugs have blown.

(iv) Allow 2 to 3 hours cooling to permit safe removal.

(v) Replace tyres, wheels and brakes.

Anti-Skid Systems

53. The function of the wheel brakes is to convert the kinetic energy of the aircraft into heat energy, through the friction in the brakes. If the wheel stops rotating (locks) and the tyre skids on the runway, the brakes have ceased to function and the energy transfer is now only between tyre and runway. Furthermore, on a wet runway directional control of the aircraft may be lost.

54. To prevent the wheels locking during braking, transport aircraft braking systems include skid control, or anti-skid systems. The principle of operation of such a system requires a device to measure wheel rotational speed (the skid-control generator) and to apply the brakes in proportion. As rotational speed diminishes, braking force is reduced sufficiently to just prevent wheel-locking.

55. The skid-control generator consists of a small DC or AC generator mounted in the wheel axle. The voltage output of the generator (and frequency in the case of AC) will be directly proportional to wheel rotary speed. This is fed as a signal to the skid control unit which compares it with the pilot's braking demands. If there is no wheel skid developing, the braking action is proportional to the pilot's pressure on the brake pedals. If a skid is developing, the skid control unit activates valves to release some of the brake actuating pressure to prevent the skid developing further. This is called pressure bias modulation.

56. Clearly, the anti-skid sensing system can only function if the wheels are rotating in the first place. If, for example, the aircraft were to touch down with wheel brakes applied, the wheels would skid and the anti-skid system would have no way of sensing this, so the wheels would remain locked. A protection circuit in the control unit prevents the brakes from being applied during the landing approach. This circuit is called touch down protection or touch down control.

57. Should any wheel lock fully when the aircraft is rolling, as can happen on a patch of ice, the anti-skid system will release the brakes fully on that wheel until it spins back up. This is known as locked-wheel skid control and is only functional at aircraft speeds above 15 to 20 mph.

58. In the case of failure of the anti-skid system a warning light or caption is activated on the flight deck and the brake system becomes fully manual.

59. A schematic diagram of an anti-skid system is shown at Figure 2-16. At Figure 2-17 is a schematic layout of the wheel braking and anti-skid system for a Boeing 757.

FIGURE 2-16

Anti-Skid System

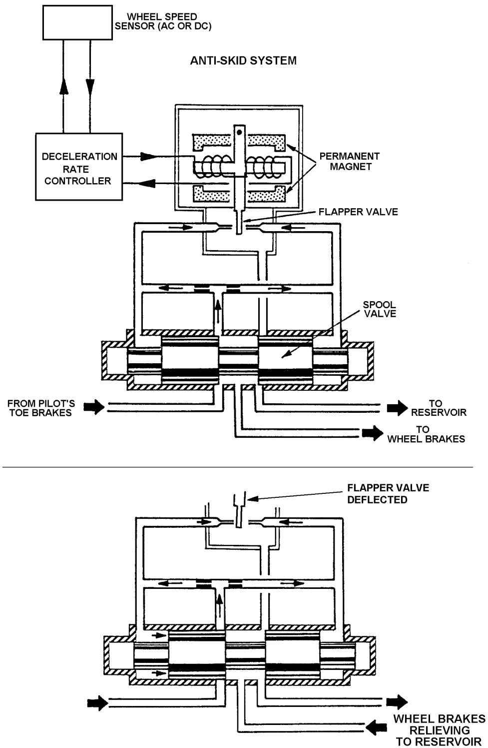
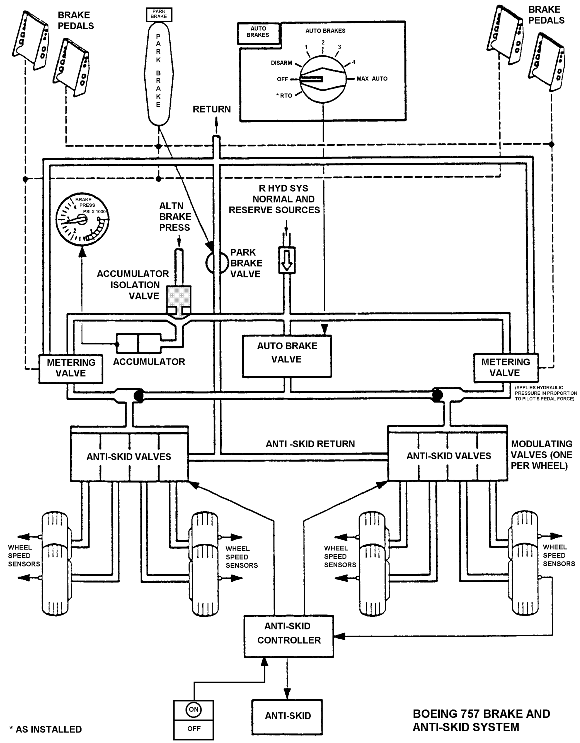


FIGURE 2-17

Anti-Skid System (B757)



BOEING 757 BRAKE AND ANTI-SKID SYSTEM

60. The output of the wheel speed sensor is fed to a deceleration rate controller, which compares actual deceleration rate of wheel rotation with a preset reference value. If the wheel deceleration rate (spin-down) is within limits there is no output from the rate controller and a permanent magnet holds a flapper valve in its mid-position. This allows hydraulic pressure from the spool valve to escape equally from the jets on either side of the flapper and maintains equal pressure at either end of the spool valve, centralising it. In this position, hydraulic pressure from the pilot's brake pedals is transmitted directly to the wheel brake cylinders. If the deceleration rate exceeds the preset value the rate controller produces an output signal which biases the permanent magnet field and causes the flapper valve to tilt, as shown in the lower diagram in Figure 2-16. This results in an increased pressure on the left side of the spool valve and a decrease on the right, which causes the spool to move over, relieving the brake cylinder pressure and releasing the brakes, thus preventing the wheels decelerating to a locked condition. Once the deceleration rate is back within limits the controller will restore normal brake operation.

Parking Brake

61. The purpose of the parking brake is to hold the aircraft stationary while the pilot is not operating the brake pedals. Applying the parking brake routes brake hydraulic pressure to the wheel brakes to hold them firmly ON so long as brake hydraulic pressure is available.

62. Operation varies with aircraft types, but in general the procedure is to depress the toe brake pedals fully and apply the parking brake, which holds the brake control valves in the fully ON position. It should be noted that, in many aircraft, application of the parking brake (even partial application) cuts out the anti-skid system by closing the return line from the anti-skid valves.

63. With parking brake applied a warning indication is illuminated on the flight deck, together with anti-skid failure warnings. Parking brakes usually only operate main wheel brakes, and in aircraft with multiple main wheels often only some of the main wheel brakes are operated by the parking brake. See Figure 2-17.

Normal, Alternate and Reserve Systems

64. The power for operating the wheel braking system is provided by the aircraft hydraulic system. Normal braking hydraulic pressure is supplied by one of the main hydraulic systems (for example system A) and alternate braking by another (for example system B).

65. The brake hydraulic system(s) always include an accumulator and a non-return valve. In the event of loss of hydraulic supply the non-return valve prevents pressure loss from the brake system to main system and the accumulator holds sufficient reserve pressure for a number of brake applications. In many modern transport aircraft reserve braking is also available by connecting an electric hydraulic pump to a reserve supply of fluid in the event of loss of main hydraulic systems. In some cases emergency brake operation employs pneumatic pressure.

Indications

66. Flight deck brake system indications are usually of brake system (accumulator) pressure, and brake temperature. Warning indications of failure of normal supply and the anti-skid system are always provided and may include aural as well as visual warnings. Aircraft with electronic instrument systems may have tyre pressures displayed. The system has a pressure transducer in each wheel which sends a signal corresponding to tyre pressure to a computer for display on a page showing landing gear information.

67. Brake wear is indicated at the brakes. A protruding wear indicator pin shows brake life remaining, the pin retracts as wear progresses. Where no wear indicator is provided, wear of the brake pads can be determined by measuring the distance between brake piston and disc with the brakes applied (see Figure 2-12). Brake temperature indications are usually numerical, increasing with increased temperature. Above a certain value a warning indication is activated, since brake efficiency decreases with increasing temperature.

68. On some aircraft with 'conventional' displays a brake temperature monitoring system is fitted which includes temperature sensors at each wheel, which feed to a central monitor and warning unit on the flight deck. The monitor has a single temperature gauge and an illuminating selector button for each wheel. The monitor is calibrated to a predetermined temperature level and the gauge normally displays the highest of the brake temperatures. If any of the individual wheel brake temperatures exceed the predetermined temperature the selector button associated with that wheel will illuminate. Pressing any of the selector buttons will cause the gauge to indicate the temperature of the brake unit on that particular wheel.

Auto Brakes

69. Modern large transport aircraft incorporate an auto-braking facility which enables the pilot to pre-select various deceleration rates. The maximum auto-brake deceleration rate is less than that available from manual braking. Anti-skid protection is maintained during auto-brake operation. The system is armed by selecting a deceleration rate, but it will only apply the brakes when the engine thrust levers are at IDLE. The auto-brake system then maintains the selected deceleration rate in conjunction with the aerodynamic speed brakes and thrust reversers. It will continue to provide braking to a complete stop or until disarmed.

70. If an anti-skid or auto-brake system fault develops, the auto-brakes disarm automatically and a warning light is displayed on the instrument panel. Brake application by the pilot will also disarm the auto brake system.

71. The auto-brake system includes a rejected take-off (RTO) selection, which can only be armed with the aircraft on the ground. With RTO selected, the auto-brake system applies maximum brake if the engine thrust levers are retarded to idle above a certain aircraft speed (typically 80 to 90 knots).

Brake Symptoms

Fading. Loss of braking action. This occurs in drum type brakes when they are hot and is due to expansion of the drum away from the brake shoes. Disc brakes are designed to resist brake fade, but this may occur with a fully worn brake during a severe rejected take-off.

Dragging. Failure of the brakes to release completely. Caused by a variety of factors including weak or broken return springs, distorted discs and air in the brake hydraulic system.

Chattering or Squealing. Instead of maintaining an even friction the brake friction varies during one revolution of the wheel. This causes a 'chatter' sound to come from the segmented brake discs. If the frequency of this chattering is high enough the brakes emit a squealing noise. The causes are warped or glazed discs or deposits of brake lining material on the discs, leading to uneven friction.

Overheating. The function of the brakes is to convert kinetic energy into heat energy. The greater the kinetic energy to be converted, the greater the heat generated in the brakes. The major single cause of brake overheating is high taxiing speeds. Overheating brakes may cause the disc to warp or cause the friction material to break up and adhere to the disc. Aircraft with very hot brakes should not be parked with the brakes applied to prevent fusing of the heat pack.

Aquaplaning

72. Aquaplaning is caused by a layer of water beneath the tyre, which can build up into a wedge and lift the tyre away from contact with the runway, thereby negating the effects of braking. It can occur in water depths as little as 0.1 of an inch and is dependent upon aircraft speed and tyre pressure. A simple formula has been derived which states that the minimum speed for initiation of aquaplaning is approximately:

$$9 \times \text{the square root of the tyre pressure in lb/in}^2$$

Thus, for a tyre pressure of 200 psi the aquaplaning speed $(9 \times \sqrt{200})$ is 127 knots

73. There are three distinct types of aquaplaning:

Dynamic. This is due to standing water where the tyre is lifted off of the runway and completely supported by the water.

Viscous. This occurs when the runway is damp and provides a very thin film of water which cannot be penetrated by the tyre. Viscous aquaplaning can occur at, or persist down to, much lower speeds than simple dynamic aquaplaning. Viscous aquaplaning is particularly associated with smooth surfaces such as the touch-down zone of the runway which is smoothed by rubber deposits.

Reverted rubber. When reverted rubber aquaplaning occurs the affected tyre(s) become tacky and take on the appearance of uncured rubber. It is normally the consequence of a long skid occurring on a wet runway, during which the friction between the tyre and the wet surface boils the water and reverts the rubber. As a consequence a seal is formed which delays water dispersal. The resulting steam then prevents the tyre from making contact with the runway surface.

Hydraulics

Hydraulics

Basic Principles

1. Most modern aircraft incorporate hydraulic systems to a greater or lesser degree. The use of fluid (hydraulic) pressure to provide power transmission has several advantages when compared with a mechanical system. With a hydraulic system the power transmitted is very large in comparison to the size and weight of the equipment required. Hydraulic lines can easily be routed into inaccessible parts of the aircraft where the use of mechanical linkages would pose engineering problems in terms of 90° turns in the transmission train, and the establishment of sufficiently strong anchor points in restricted and/or non-reinforced areas.

Hydrostatics

2. The pressure produced at the base of a static column of fluid is directly proportional to the height of the column. Neither the shape nor the cross-sectional area of the fluid container alters this fundamental principle of hydrostatics. Imagine 3 containers, each 3 metres tall and filled with fluid. One is cylindrical and 3 metres in diameter, the second is a pipe 2 cm in diameter and the third an inverted cone 3 metres in diameter at the top and 2 cm at the base. The fluid pressure at the base of each would be identical.

Principles of Hydraulic Power Transmission

3. Thanks to Mr. Blaise Pascal we know that if a force is applied to a confined fluid the resulting pressure is transmitted equally in all directions. Providing that the fluid is not compressed by the action of the applied force, then the pressure transmitted in each and every direction is undiminished by distance.

4. Pascal's law applies only to a confined fluid which is not in motion. The situation is illustrated at Figure 3-1.

FIGURE 3-1
Fluid Motion

5. Since pressure is equal to force divided by area the pressure in the hydraulic system shown at Figure 3-1 is 100 gm per sq cm throughout, since the applied force of 1000 gram is acting on a piston of 10 square centimetres.

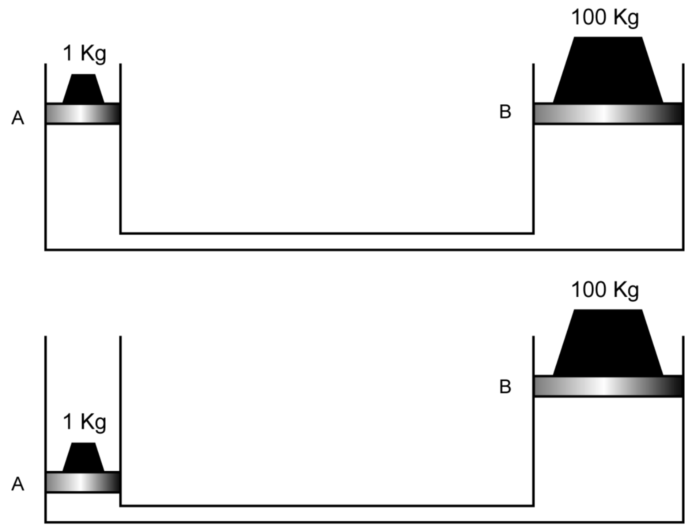
6. Since the piston at the other end of the system is 1000 square centimetres, and since, by transposition, force is equal to pressure times area, the force acting on this piston is 100 kg.

7. What we have in fact achieved is a mechanical advantage, in this case of 100:1, since an applied force of 1 kg has achieved a resultant force of 100 kg.

8. Strictly speaking Pascal's law only applies providing that nothing moves. Obviously we are interested in a system whereby an applied force causes a resultant movement, and now we discover that we have not in fact achieved something for nothing.

9. Reproducing Figure 3-1, but now assuming that we want to displace a 100 kg mass by applying a 1 kg force, we will find that it is necessary to move the piston at A by 10 cm in order to move the piston at B by 1 mm. This is shown at Figure 3-1.

FIGURE 3-2
Moving Fluid with Advantage



10. At Figure 3-2 the work applied at A will equal the work achieved at B (assuming that no work has been done either in compressing the fluid, or in overcoming friction in the system).

11. Now, since work is equal to force times distance, the work done at A (1 kg x 10 cm = 100,000 gm-mm), must equal the work achieved at B (100 kg x 1 mm = 100,000 gm-mm).

12. This is precisely the principle behind an airborne (or any other) hydraulic system. By moving the small applied force through a large distance you can move a large load through a small distance.

Hydraulic Fluids

13. Any work which is done in compressing the fluid in the system is work wasted. One of the essential qualities of a hydraulic fluid is therefore that it should be effectively incompressible throughout its entire range of approved operating pressures.

14. Other required properties of the fluid are listed below.

(a) Low viscosity

(b) Good lubrication properties.

(c) Non flammable.

(d) Non toxic.

(e) Low freezing point.

(f) High boiling point.

(g) No foaming.

(h) Stable. The fluid should show no decomposition or waxing.

(i) Compatibility. The fluid must be compatible with the elastic material used in the seals and with the metal in the hydraulic system components.

(j) Coloured. This is for easy identification and also to aid in the detection of leaks.

Types of Fluid

15. Only fluid of the type specified by the manufacturer may be used in an aircraft hydraulic system. Mineral or petroleum based oil is most frequently used in small aircraft. It is red in colour and should be used with synthetic rubber, leather or metal seals. It has good lubricating properties, is chemically stable and has additives to prevent foaming and corrosion. There is little change of viscosity with change of temperature. However, mineral based hydraulic fluid has serious flammability limitations such that, in the event of a high pressure leak, there is a serious fire hazard. Mineral oil is corrosive to natural rubber and seals of this substance must not be used with it. Synthetic rubber seals are used with mineral based fluid. Mineral based fluid is designated DTD 585 or MIL-H-5606.

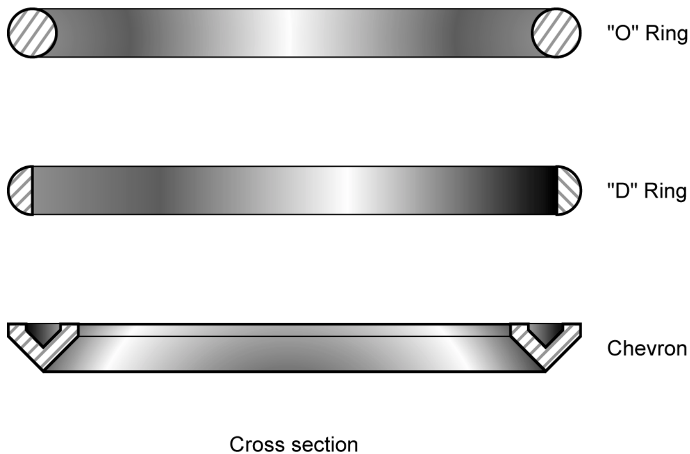
Synthetic Based Fluids

16. In high performance aircraft there is a requirement for fluids having a large temperature range. This is provided by SKYDROL. The common grade is Skydrol 500B which is light purple in colour. Other grades of Skydrol are coloured green or blue and have an operating range of temperatures varying from about -55°C to more than +105°C. Skydrol is highly fire resistant and the seals used in association with it are made of materials such as silicone, fluorocarbon or butyl. The drawback of Skydrol is its susceptibility to water and atmospheric contaminants and thus a system using it must be sealed. If overheated, skydrol turns acidic and then forms a sludge. Additionally, any fluid leak will attack the insulation of electrical wiring or anything made of polyvinylchloride (PVC). Skydrol is also very harmful to the skin and especially the eyes, and must therefore be handled with great care.

Hydraulic Seals

17. To be of any use a hydraulic system must suffer little or no leakage. Moving parts within the system, such as pistons, are fitted with seals, which prevent seepage of fluid from one side of the piston to the other. Various shapes of seal are used, the more common of which are illustrated at Figure 3-3. It is important that, as far as possible, the seal be friction free, so as to prevent any significant decrease of efficiency within the system.

FIGURE 3-3
Hydraulic Seals

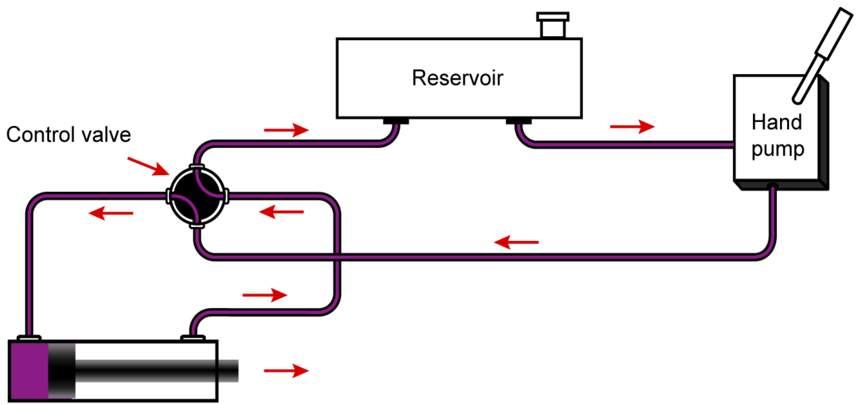


Basic Aircraft Hydraulic Systems

18. An aircraft hydraulic system is a source of pressure energy for activating a wide range of aircraft systems and components such as landing gear, wheel brakes, flaps and flying controls, and nosewheel steering. All hydraulic systems must, therefore, contain a pump to provide the pressure, a reservoir of fluid for the pump to draw upon, actuators to operate the various components listed above and selectors with which the desired functions are controlled.

19. The hydraulic system shown in Figure 3-4 is a passive hydraulic system without a pressure pump. The system is not pressurised unless the pump is operated.

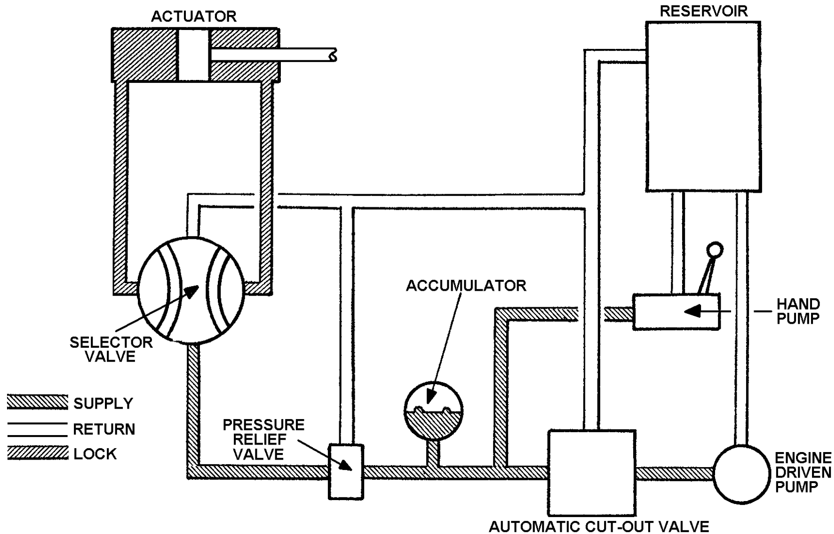
FIGURE 3-4
Passive Hydraulic System



20. As an introduction to the following sections, Figure 3-5 shows a basic hydraulic system, with the added refinements of an automatic cut-out valve to unload the pump when the actuators are not being used, an accumulator to store pressure whilst the pump is unloaded, and a hand pump for emergency operation. This is an active hydraulic system.

21. Hydraulic pumps may be powered from various sources. Pumps may be engine driven, electric motor driven or powered by an air motor. The hydraulic pipe-lines are normally stainless steel for pressure lines and may be light alloy for low pressure return lines.

FIGURE 3-5
Active Hydraulic System



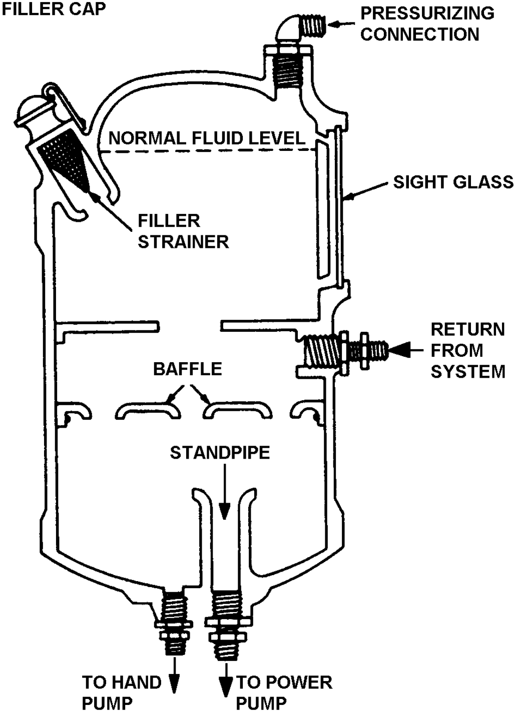
Hydraulic Components

22. The major components which are to be found in a hydraulic system are described in the following pages, together with an outline of the function or functions that they serve.

The Reservoir

23. The reservoir is effectively a storage vessel for the hydraulic fluid within a system. The reservoir will also contain an additional quantity of fluid to allow for slight losses through minor leakages. A reservoir is illustrated at Figure 3-6.

FIGURE 3-6
Hydraulic Reservoir



24. Under conditions of thermal expansion the reservoir will hold the excess volume of fluid which results. When units are actuated, for example the landing gear is raised or lowered, varying volumes of liquid are required as the piston rods move inside their jacks. The reservoir will contain the surplus fluid, or alternatively supply an increased demand.

25. During the passage of fluid round a system air bubbles may become trapped in it. The returning fluid is directed in such a way that foaming is minimised and any air in the fluid will be swirled out or extracted. The device which does this is known as a de-aerator.

26. The reservoir normally contains a screened filter to prevent foreign matter from entering the system, a vent to allow air to leave or enter the tank when the fluid level rises or falls (assuming an unpressurised system), and baffles to prevent splashing of the fluid. A sight glass enables the fluid level to be checked. Minimum and maximum level lines are normally etched onto the sight glass. The results of allowing the system to become depleted are obvious. Overfilling the system should also be avoided, since this could result in an overflow of hydraulic fluid, or possible rupture of the system if the vent could not cope with the fluid surge in the event of a high fluid volume return from a retracting piston.

27. In the event of a major leak in the system, the level of fluid stored in the reservoir would diminish. In order to prevent the power driven pump within the system forcing all of the fluid through the leak, a standpipe is fitted in the reservoir. Once the level of fluid drops to the top of the standpipe the supply of fluid to the powered pump ceases. A reserve of fluid is still available however for use with the emergency back-up system, which is frequently in the form of a hand pump on small aircraft.

28. For aircraft that fly at high altitude where the atmospheric pressure is correspondingly low, the reservoir is pressurised, typically to between 10 psi and 30 psi depending on the manufacturer. Pressurisation is provided by a filtered air bleed from the engine compressor and the desired pressure is controlled by a pressure relief valve. Pressurising the reservoir ensures that the system receives a constant supply of fluid and that pump cavitation is avoided. Cavitation occurs (typically on the inlet or suction side of a hydraulic pump) when the fluid pressure is so low that cavities form due to entrapped gas expansion. Foaming can be prevalent when the outside air pressure falls to a low level. After entering the reservoir, the fluid will be de-aerated.

The Power-Driven Pump

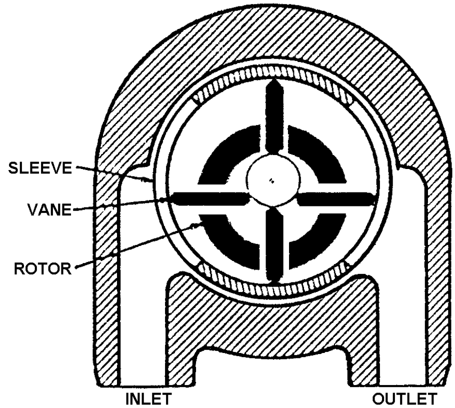
29. The powered pump in the system ensures that a supply of hydraulic fluid at the designed system pressure is always available when required, at each and every part of the system.

30. The required system pressure at all points downstream of the pump is maintained in one of two ways.

31. In the constant delivery system the pump, known as a constant displacement pump, is operating continuously and is driven by the engine. This type of pump will move a given amount of fluid for each revolution and is used when a fairly large volume of fluid needs to be moved at a relatively low pressure. It means, though, that once the required pressure has been attained a cut-out, or pressure relief valve, will be required to prevent any further increase in pressure. A hydraulic accumulator will normally be found in any system using a constant displacement pump.

Vane Type Constant Displacement Pump

FIGURE 3-7
Vane Type Pump



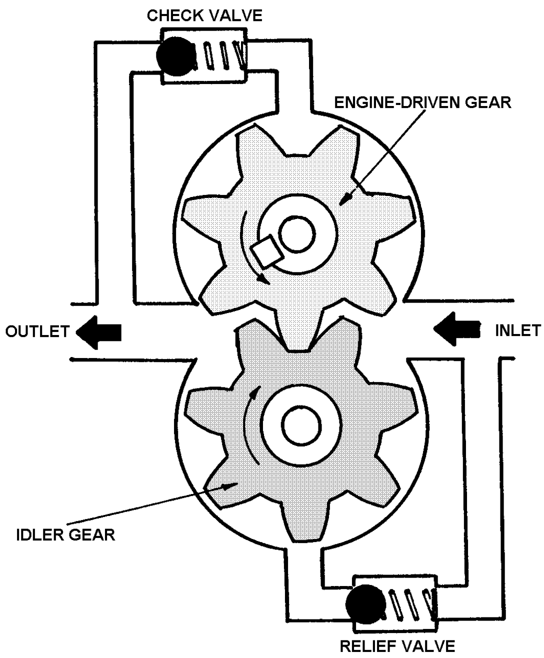
32. The vanes, which are held against the wall of the sleeve by a spacer, are free floating in the rotor. As the rotor rotates, the volume between the vanes on the inlet side of the pump will be increasing, so that fluid will be drawn in. On the outlet side of the rotor, the volume between the vanes will be decreasing and thus fluid will be forced out through the outlet port. This type of pump is illustrated at Figure 3-7.

Spur Gear Displacement Pump

33. This type of pump will be used when a medium volume of fluid needs to be moved at pressures up to approximately 1500 psi. One of the gears is driven by the engine via the engine accessory gear box and this gear will, in turn, drive the other gear within the pump. As the two gears rotate, the space between the gear teeth carries the oil from the inlet to the outlet side of the pump.

FIGURE 3-8

Spur Gear Pump



34. For purposes of cooling, sealing and lubrication, a small amount of fluid is allowed to leak past the gears. This oil is kept within the hollow shafts until the low pressure relief valve opens at approximately 15 psi. This so-called case pressure prevents air being drawn into the pump in the event of shaft or seal wear.

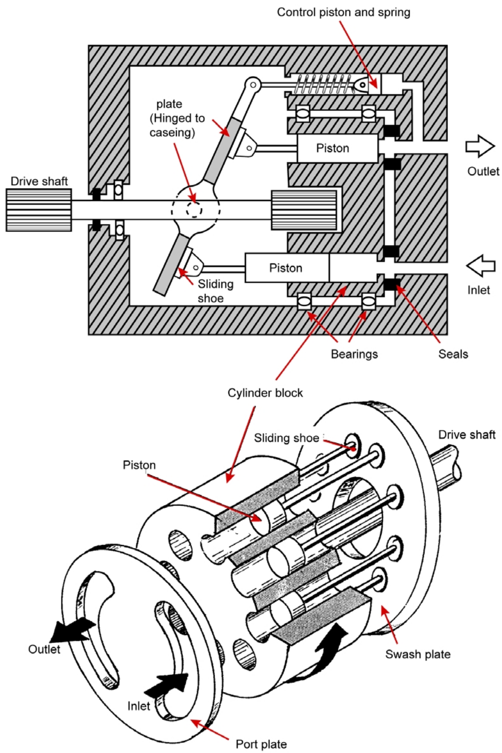
35. There is also a tendency for the case to distort as pressure builds up. On some pumps, therefore, high pressure oil is fed past the check valve and into a cavity behind the bushing flanges which are thus forced hard against the side of the gears. This will minimise leakage by decreasing side clearance and will also compensate for bushing wear. A spur gear pump is illustrated at Figure 3-8.

Variable Displacement Pumps

36. In a constant pressure live line system the pump physically ceases to operate once the required line pressure is achieved. As and when the line pressure falls the pump again delivers fluid into the system until the pressure is restored. This type of system requires no cut-out valve or accumulator for pressure control, however a pump by-pass valve is essential, to prevent over-pressurisation of the system in the event that the pump fails to cease delivery once the required line pressure is achieved.

37. The constant pressure live line system employs a variable displacement pump. In the variable displacement system, pump output volume, or displacement, is varied to maintain constant discharge pressure. An axial piston swash plate pump is often used in such systems. This type of pump is illustrated at Figure 3-9.

FIGURE 3-9
Variable Displacement Pump



38. The pistons in the rotating cylinder block are held in sliding contact with the stationary swash plate through shoes, pivoted to the ends of the piston rods, by springs acting upon the pistons. Since the shoes slide around the angled, stationary, swash plate, the pistons will move backwards and forwards in their cylinders as the cylinder block is rotated by the engine-driven shaft.

39. The extent of the piston stroke will depend upon swash plate angle, and this in turn is adjusted by pump outlet pressure acting upon a spring-loaded control piston. When the outlet pressure is zero or low, the stroke will be at its greatest and the maximum volume of fluid will be displaced. As pressure increases, the control piston overcomes spring pressure, reducing the angle of the swash plate and progressively shortening the length of piston stroke. A point will eventually be reached where the pistons will not be pumping at all. When this occurs the pump is said to be idling.

40. The same device is often used as a rotary motor. If hydraulic pressure is supplied to one side of the ported plate and the other side is connected to the return line to reservoir, the pressure acting upon the 'retarding' pistons, that is the pistons extended by hydraulic inlet pressure, will be converted into rotary motion by the shoes sliding around the angled swash plate. Alteration of the swash plate angle will alter the speed of rotation (for a given supply pressure) and reversal of swash plate angle will reverse the direction of rotation.

41. Lubrication and cooling of the pump is achieved by allowing a small amount of oil to escape from the pistons into the central chamber of the pump and thence back to the reservoir. This is generally referred to as the case drain cooling flow. The fluid can be routed via a heat exchanger in the aircraft fuel tank before returning to the reservoir. The fuel in the aircraft tank is used to cool the hydraulic fluid.

The Hand Pump

42. While the hand pump may be the only source of power in a small light aircraft hydraulic system, in larger aircraft it will be installed:

(a) As an emergency standby unit.

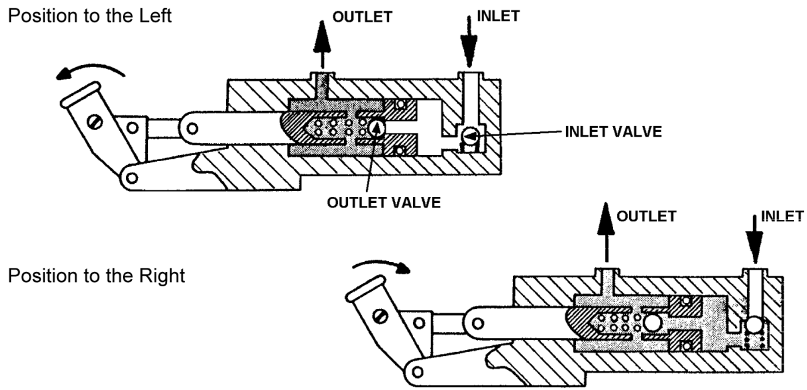
(b) To allow ground servicing without the need for a running engine.

(c) So that pressure joints and lines can be tested.

(d) So that cargo doors, undercarriage bay doors, and so on, can be operated.

43. The hand pump is normally double acting. It incorporates non-return valves and also a relief valve. In the air, should a failure occur of either the power driven pump or the power unit driving the pump, the hand pump will provide an alternative method of lowering the gear and flaps, as well as providing vigorous exercise for the most junior of the pilots. Similarly, if a major leak occurs, and the system depletes to the level of the top of the standpipe in the reservoir, the hand pump will be required. In this case fluid is drawn from the bottom of the reservoir and fed to the essential hydraulic services via the hand pump and duplicate hydraulic circuits. A hand pump is illustrated at Figure 3-10.

FIGURE 3-10
Double Acting Hand Pump



DOUBLE-ACTING HAND PUMP

Additional Safety Features

44. Because of the large number of services operated hydraulically on large modern aircraft, it is necessary to duplicate, or even triplicate pumps and some of the associated services. This will provide redundancy in the event of system failure whilst meeting the requirements of the JAR and FAA.

45. Other means of emergency operation of essential services include:

(e) Landing gear designed for free fall and lockdown under the influence of gravity, possibly supplemented by the application of positive 'g'.

(f) Electrically driven back-up pumps, with or without a hand pump to supplement them.

(g) Air, nitrogen or CO_2 bottles providing pressure via separate selector valves to selected services.

(h) A drop out ram air turbine driving a hydraulic pump.

(i) Power pack units, where all components other than the actuators are housed in one unit.

Accumulators

46. The accumulator serves four functions in a hydraulic system:

(a) It absorbs pressure surges, which occur when operation of a system component causes a pressure drop, followed by a pressure rise as the pump control responds.

(b) It provides supplemental system pressure when large fluid demands are made, by supplementing the fluid flow from the hydraulic pump.

(c) It maintains system pressure when the hydraulic pump is disconnected from the system by the Automatic Cut-Out Valve.

(d) It serves as a pressure storage unit to permit limited operation of hydraulic services when the pump is not operating. This allows, for example, flap operation for ground servicing or brake operation during towing, when the engines are not running.

47. The most common types of accumulator are the Diaphragm Type (the bladder type is essentially similar), and the Piston Type. Both types are illustrated at Figure 3-11.

48. Hydraulic fluid at system pressure is connected to one side of the diaphragm, or piston, and the other side is charged with nitrogen gas under pressure. System hydraulic fluid can flow freely into or out of the accumulator. Since gas is compressible it acts rather like a spring to absorb pressure surges, the first of the accumulator functions listed above. The remaining three functions are met by virtue of the gas pressure acting on the diaphragm or piston, forcing hydraulic fluid out of the accumulator as required.

49. The accumulator is charged with gas during ground servicing, typically to a pressure of approximately half system pressure. When the engines are started the hydraulic pumps pressurize the system and fluid enters the fluid chamber of the accumulator, depressing the diaphragm, or piston, and compressing the gas in the gas chamber. Thus, when the hydraulic system is operating the accumulator gas pressure and system hydraulic pressure are the same value.

50. When the hydraulic pumps are not operating the gas pressure flexes the diaphragm, or moves the piston, to displace fluid into the system as required. To ensure freedom of movement of the piston in the piston-type accumulator a drilling is provided for lubrication of the lower (gas) sealing ring.

51. Accumulators usually incorporate a pressure gauge on the gas side for use during charging. System hydraulic pressure is displayed on a pressure gauge which reads system hydraulic fluid pressure. This is illustrated at Figure 3-12.

52. A bypass valve may be provided to allow fluid in the accumulator to return to the reservoir, thus permitting the accumulator gas charge to be checked.

FIGURE 3-11
Diaphragm Type Accumulator

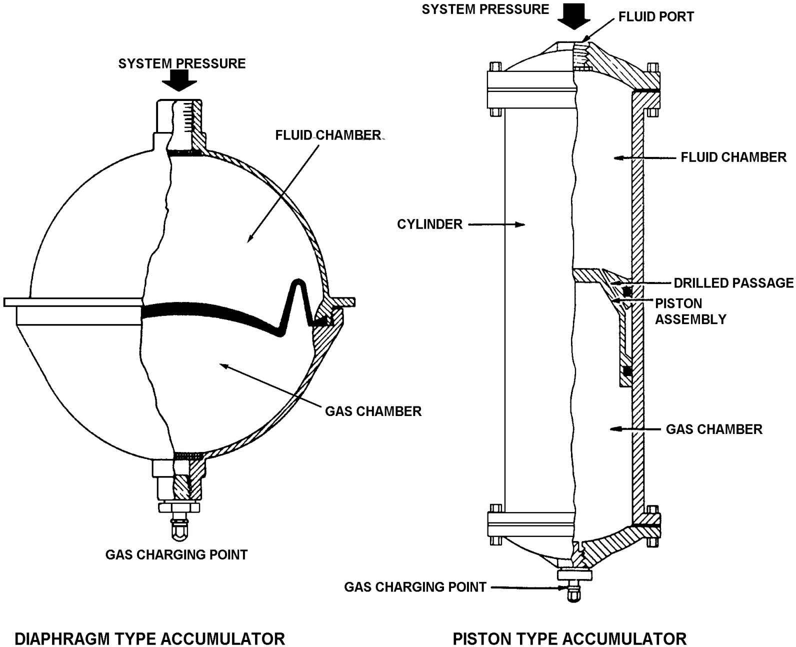
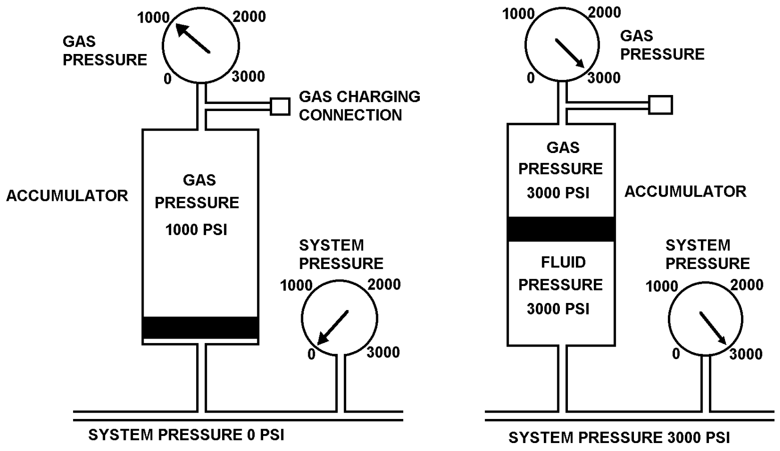


FIGURE 3-12
Accumulator Pressure



Cut-Out Valves

53. As already described, the cut-out valve is used in the constant delivery system to divert fluid back to the reservoir once the required line pressure is achieved. Figure 3-13 and Figure 3-14 show a cut-out valve. At Figure 3-13 the pump is pressurising the system, including the accumulator. At Figure 3-14 the accumulator is pressurising the system. We can see then that an automatic cut-out valve of the type shown will:

FIGURE 3-13
Pump Pressurising the System and Accumulator

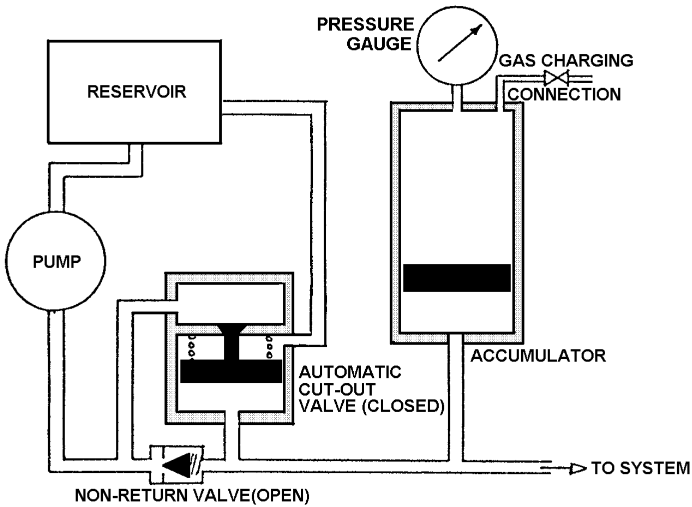
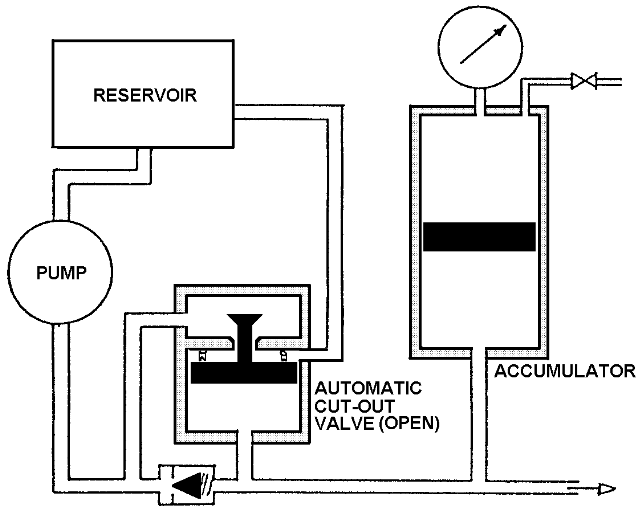


FIGURE 3-14
Accumulator Pressurising the System



(a) Provide a return line to the hydraulic reservoir.

(b) Close the return line and direct fluid into the system when pressure falls below a set figure.

(c) Give a smooth transition between pump and accumulator pressure control.

(d) Automatically cut out to off-load the pump and permit an idling circuit from pump to reservoir. This will reduce pump wear and consumption of engine power and, at the same time, prevent overheating of the hydraulic fluid.

(e) Cut in automatically when a service is selected and pressure is required to operate it. (Cut-in will also occur if a leak causes a drop in hydraulic pressure).

(f) Act as a non-return valve during cut-out periods and seal off the system to maintain pressure when the hydraulic pump is idling.

54. The interval between cut-out and cut-in of the automatic valve provides reliable information concerning the condition of the hydraulic system. A leak, whether internal or external, will cause a reduction in the period between cut-out and cut-in.

55. If the accumulator gas charge pressure is too high there will be insufficient fluid in the accumulator when system and gas pressures have equalized. This inadequate quantity of 'stored' fluid under pressure will be rapidly exhausted after a few service operations, causing the cut-out valve to cut the pump in. Pressure will be immediately restored, only to be quickly exhausted again as the fluid contents of the accumulator are once again depleted, resulting in a rapid on/off cycle.

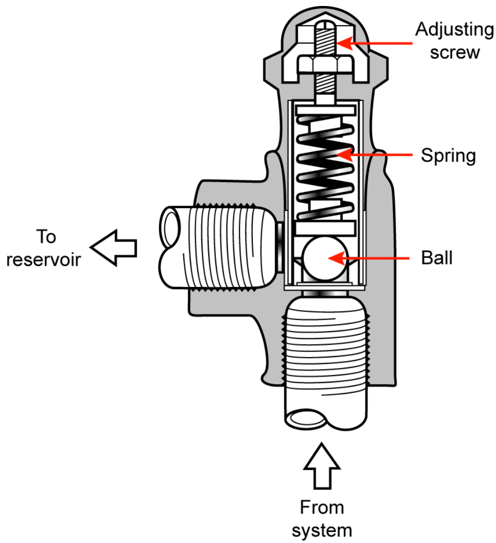
56. Too low an accumulator pressure will cause rapid fluctuations in pressure since the accumulator will no longer serve its function of absorbing pressure surges. The accumulator piston contacts an internal stop and rapid on/off cycling of the automatic cut-out valve frequently leads to 'hammering', a loud knocking noise, in the system.

Non-Return Valves

57. These valves permit the flow of hydraulic fluid in one direction only. A non return valve is shown at Figure 3-13 and Figure 3-14 as part of the plumbing associated with the cut-out valve.

Pressure Relief Valves

FIGURE 3-15
Simple Pressure Relief Valve



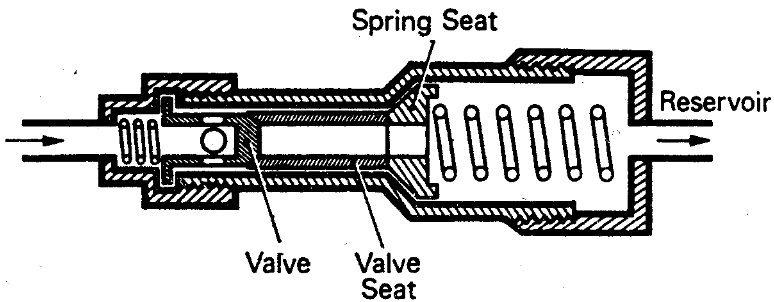
58. In order to prevent damage to the hydraulic system caused by excess pressure, a pressure relief valve is invariably incorporated into the system. It is used as a safety device and is designed to open when system pressure reaches a preset value which is slightly higher than the intended system pressure. The valve comprises a simple ball valve which is held against its seat by a spring. The spring tension is adjustable and is set to relieve a small amount of fluid to the return line to the reservoir. A simple pressure relief valve is shown at Figure 3-15.

Cracking Pressure

59. Cracking pressure is the term given to the pressure at which a pressure relief valve begins to open. For example, in a typical aircraft hydraulic system which is designed to operate at 3000 psi, the pressure relief valve might be designed to be fully open at 3650 psi and to reset at 3190 psi. The cracking pressure will therefore be somewhere between these values of the fully open and the fully reseated pressures.

Full Flow Relief Valves

FIGURE 3-16
Full Flow Relief Valve



60. A full flow relief valve protects the system against over-pressurisation due to a variable displacement pump pressure control failure which leaves the pump stuck on maximum flow. This type of relief valve must be capable of passing the maximum pump output while limiting the pressure to approximately 10% above normal working pressure. A full flow relief valve is shown at Figure 3-16.

Thermal Relief Valves

61. Just like normal pressure relief valves, thermal relief valves are designed to open when the pre-set pressure at the valve is exceeded. In this case however the increase in pressure is caused by an increase in the temperature and subsequent expansion of the fluid in a part of the system where the fluid is trapped, typically between a non return valve and an operating jack. Thermal relief valves are designed to operate at a higher pressure than the conventional pressure relief valves in the same system. The thermal relief valves do not therefore interfere with the normal operation of the system.

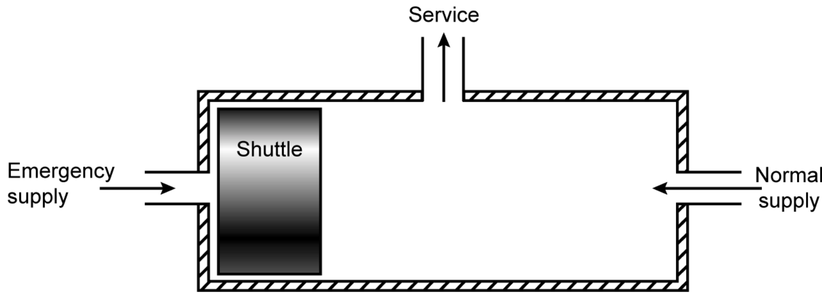
Shut-Off Valves

62. Positioned between the hydraulic reservoir and the pump, shut-off valves are normally operated electrically to cut off fluid supply to the pump. They are used to facilitate ground servicing and to isolate the fluid supply in the event of engine fire.

Shuttle Valves

63. Shuttle valves are used to disconnect one source of hydraulic fluid whilst connecting another. As already described, following a leak in the system the powered pump would become useless as the level of fluid in the reservoir dropped to the level of the standpipe. It would therefore be necessary to supply fluid from the bottom of the reservoir via the hand pump. The necessary re-routing of the circuit would be achieved using shuttle valves. In the event of a shuttle valve becoming stuck in its 'normal' position (as illustrated at Figure 3-17) it would be incapable of connecting the emergency supply to the system.

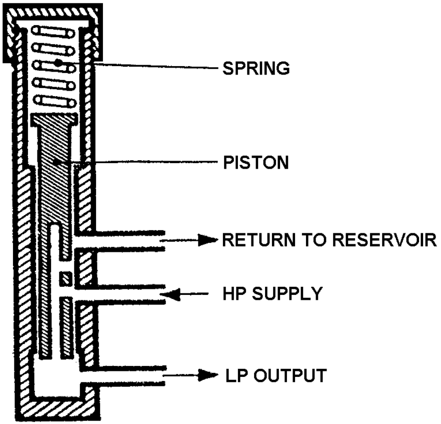
FIGURE 3-17
Shuttle Valve



Pressure Reducing Valves

64. These will reduce the main system pressure to that required for a particular service, for example the brake system.

FIGURE 3-18
Pressure Reducing Valve

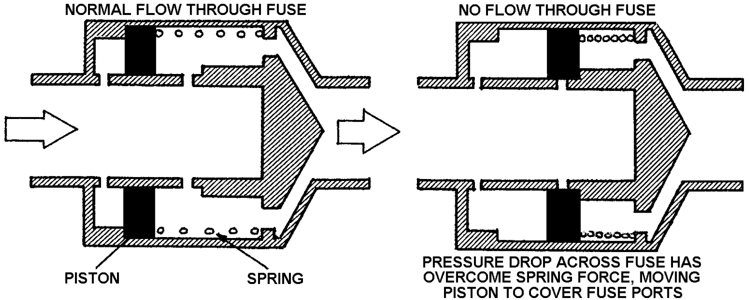


65. Figure 3-18 shows a pressure reducing valve in a non operating condition. The spring is designed to resist the required service pressure. The main high pressure (HP) supply will, by definition, deliver a greater pressure than the required service pressure. When main high pressure is supplied to the valve the piston is forced against the spring which compresses, allowing the piston to move. This movement closes off the HP supply port to a position where the pressure of fluid entering the valve is balanced by the spring pressure and the low pressure (LP) output will therefore be required system pressure. The movement of the piston will also uncover the return port to route excess fluid back to the reservoir.

Hydraulic Fuses

66. Hydraulic fuses prevent fluid loss when a leak occurs downstream of the fuse. They are often incorporated in braking, flap and thrust reverser systems and they operate when pressure drop across the fuse exceeds a preset value. Figure 3-19 shows a hydraulic fuse in the normal and closed positions.

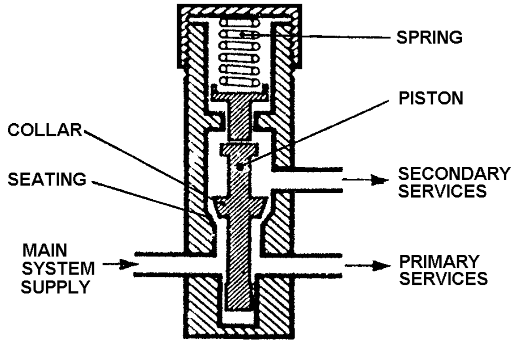
FIGURE 3-19
Hydraulic Fuse



Pressure Maintaining Valves

67. These are designed to maintain a desired pressure for some essential services by isolating supply to non-essential services if system pressure falls below a preset value.

FIGURE 3-20
Pressure Maintaining Valve

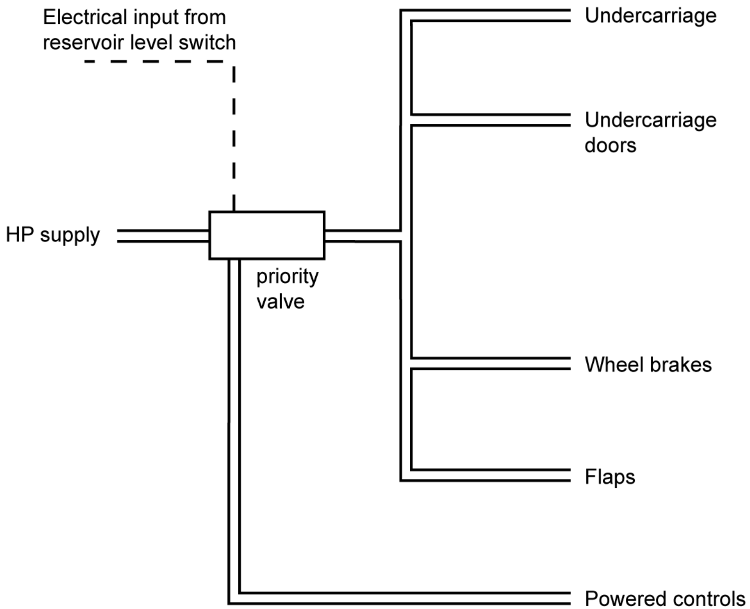


68. Figure 3-20 shows a pressure maintaining valve in the normal operating configuration. Main system supply pressure is acting on the piston and depressing the spring which is designed to balance the main supply pressure. There will therefore be pressure supplied to both primary and secondary services. In the event that the main system pressure is reduced, for example by component failure, the spring will overcome this reduced pressure and force the piston collar onto the seating, cutting off the pressure supply to the secondary services. This will allow all residual pressure to be channelled to the primary services, for example the flying controls.

Priority Valves

69. On some aircraft systems a priority valve is used instead of a pressure maintaining valve. The end result is the same but is achieved by different means. Let us assume that the system fluid content is falling due to a leak. At a pre-determined level, a switch located in the reservoir will be activated. This will transmit a signal to the priority valve which then closes off the supply line to the non-essential services. An example of a system using a priority valve is shown at Figure 3-21, where the powered controls are necessarily the essential services. The undercarriages and gear doors can be lowered under free-fall, the wheel brake accumulators will supply sufficient energy to stop the aircraft on the ground and the aircraft can land without flaps. Consequently these are considered to be non-essential services.

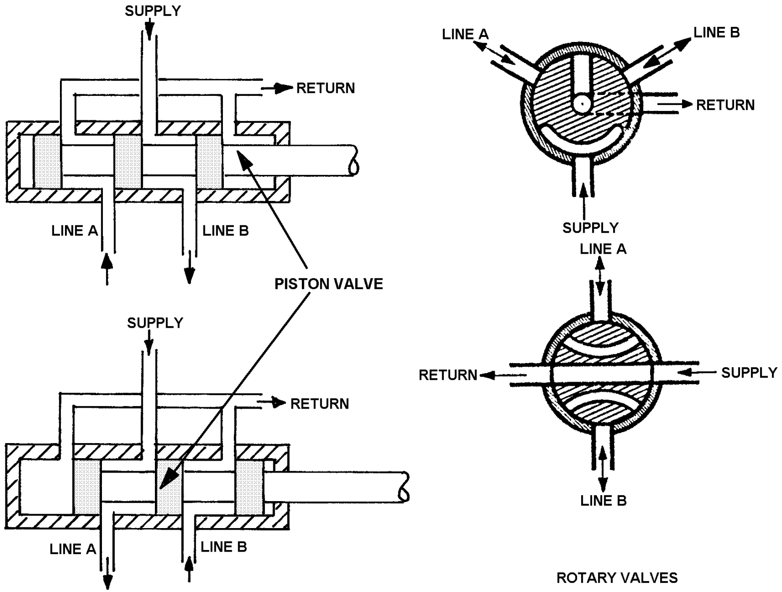
FIGURE 3-21
Priority Valve



Selector Valves

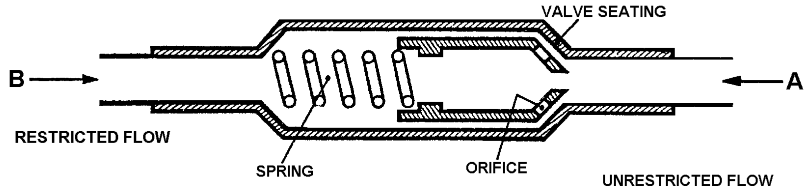
70. In a complex hydraulic system, control valves or selector valves are needed to control fluid flow. The simplest type of all is an on/off valve which opens or closes a line as required. More usually hydraulic jacks are required to operate in both directions. In such cases it is necessary to incorporate a four-way valve which permits fluid flow in either direction. Selector valves may be of rotary, poppet or piston type. Rotary and piston type selector valves are shown at Figure 3-22.

FIGURE 3-22
Selector Valve



Orifice Check Valves, or Restrictor Valves

FIGURE 3-23
Orifice Check Valve or Restrictor Valve



71. An orifice check valve is designed to provide free flow of hydraulic fluid in one direction and restricted flow in the opposite direction. Figure 3-23 shows a restrictor valve. Flow in the direction A to B compresses the spring, which opens the orifice to allow unrestricted flow in that direction. Flow in the direction B to A acts together with the spring force to move the valve onto the seating, thereby closing the orifice and restricting fluid flow in that direction. One of the most common applications of this device is in the up line of a landing gear system. Since the undercarriage is heavy it will tend to fall too rapidly when being lowered, unless some means of restricting its movement is used. Since the up line is the return line for the hydraulic fluid during the operation, any restriction in this line will limit the speed of movement of the gear. Similarly in some hydraulically operated flap systems a one way restrictor is fitted in both the flap up and flap down lines.

The restrictor in the flap up line controls the lowering speed of the flaps during flap extension to reduce the effects of aircraft trim change.

The restrictor in the flap down line controls the raising speed of the flaps during flap retraction when the slipstream tends to blow the flaps up.

Metering Check Valves

72. A metering check valve, sometimes called a one-way restrictor, serves the same purpose as an orifice check valve, but is adjustable to permit a metered flow in one direction and full flow in the opposite direction.

Sequence Valves

73. A sequence valve (or timing valve) organises a series of hydraulic operations into the required sequence. A common example of the use of this valve is in the landing gear system. The undercarriage doors must be opened before the undercarriage itself is extended, and these operations are properly sequenced by this type of valve. Examples of sequence valves will be seen in the section dealing with landing gears.

Manual Pressure Relief Valves

74. Alternatively known as off loading valves or dumping valves. They allow system pressure to be released back to the hydraulic reservoir and would be operated:

(a) to exhaust system pressure prior to gas-charging the accumulator, and

(b) to permit servicing or exchanging of a hydraulic component.

Flow Dividers

75. Their function is to divide equally the flow between two components, thus avoiding a situation where there is pressure in one line but cavitation in another.

Pressure Relay Valves

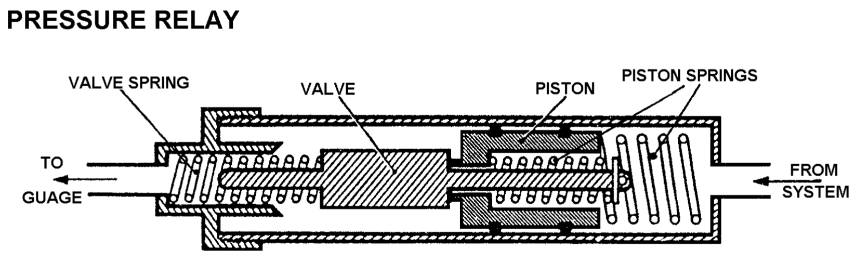
76. A pressure relay valve in the normal operating configuration is shown at Figure 3-24. These valves are fitted into pressure gauge lines and have three functions:

(a) To protect the gauge from pressure fluctuations. This is achieved by the springs within the valve, which dampen out random pressure changes.

(b) To prevent hydraulic fluid loss in the event of a burst pressure gauge. Under normal operating conditions back pressure will act on the piston and compress the piston springs. The valve spring will retain the valve in the open position. If the pressure downstream of the valve drops due to component failure the higher system pressure will move the piston in order to overcome valve spring pressure and seat the valve, thereby closing off the flow to the burst gauge or feed line.

(c) To allow servicing/replacement of the pressure gauge. Disconnecting the gauge will have the same effect as a burst gauge. With a pressure relay fitted in the line the gauge may be removed for servicing/replacement without the risk of loss of hydraulic system contents.

FIGURE 3-24
Pressure Relay Valve



Modulator Valves

77. These will be found in brake systems in association with axle mounted anti-skid units. On initial brake application, an unrestricted fluid flow is provided to match the capacity of the brake, but then fluid flow will be modulated to only allow reduced pressure. The reduced flow conserves main system pressure and allows the brake unit to completely exhaust operating pressure when the anti-skid unit comes into operation.

Hydraulic Jacks

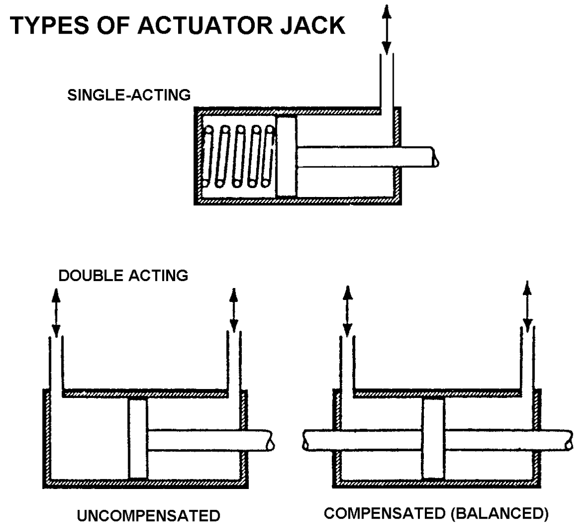
78. The purpose of any hydraulic jack is to convert the force produced by hydraulic pressure into a linear movement of a piston rod, or ram. Two basic types of jacks are commonly used as described below:

(a) Single acting jack. These move under hydraulic pressure in one direction, and in the other direction under the influence of a non hydraulic force, such as a spring. A common application of the single acting jack is landing gear and door locks.

(b) Double acting jack. In this case the movement in either direction is due to hydraulic pressure, and is controlled by means of a selector valve. The jack may be compensated (balanced) or non-compensated. In the first case the area on either side of the piston is identical, since there is a piston rod or ram on both sides of the piston. In the second case the area on the side of the piston which is remote from the rod or ram is greater than on the other side. A non-compensated system is normally used for landing gear and flap systems, where a greater force is needed in raising the gear or extending the flaps than in lowering the gear or retracting the flaps.

79. Single-acting and double-acting (compensated and non-compensated) jacks are shown at Figure 3-25.

FIGURE 3-25
Types of Actuator Jacks

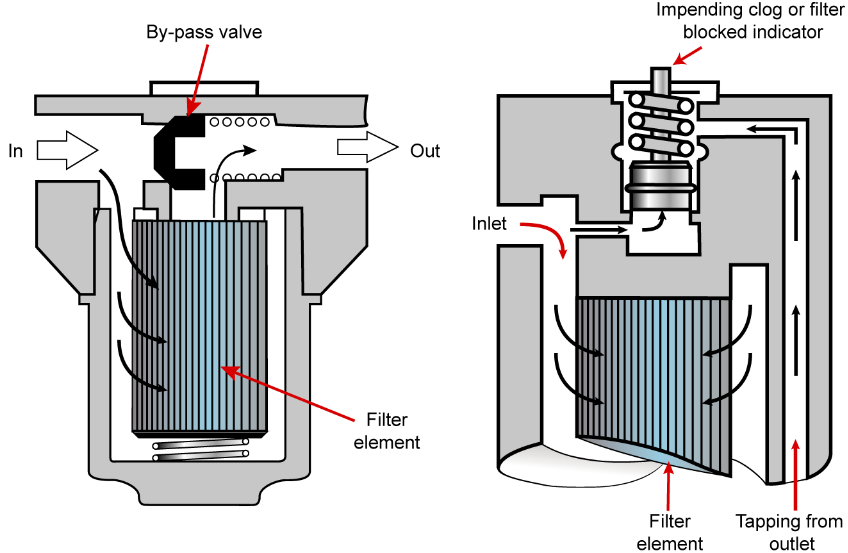


Mechanical Locks

80. Locking devices are mechanical latches, braces or pins which are automatically engaged once the required hydraulic function has been achieved. They must be automatically disengaged before the process can be reversed. An obvious example of this is the undercarriage, which is well protected by locking devices to prevent its retraction until required by the pilot, otherwise taxiing can prove a little difficult.

Filters

FIGURE 3-26
Pressure Filter



81. The tiny particles of metal, dust and seal material that accumulate in a hydraulic system would cause significant damage if allowed to circulate freely. To prevent this, filters are installed in the pressure and return lines.

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82. A hydraulic filter consists of a renewable ‘element’, usually made of cellulose material, situated in a bowl-shaped container. Hydraulic fluid enters the bowl and must pass through the cylindrical filter element in order to reach the hydraulic system. A typical return filter is shown at Figure 3-26.

83. Return filters normally incorporate a spring-loaded by-pass valve. As the element becomes clogged the pressure differential across the filter increases. When this differential reaches a pre-set value, the by-pass valve opens and allows unfiltered fluid to pass direct to the hydraulic system.

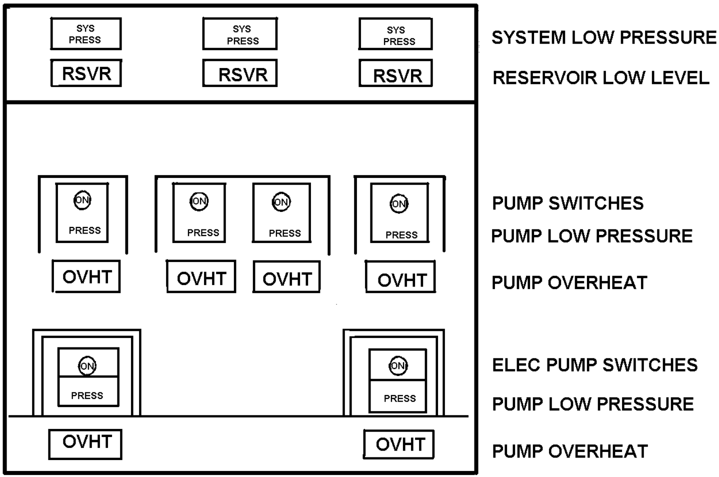
84. The clog indicator pin on some filters is restrained at low temperature (0°C) to prevent false warnings due to viscous fluid.

85. In some cases this actuates an indicator, which extends a red protruding pin, or button, about 5 mm (3/16 inch) to alert the operator to the fact that the filter element requires changing before flight. On aircraft with powered flying controls, pressure filters are non-bypass. This is to ensure that no debris is permitted to pass which could jam the control valves.

Hydraulic System Indications

86. Larger aircraft normally have a hydraulic services panel on the flight deck, containing indications of fluid quantity, pressure and temperature. Loss of hydraulic pressure, excess temperature and loss of fluid flow (indicating pump failure) may activate warning lights or captions. Figure 3-27 shows a typical hydraulic control panel for a twin-engined jet aircraft.

FIGURE 3-27
Typical Hydraulic Control Panel



87. The implications of the hydraulic system warning lights are as follows:

(a) **System low pressure.** This may be caused by loss of fluid, pump failure or pressure filter blockage.

(b) **Reservoir Low Level.** This may be low fluid level or low air pressure. Low air pressure may cause pump cavitation.

(c) **Pump Low Pressure.** This may be caused by pump failure or drive shaft failure.

(d) **Pump Overheat.** Pump swash plate stuck in high flow position. Blockage of case drain cooling flow.

88. Some large aircraft have two engine driven pumps supplying one hydraulic system. During normal operation, one pump is capable of supplying the demand and the other pump is off loaded. During periods of high demand i.e. simultaneous operation of landing gear, flaps and slats the other pump may either automatically come on-line, or be selected to operate.

Hydraulic Systems

89. Figure 3-28 shows a hydraulic system as fitted in a twin-engined executive jet type of aircraft. The engine-driven constant displacement pumps (EDPs) supply pressure to an automatic cut-out valve (ACOV), which maintains system pressure between 1250 and 1500 psi. Standby hydraulic pressure can be provided by an electrically-driven auxiliary pump, for emergency or ground operations. The reservoir is pressurised to 10 psi from the cabin air conditioning system. A system relief valve is set to begin opening at 1700 psi.

90. Figure 3-29 shows the concept usually found in larger passenger aircraft where system redundancy is important. Three independent hydraulic systems, (A, B and standby), provide power to operate the aircraft's primary flying controls, spoilers, landing gear extension and retraction, wheel braking systems, nose-wheel steering and wing flaps.

91. System operating pressure is typically 3000 psi. System A is pressurised by two engine-driven pumps and supplies hydraulic power for operation of wing flaps, outboard flight spoilers and ground spoilers, ailerons, elevators, lower rudder, landing gear extension/retraction and nose-wheel braking. It is the alternate pressure source for mainwheel braking.

92. System B is pressurised by two electrically-driven pumps and operates ailerons, elevators, upper rudder, inboard flight spoilers, mainwheel brakes, cargo door and aft passenger stairs. System B can be interconnected on the ground to pressurise System A.

93. The standby system is pressurised by one electrically-driven pump powered from the essential AC bus. It is the standby to system A and will supply operating pressure for a standby rudder actuator and the wing leading edge flaps.

94. Maintenance isolation valves may be provided to isolate sections of the hydraulic system as an aid to trouble-shoot internal leaks in the hydraulic system. An internal leak will cause a fluid temperature rise in the system.

95. It must, of course, be appreciated that the system shown at Figure 3-29 is merely a typical system and that specific aircraft systems will vary in detail.

96. In the Boeing 757, for example, there are three separate and independent hydraulic systems, left, centre and right. The left and right systems are each powered by one engine driven pump (EDP) and one electric pump. The centre hydraulic system is powered by two electric pumps with, in some aircraft, a ram air turbine (RAT) power pump as back-up. In normal operation all three systems are in use. Each system powers all the primary flying controls and some spoiler surfaces. In the event of left engine failure on take-off, a power transfer unit supplies power from the right system to retract the landing gear and lift devices. The power transfer unit transfers energy from one system to another. There is no fluid transfer. A ram air turbine powered pump (RAT) may be provided as an emergency source of hydraulic power to be used in the event of engine or engine driven pump failure. Once deployed in flight, the RAT is normally stowed on the ground by maintenance. This system is illustrated schematically at Figure 3-30.

FIGURE 3-28

Hydraulic System - Twin Executive Jet

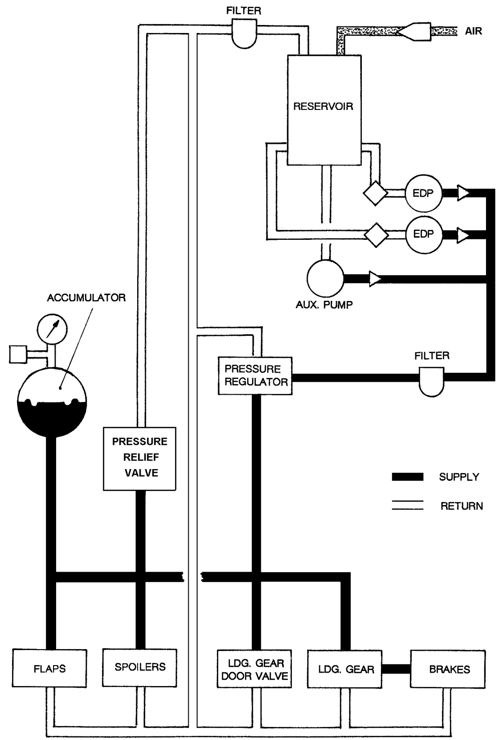


FIGURE 3-29

Hydraulic System - Typical System (B727)

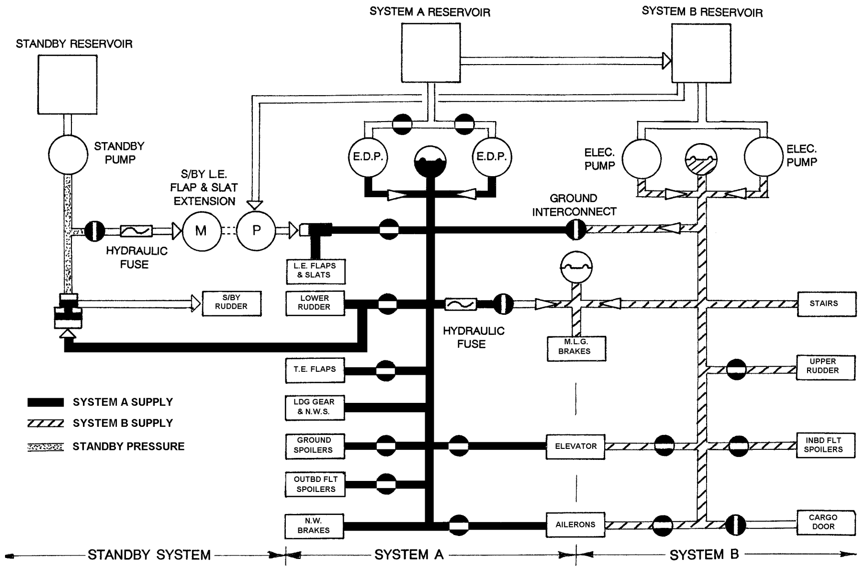
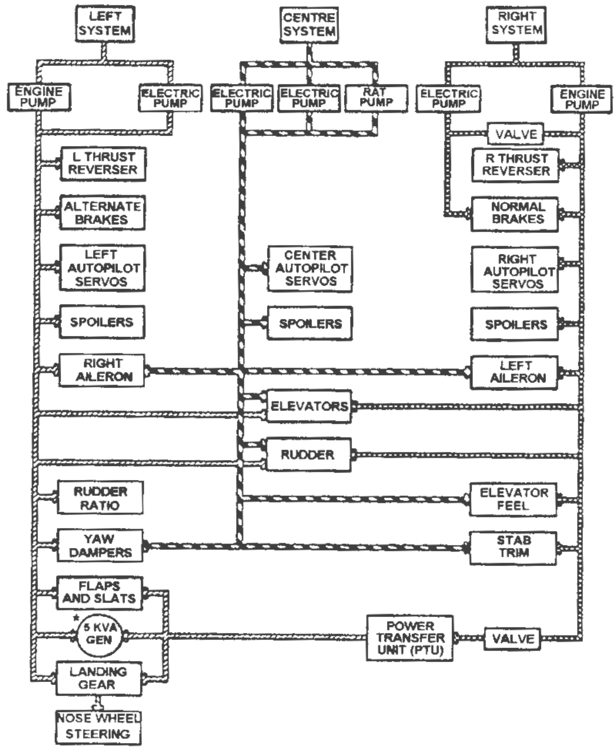


FIGURE 3-30
Hydraulic System - B757



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Light Aircraft Hydraulic Systems

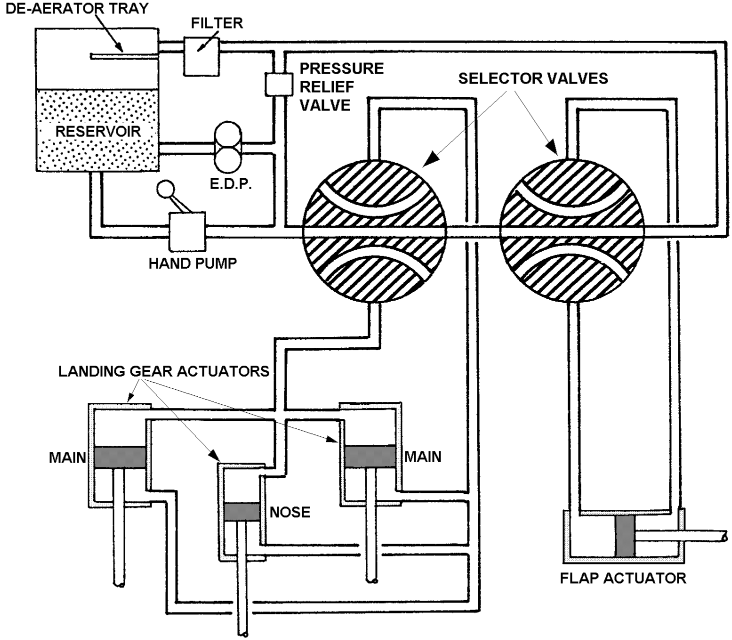
97. In order to minimise the complexity of the hydraulic systems of light aircraft, where only a few components such as flaps and landing gear are hydraulically operated, either an open centre system or a self-contained hydraulic power pack may be used.

Open Centre Systems

98. These use selector valves in series with one another. When no components are being used the selector valves are in a neutral position and hydraulic fluid can flow freely from the engine-driven pump, through the open centre of each selector valve, and return to the reservoir. Thus the pump is operating on virtually no load, with the component actuators (jacks) in a state of hydraulic lock.

99. Only when a selection is made does pump pressure build up to move the actuating jack. When jack travel is completed, pump pressure builds up further and returns the selector valve to its neutral (open centre) position. The main disadvantage of such a system is that, with the selector valves in series, only one actuator can be operated at any one time. An open centre hydraulic system is illustrated at Figure 3-31.

FIGURE 3-31
Hydraulic System - Open Centre

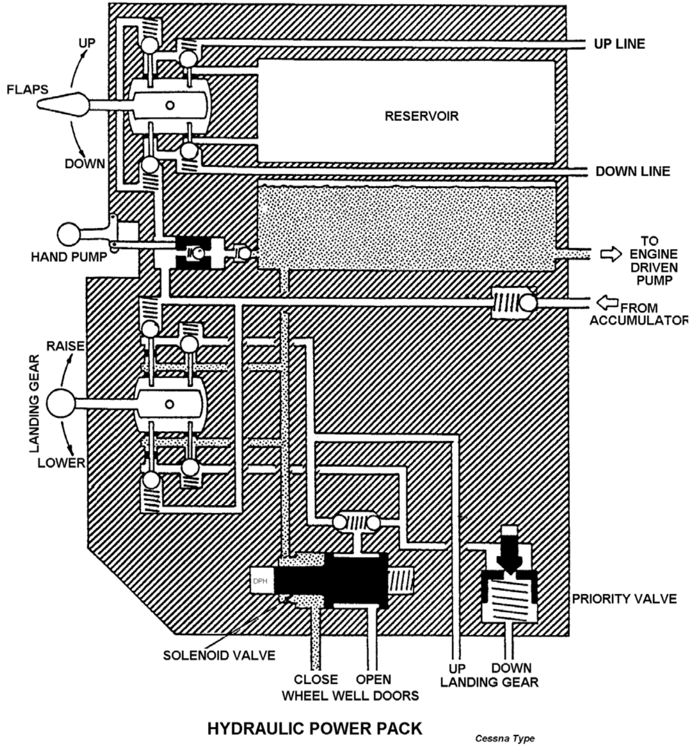


Hydraulic Power Packs

100. The trend in modern light aircraft with only limited hydraulic system requirements is to incorporate the whole of the hydraulic power system (hand pump, reservoir and selector valves) into one easily-serviced unit. A schematic diagram of a hydraulic power pack as fitted in some Cessna light aircraft is shown at Figure 3-32. The selector valves in this pack are of the poppet type.

FIGURE 3-32

Hydraulic System - Light Aircraft Power Pack



Self Assessed Exercise No. 1

QUESTIONS

QUESTION 1.

What is the function of the landing gear oleo/shock strut.

QUESTION 2.

What is the function of the bogie unit.

QUESTION 3.

What is the purpose of the drag strut on the undercarriage unit.

QUESTION 4.

What is the function of the torque link.

QUESTION 5.

How is the gear prevented from collapsing on the ground.

QUESTION 6.

What methods are used to provide alternative lowering of the landing gear.

QUESTION 7.

What is meant by the term 'shimmy' relating to a nosewheel.

QUESTION 8.

What is the function of an autoretract brake system.

QUESTION 9.

Describe the operating principle of the anti-skid system.

QUESTION 10.

List the properties of the ideal hydraulic fluid.

QUESTION 11.

Identify the types of hydraulic fluids in use in aircraft hydraulic systems.

QUESTION 12.

Explain the working principle of passive hydraulic system.

QUESTION 13.

Explain the working principle of an active hydraulic system.

QUESTION 14.

Describe the function of the Hydraulic System reservoir.

QUESTION 15.

Describe the function of the Hydraulic System accumulator.

QUESTION 16.

Describe the working principle of a pneumatic pump.

QUESTION 17.

Describe the working principle of a Hydraulic Pump Case Drain system and the Fluid Cooler.

QUESTION 18.

Describe the working principle of a hydraulic motor.

QUESTION 19.

Describe the function of a check valve.

QUESTION 20.

What is the function of a Priority Valve.

QUESTION 21.

What is the normal hydraulic system pressure of Jet Transport Aircraft.

QUESTION 22.

Describe the operation of a single acting jack.

QUESTION 23.

What indicating instruments are used to monitor the hydraulic system.

QUESTION 24.

Describe the operation of the anti-skid system.

QUESTION 25.

What is meant by tyre creep.

QUESTION 26.

Why are some tyres limited to a maximum ground speed.

QUESTION 27.

What is the function of the brake wear indicator.

QUESTION 28.

Identify a tyre tread and describe the advantages of various tread patterns.

QUESTION 29.

What is meant by a tyres 'ply rating'.

QUESTION 30.

List the stresses acting upon an aircraft in flight.

QUESTION 31.

In flight, the lower surface of a wing is subjected to stress called.

QUESTION 32.

In a monoque type fuselage construction, the strength and ability to withstand stress is in.

QUESTION 33.

In a cantilever wing construction, the bending stresses are borne by the ...... and the torsional stresses by the _________ ribs.

QUESTION 34.

List the different types of fuselage construction.

QUESTION 35.

What do you understand by damage-tolerant design.

QUESTION 36.

Define the maximum zero-fuel weight.

QUESTION 37.

What limits the airframe life.

QUESTION 38.

What are the two major stresses to which the fuselage in a pressurised aeroplane are subjected.

QUESTION 39.

How is stress concentration avoided in the cabin windows of a pressurised aeroplane.

QUESTION 40.

What type of construction is used in large jet transports.

ANSWERS

ANSWER 1.

To absorb the shock loads of landing and taxiing over uneven ground.

ANSWER 2.

A bogie unit consists of a multi – wheels unit design using more than one axle. This enables the aircraft weight to be spread over a greater surface area.

ANSWER 3.

A drag strut will absorb the drag (fore/aft) loads imposed on the undercarriage, especially on landing.

ANSWER 4.

To maintain wheels alignment with the longitudinal axis of the aircraft.

ANSWER 5.

Down locks prevent the gear from collapsing on the ground. An alternative to this would be the provision of a geometric lock associated with the sidestays.

ANSWER 6.

Pneumatic pressure can be used to lower the gear alternately or it can be allowed to free fall under gravity.

ANSWER 7.

Nosewheel shimmy is a situation which occurs when oscillation about the centreline of a naturally castering nosewheel becomes excessive.

ANSWER 8.

To reduce the stress on the main wheels spinning before retraction takes place, the wheels are automatically braked.

ANSWER 9.

The anti-skid system will provide maximum retardation of the aircraft without the wheels skidding for varying runway conditions.

ANSWER 10.

Low viscosity. Good lubrication. Non flammable. Non toxic. Low freezing point. Non foaming. Stable. Compatibility.

ANSWER 11.

Mineral based oil, designated DTD 585 (MIL-H5606) coloured red and generally used in light aircraft. Synthetic oil (phosphate ester based) coloured light purple and used in high performance aircraft.

ANSWER 12.

Fluid pressure is transmitted in an enclosed hydraulic fluid system by exerting a force on a small area piston and moving it over a set distance. This will produce a larger force on a piston having a larger area and will move it over a shorter distance.

ANSWER 13.

In order to operate such aircraft components as Landing Gear, Flaps, Flying controls etc. A high pressure energy source is required. This is provided by the system pressure pump and with suitable routing and control, the various components are actuated.

ANSWER 14.

The reservoir will: Store Hydraulic fluid. Make up for small fluid losses. Allow for thermal expansion. De-aerate the fluid. Allow for varying volumes of fluid displacement. Provide a reserve of fluid.

ANSWER 15.

The accumulator serves the following functions; Absorbs pressure surges. Provided supplemental system pressure on high fluid demand. Maintains system pressure when pump is idling (in association with ACOV). Stores Hydraulic pressure.

ANSWER 16.

An air driven motor, supplied from the aircraft pneumatic system, is coupled to a hydraulic pump via a drive shaft. The hydraulic pump ensures that system pressure is available when required.

ANSWER 17.

Hydraulic fluid is designed to flow through the internal working part of the pump and routed to return to the reservoir. It will pass through a heat exchanger, normally situated in the lower part of a fuel tank, which will cool the hydraulic fluid.

ANSWER 18.

A hydraulic motor is the reverse of a hydraulic pump. Hydraulic pressure applied to the pistons will cause the shoes, in contact with the variable angle swash plate, to convert the resulting force into rotary motion.

ANSWER 19.

A check valve (or restrictor valve) is designed to provide full hydraulic fluid flow in one direction and a restricted flow in the opposite direction.

ANSWER 20.

A priority Valve will maintain desired system pressure to essential hydraulic services by isolating supply to non essential services if system pressure falls below a pre-set value.

ANSWER 21.

Normal system pressure is 3000psi.

ANSWER 22.

The jack moves under hydraulic pressure in one direction only. Once hydraulic pressure is released the jack is returned by spring operation.

ANSWER 23.

Gauges are used to indicate system pressure, temperature and contents. In addition, warning lights can be used to alert the operator when abnormal conditions are present.

ANSWER 24.

Wheel rotation speed is measured by a transducer, passed to a computer and is compared to the barking demand and aircraft de-acceleration rate. As rotational speed diminishes, braking force is reduced to prevent wheel locking.

ANSWER 25.

Tyre creep is the rotational movement of the tyre in relation to the wheel. It can be identified by the change in position of corresponding marks painted on the wheel and tyre.

ANSWER 26.

It forms part of a tyres rating. It means that the tyre rotational speed could limit take-off performance.

ANSWER 27.

A brake wear indicator will indicate, by the amount of pin external protrusion, the amount of brake pad wear.

ANSWER 28.

The tread refers to the area forming the crown and shoulder of the tyre. A ribbed tread is most commonly used, which provides good traction, long tread wear and directional stability. Patterned (diamond) treads are more suitable for unpaved surfaces.

ANSWER 29.

Ply rating is an index of the strength of that tyre. It does not necessarily reflect the number of plies used in the construction.

ANSWER 30.

Bending. Tension. Compression. Torsion. Shear.

ANSWER 31.

Tension.

ANSWER 32.

The fuselage skin.

ANSWER 33.

Spar's, chordwise

ANSWER 34.

Steel tube (or truss). Monoque. Semi-monoque.

ANSWER 35.

It is assumed that production components will have minor flows and anticipates their growth, which are then monitored by precise inspection procedures.

ANSWER 36.

The maximum permissible weight of an aeroplane with no usable fuel.

ANSWER 37.

Fatigue due to the load cycles on T/O, landing and pressurisation.

ANSWER 38.

Bending and hoop stress

ANSWER 39.

Well-rounded edges

ANSWER 40.

Semi-monocoque

Air Conditioning & Pressurisation

Air Conditioning & Pressurisation

Compressed Air Sources and Uses

1. When discussing braking systems we saw how compressed air may be used as an emergency pressure supply in the event of hydraulic system failure. When discussing gas turbines, air starting systems are covered. We know that low pressure air is used to operate the gyroscopic instruments in some light aircraft.

2. In addition, air is supplied under pressure to the cabin air conditioning system. In the case of aircraft operating above 10,000 feet altitude, this air is also used to pressurise the cabin. In some aircraft compressed air is used to operate ice protection systems and a number of aircraft currently in service have air-operated leading and trailing edge flaps.

3. Some piston engined aircraft use high pressure air exclusively to operate services. Both pneumatic and hydraulic systems are completely enclosed systems. The most significant difference in the two systems being that hydraulic fluid is practically incompressible whereas air is highly compressible.

Uses

4. Extensive high-pressure pneumatic systems powered by engine driven compressors are generally fitted on the older types of piston engined aircraft and are used to operate services such as:

(a) Landing Gear

(b) Wing Flaps

(c) Wheel Brakes

(d) Radiator Shutters

(e) Aerofoil De-icing

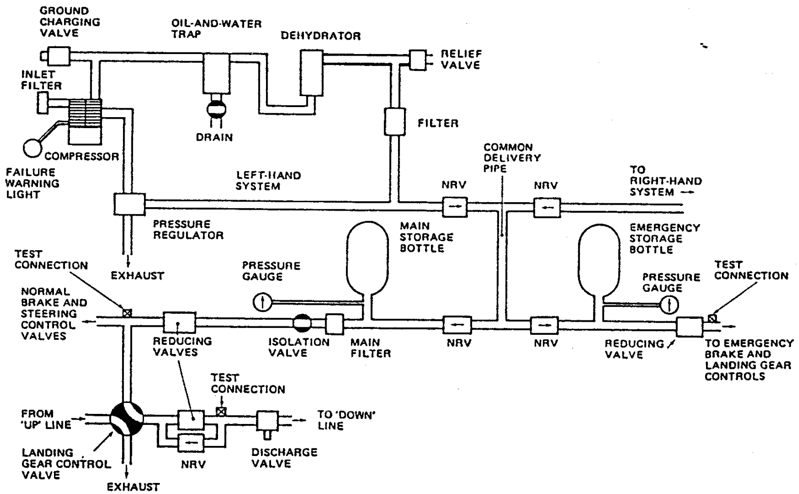
Power Sources

5. Engine driven compressors provide the required pneumatic system pressure. Working pressures range from 450 psi to 3500 psi (typically 3000 psi). Depending upon required working pressure, compressors will have several stages of compression. Intercoolers are used to cool the air between each stage of compression. Finned tubes may be used to dissipate heat in the compressor delivery line.

System Description and Components

6. A typical high pressure pneumatic system is illustrated at Figure 4-1.

FIGURE 4-1



The main components are:

(a) Engine driven compressor

(b) Pressure regulator

(c) Main storage bottle

(d) Pressure reducing valve

(e) Oil and water trap

(f) Indication and warning

Pressure Regulator

7. A pressure regulator valve controls the system pressure and provides a means of off-loading the compressor. A relief valve protects the system in the event of regulator valve failure.

Main Storage Bottle

8. The main storage bottle acts as a reservoir to store compressed air, giving a reserve of power for short bursts of heavy service operation, or emergency use in the event of engine or compressor failure.

Pressure Reducing Valve

9. For actuators requiring a pressure lower than the main system pressure (e.g. brakes) a pressure reducing valve is fitted.

Oil and Water Trap

10. The oil and water trap is fitted to remove oil and water from the compressed air passing from the compressor to the air storage bottle. The oil originates from compressor lubrication and the water is precipitated from the air in the compression process.

Indication and Warning

11. Pressure gauges, located in the cockpit, provide indication of pressure in the main and emergency storage bottles. Failure of a compressor is indicated by the illumination of a warning light.

Compressor Bleed Air Pneumatic Systems

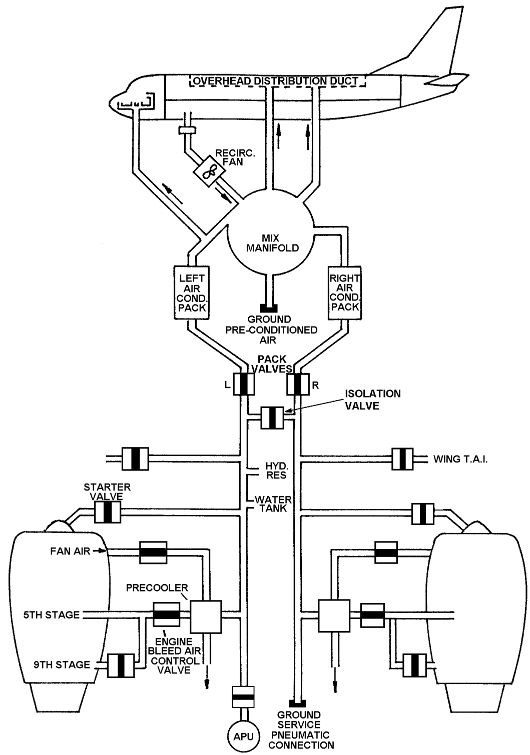
12. Gas turbine powered aircraft, especially those using high by-pass ratio engines, have a superfluity of compressed air produced by the gas turbine compressors. Air is bled from the later stages of the HP compressor, where the pressure is sufficiently high to satisfy the requirements of all the air-driven services. An example of such a system is shown at Figure 4-2.

13. Air is bled from the 5th stage of the engine HP compressor and supplied to the system at a controlled rate by the engine bleed air valve. This bleed can be supplemented from the HP compressor 9th stage when system demand exceeds 5th stage supply. The very high temperature of the bleed air is reduced to manageable proportions by passing it through a pre-cooler heat exchanger, cooled by air bled from the turbo-fan outlet (by-pass air).

14. When main engines are not operating, the bleed air system can be supplied from the auxiliary power unit (APU) compressor bleed. This source can also be used to supplement the pneumatic system in flight in the event of loss of engine bleed air for any reason.

15. The bleed air (pneumatic) system supplies air for operating the main engine air turbine starters, wing leading edge thermal anti-icing (TAI), pressurisation of the hydraulic system reservoirs and the domestic water tank, and for the two aircraft air conditioning packs. The system is divided into two halves, separated by an Isolation Valve. In flight this would normally be closed, with the left system supplied by the number one engine and the right by the number two engine. Either of the engines, or the APU, is capable of supplying the normal pneumatic requirements of the aircraft and one air conditioning pack is capable of meeting the demands of air conditioning and pressurisation. The air conditioning load is reduced by recirculating some cabin air back to the mix manifold to supplement incoming fresh air. A simple replaceable filter is fitted to extract any contaminants from the recirculation flow before it passes back to the cabin.

FIGURE 4-2
Air Driven
Services



16. Emergency pneumatic operation of landing gear and brakes is often provided for by air storage bottles. These may also be fitted for air starting the engines in the event of APU failure.

17. Pneumatically powered systems have the advantage of being lighter than hydraulic systems, since no return lines are required and the reservoir (the atmosphere) is virtually inexhaustible. For this reason some aircraft (the F27 is the only surviving example) have employed pneumatic powered services throughout. In others, those systems used only at certain stages of flight (flaps, landing gear and brakes) are pneumatically operated, primarily because of the weight saving advantage.

18. In construction and actuation, these systems are basically the same as hydraulic systems, with control effected through selector valves and actuation by rams, or jacks. The disadvantages of pneumatically operated systems are the non-lubricating properties of air and the dangers of its water content freezing at altitude. Also, high pressure air storage cylinders are a potential explosive hazard.

Air Conditioning

19. The requirements of an air conditioning system are discussed briefly below:

Provision of Fresh Air.	Fresh air must be provided at the rate of 1lb per person per minute in normal circumstances, or at not less than half this rate following a failure of any part of the duplicated air conditioning system.
Temperature.	Cabin air temperature should be maintained within the range +18°C to +24°C (65°F to 75°F).
Relative Humidity.	The relative humidity of the cabin air must be maintained at, or close to, 30% (in the atmosphere at 40,000 ft the relative humidity is only 1 to 2%).

Contamination.	Carbon monoxide contamination of the cabin air must not exceed one part in twenty thousand.
Ventilation.	Adequate ventilation must be provided on the ground and during unpressurised phases of flight.
Duplication.	The air conditioning system must be duplicated to the extent that no single component failure will cause the provision of fresh air to fall to a rate which is lower than 0.5lb per person per minute.

20. An aircraft air conditioning system must therefore be capable of maintaining an adequate supply of air for ventilation and pressurisation at a comfortable temperature and relative humidity. These requirements are met as follows:

Adequate Supply.	The mass flow of air into the aircraft cabin is maintained at a constant value, sufficient to achieve cabin pressurisation when cruising at maximum operating altitude.
Temperature.	The temperature of the air supply to the cabin is controlled by mixing hot and cold air in variable proportions to maintain the cabin air temperature within prescribed limits.
Humidity.	Atomised water is added to the cabin air supply at a controlled rate, to maintain a comfortable level of humidity.

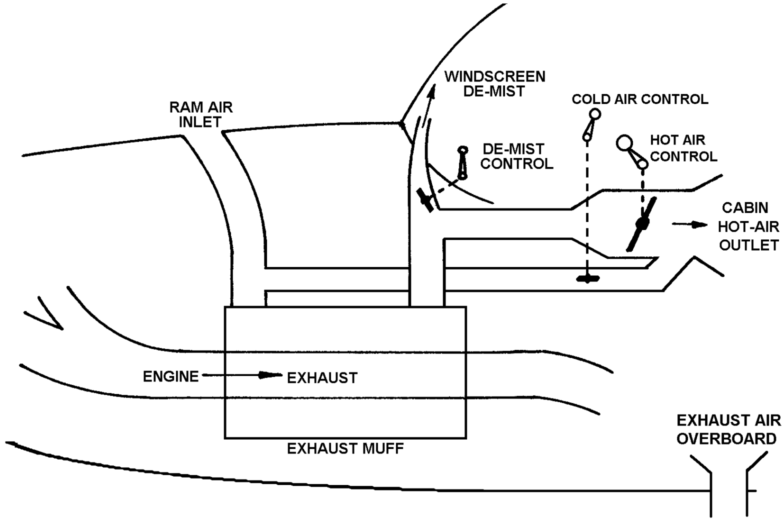
21. Depending upon the type of power unit fitted, and the operating characteristics of the aircraft, the source of cabin air and the method of conditioning will vary.

Ram Air Systems

22. In light, unpressurised, piston-engined aircraft ambient atmospheric air is admitted to the cabin by way of intakes facing into the airflow. These are known as ram air systems. Some of the ram air can be heated and added to the cold ambient air in variable proportions to achieve a comfortable cabin temperature.

23. Heating is achieved either by passing some of the ram air through a muffler surrounding the engine exhaust system, or by passing it through a duct surrounding the combustion chamber of a fuel-burning heater. The operating principles of the exhaust muff and combustion heater systems are shown in Figure 4-3 and Figure 4-4.

FIGURE 4-3
Muffler Heating



24. The combustion heater works by burning a fuel/air mixture within the combustion chamber. Air for combustion is supplied by a blower or fan and fuel is supplied from the normal fuel system via a solenoid operated fuel valve. Control of the fuel valve solenoid may be effected by duct temperature sensors, or by manual override.

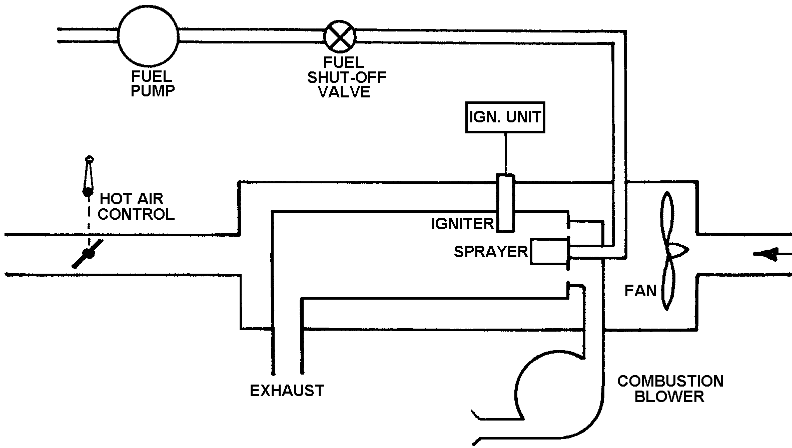
25. The combustion chamber must be so designed that essentially no possibility exists that any leak can occur from within the chamber into the air flowing around the chamber. The system must also be strong enough to completely contain any explosion within the combustion chamber. Safety devices must include:

(a) An automatic shut-off controller, to operate if the air outlet temperature becomes too high.

(b) Automatic fuel cut-off in the event of any malfunction.

(c) Totally adequate fire protection in the event of failure of the structural integrity of the combustion chamber.

FIGURE 4-4
Combustion Heating



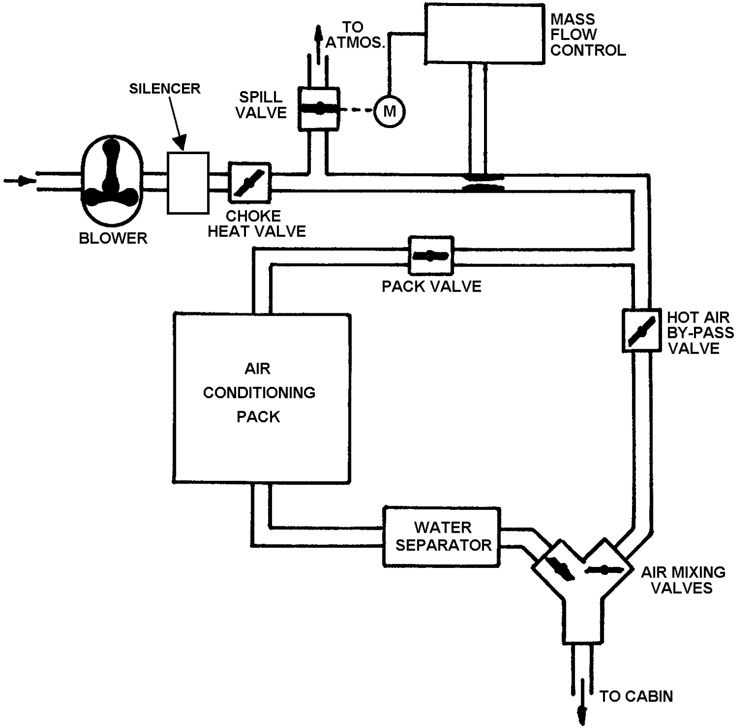
Engine-Driven Blower Systems

26. For pressurised aircraft, where a supply of bleed air from a gas turbine compressor is not available, cabin air supply is provided by blowers driven through the engine accessory gearbox. Such a system was necessary in piston engined airliners and is still used in some turbo-prop aircraft. These blowers are either of the positive displacement (Roots) type, or of the centrifugal type.

27. The blower must be capable of supplying the mass air flow necessary in order to maintain cabin pressurisation at maximum altitude with the engine at cruise rpm. Consequently, at sea level with the engine at full power the blower will be delivering a far higher mass air flow than is required. In order to prevent over-pressurisation of the air ducting the excess mass air flow is dumped overboard by spill valves, controlled by a mass flow controller. This system is shown schematically at Figure 4-5.

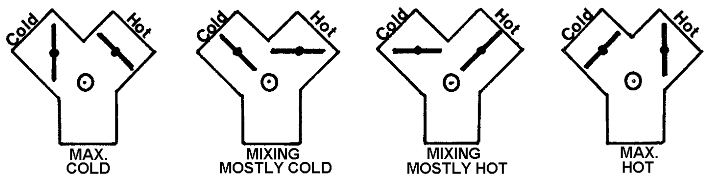
28. In such a system, the mass flow produced by the engine-driven blower is dependent upon blower rotary speed, which is limited only by the drive gear ratio. If this excess mass flow is restricted the resulting back pressure will increase the temperature of the air discharged by the blower to a level sufficient for cabin heating. The temperature of the blower discharge air is sensed and used to control the position of the choke heat valve. As temperature increases the valve will be progressively opened to prevent excess pressure/temperature. A reduction of temperature will cause the valve to partially close, increasing pressure and temperature. The method of control often used for this is a wheatstone bridge circuit.

FIGURE 4-5
Mass Flow Controller



29. Some of the hot air from the blower is cooled in the air conditioning pack. This chilled air is mixed with hot air, which has by-passed the pack, to provide air supply to the aircraft cabin at a comfortable temperature. The mixing unit consists of two butterfly valves, the setting of which is controlled by a wheatstone bridge circuit. The cabin supply air temperature is sensed and the bridge circuit used to position the 'hot' and 'cold' butterfly valves to increase or decrease the quantities of hot or cold air supplied to the cabin. Such a temperature control mixing unit is shown at Figure 4-5. The two valves are geared to a single spindle which rotates through 180° to drive the valves from the 'maximum cold' position to the 'maximum hot' position. This principle is illustrated diagrammatically at Figure 4-6.

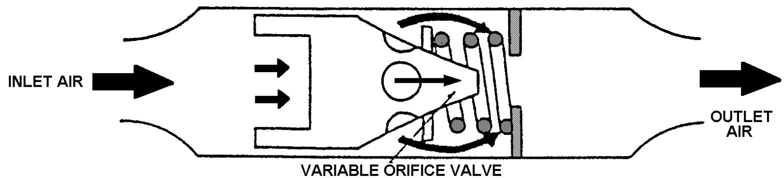
FIGURE 4-6
Butterfly Heat/ Cold Control Valves



Engine Bleed Air Systems

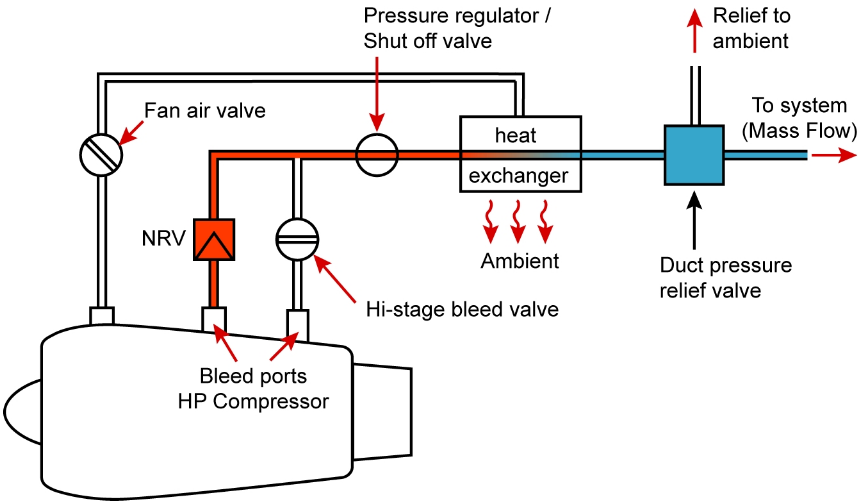
30. This system is widely used on modern jet-engined aircraft. Air for the aircraft pneumatic system is generally bled from an intermediate stage (IP) and high pressure stage (HP) of the engine-high pressure (HP) compressor. In order to supplement the low pressure of air supplied at reduced thrust settings (for example during the descent), air is automatically bled from the high pressure stage (HP) of the compressor. This ensures that adequate pressure is available at all thrust settings.

FIGURE 4-7
Flow Control Valve



31. The pressure and flow rate of the bleed air is controlled by an electro-pneumatic pressure regulating valve or flow control valve. This valve additionally incorporates a shut-off facility which is used in the event of a system malfunction. The flow control valve illustrated at Figure 4-7 is in effect a variable orifice device, the orifice being set by the pressure of the air flow passing through the valve. A non return valve (NRV) is used in order to prevent a reverse flow of air back into the engine compressor. Overpressure of the pneumatic system is prevented by a duct pressure relief valve which is located downstream of the pressure regulating valve. A simple system is illustrated at Figure 4-8.

FIGURE 4-8
Pneumatic System

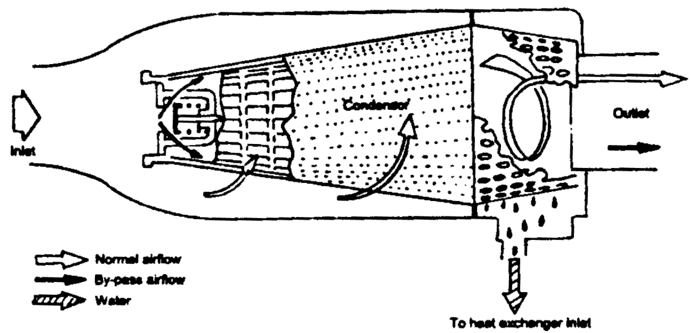


32. The pneumatic system is monitored by reference to duct pressure gauges located on the flight deck and warning lights indicating a sensed duct leak. A small leak from a pneumatic duct could result in the activation of a thermal sensing device (fitted adjacent to duct) without a noticeable loss of duct pressure. A severe duct leak (rupture or broken assembly) will be indicated by a loss of pressure on the associated duct pressure gauge. Thermal sensors are fitted adjacent to the ducting throughout its length. The detectors can be either a spot type or a continuous detection sensing element system.

33. Some systems incorporate an air cleaner in the bleed air system to filter the air used for air conditioning before it enters the pneumatic system.

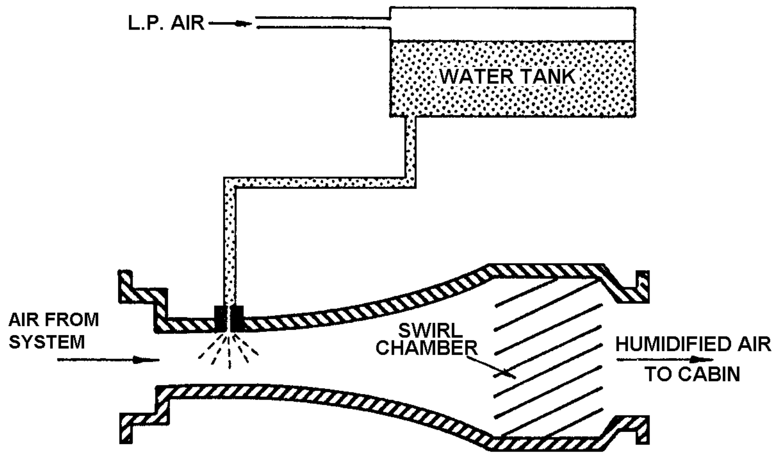
Humidity Control

FIGURE 4-9
Water Separator



34. The humidity control system consists of two components, a water extractor and a humidifier. The water extractor is used to reduce the moisture content of humid air, in order to prevent precipitation in the ducting and in the cabin itself. The unit is situated downstream of the cooling unit, since the cooled air is more likely to have condensed water droplets in it than the warm air upstream of the cooling unit. Protection devices are fitted to water extractors to prevent blockage due to formation of ice. They may take the form of a bypass valve or routing a warm air bleed to the extractor should ice begin to build up. A water separator is illustrated at Figure 4-9.

FIGURE 4-10
Humidifier Unit



35. The humidifier is used at high altitude, where the air is very dry and consequently the relative humidity of the air introduced to the cabin, without humidification, would be less than the required minima, resulting in crew/passenger discomfort. The aircraft's normal domestic water supply is used, the water being atomised by air which is bled from the main air conditioning supply duct. A pressure switch, which is located downstream of the humidifier itself, stops the supply of water if the duct air pressure drops below a preset level, since this would prevent efficient atomization and precipitation would occur in the cabin. A schematic diagram of a humidifier unit is shown at Figure 4-10. The humidifier is automatically prevented from operating at lower altitudes by an altitude switch.

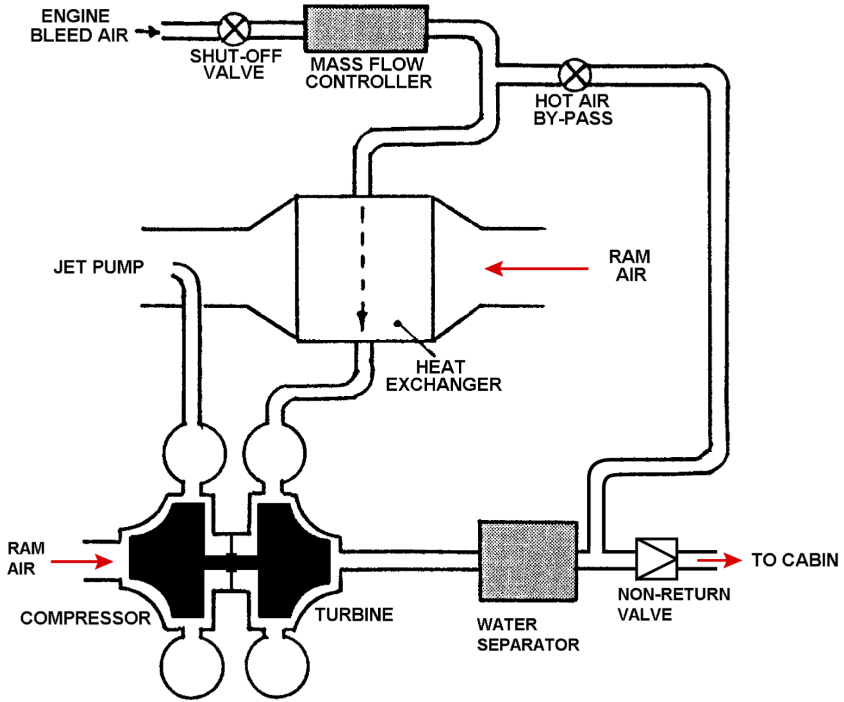
Air Cooling Systems

36. The air which is supplied to the cabin and flight deck will invariably need to be cooled, since the source of supply is hot or, in the case of engine bleed air, very hot. The most obvious source of cooling is ambient air, used in a heat exchanger to extract heat from the cabin fresh air supply. However, largely because of the ram air heating effect at high subsonic speeds, ambient air alone cannot reduce the cabin supply air temperature to an acceptable level. Furthermore, even in lower speed aircraft the size of such a heat exchanger would impose a significant weight and drag penalty. Some method of refrigeration is necessary in order to reduce the temperature of the fresh air supply to a level where, by mixing chilled and hot air, a comfortable cabin temperature can be maintained. Most modern passenger aircraft use an air cycle system of refrigeration.

Air Cycle Cooling

37. If the pressure of a volume of gas is reduced, its temperature must also be reduced. An efficient means of reducing the pressure of the cabin air supply is to make the air drive a turbine. The expansion of the air as it passes through the turbine causes its temperature to drop. The more the work done by the turbine, the more the air must expand and the greater the temperature drop. This principle is used in all air cycle refrigeration systems, all that varies is the means by which the turbine is loaded (made to do work). Typical temperature drop across the turbine is in excess of 100°C.

FIGURE 4-11
Brake Turbine

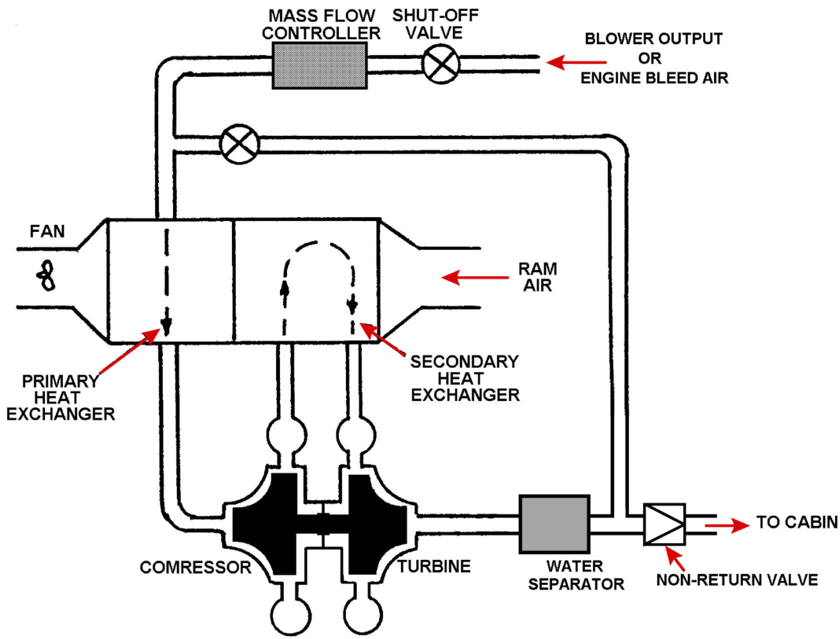


38. A brake turbine is illustrated at Figure 4-11 and is the simplest air cycle cooling unit in common use. Commonly referred to as an air cycle machine (ACM).

39. In order to drive an air turbine the air must be at high pressure, so the brake turbine air cycle machine uses bleed air from the high pressure end of the engine compressors. Engine bleed air is passed through a ram air-cooled heat exchanger, which extracts some of its heat without significantly reducing its pressure. The high pressure air is then used to drive a turbine, which in turn drives a compressor. The compressor discharge is ejected through a nozzle in the heat exchanger outlet and this nozzle induces the ambient air flow through the heat exchanger.

40. The unit has the benefit of simplicity and light weight, typically less than 12 lb (5 kg). An alternative form of brake unit uses a fan, instead of a compressor, as a brake load on the turbine. The fan is used to pump ambient air through the heat exchanger. The high supply pressure necessary to achieve a large pressure/temperature ratio across the turbine means that brake turbine cold air units are only suitable for use in multi-spool turbo-jet or turbo-fan powered aircraft, where HP bleed air pressure is high.

FIGURE 4-12
Bootstrap Cooling System

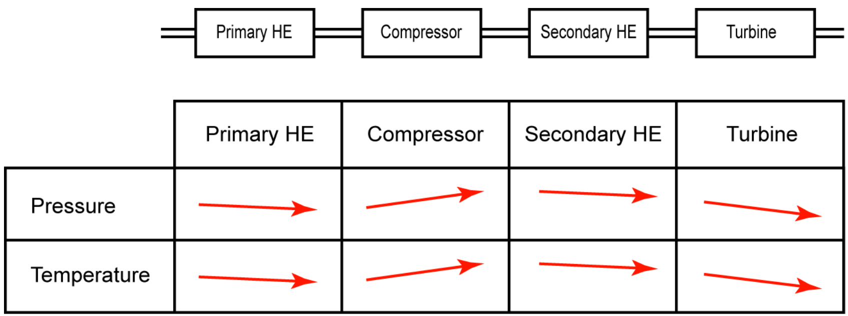


41. A bootstrap cooling system is illustrated at Figure 4-12. Where the air supply pressure is not high enough to satisfactorily operate a brake turbine a bootstrap system can be used. The name bootstrap derives from the way in which the air apparently lifts its pressure by its own bootstraps.

(a) The low pressure bleed air (or blower output) is pressure-boosted by the turbine-driven compressor in order to produce an adequate pressure/temperature drop across the turbine. A secondary heat exchanger, or intercooler, is necessary to remove the unwanted temperature rise in the compressor. An electrically-driven fan provides the cooling air flow through the heat exchanger when ram air is not present. (Aircraft parked).

42. Figure 4-13 shows the relationship between temperature and pressure as air flows through the bootstrap system.

FIGURE 4-13
Bootstrap Pressure & Temperature Relationship



43. The bootstrap cold air unit is heavier and more complex than the brake turbine, but it requires less power to operate. It is favoured in aircraft where high pressure bleed air is not available or is undesirable, for example in small turbo-prop and aircraft utilising high by-pass ratio engines.

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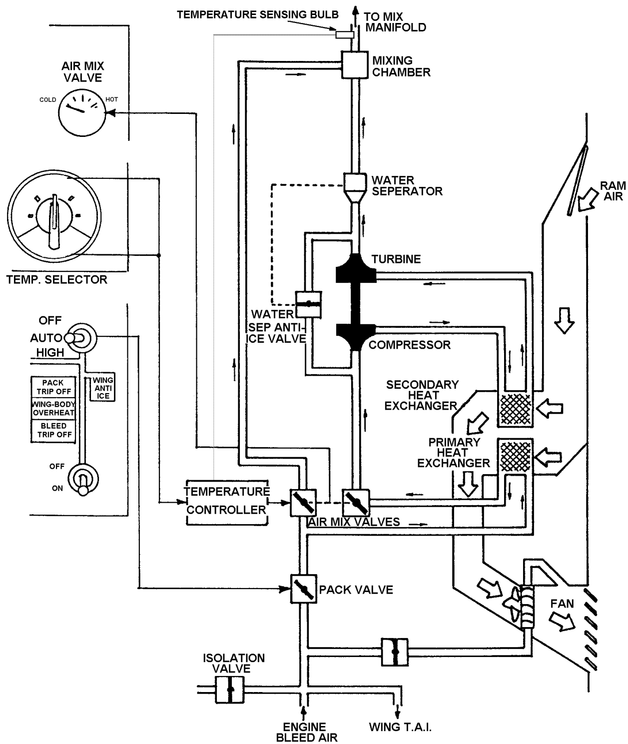
44. The air conditioning system shown schematically at Figure 4-14 is of the type fitted to the Boeing 737.

FIGURE 4-14
Air Conditioning System



45. The aircraft has two air cycle air conditioning packs which are completely independent of each other and either of which is capable of supplying sufficient conditioned air to maintain a cabin altitude of 8000 feet or less when the aircraft is operated at its maximum certified ceiling. Normally the left pack uses bleed air from the number one engine and the right pack from the number two engine. The output of the two packs is combined in the mix manifold. The right system only is illustrated at Figure 4-14.

46. Flow of air to the air conditioning pack is controlled by its pack valve. The flow of air through the pack, or by-passing the pack, is controlled by the two air mix valves. In the automatic setting these are positioned by the temperature controller to achieve the requisite mix of hot and cold air in the mixing chamber. In the manual mode the valves are set according to the position of the temperature selector control. When the pack valve is closed the air mix valves are driven to the full cold position (by-pass closed, air cycle inlet open) to prevent initial cabin overheat on pack start-up.

47. The heat exchangers are cooled by ram air in flight, the ram air flow is automatically controlled by adjustment of the ram air inlet door and exhaust louvres, according to the sensed air cycle machine compressor discharge temperature. On the ground, or during slow flight (flaps extended) the inlet door is full open and the bleed air driven turbo-fan (pack cooling fan) operates to augment the ram air flow. The pack cooling fan (turbo fan) is controlled by the turbo fan air valve which passes pneumatic air to the turbo fan at varying flows depending on aircraft position and configuration i.e. ground/airborne or slow/high speed flight.

48. A water separator is located downstream of the air cycle machine turbine, to remove water which has condensed during the cooling process. A temperature sensor in the separator controls an anti-ice by-pass valve to prevent icing in the water separator. Various temperature sensors are located throughout the air conditioning system. They function to:

(a) sense cabin temperature and provide a signal for control and indication of cabin temperature on the flight deck.

(b) sense duct temperature and provide a warning to the flight deck. Some systems include shutdown of system on overheat detection automatically.

(c) warn of hot air leaks from the ducting.

(d) sense blocking of water separator and provide a warm air bypass to remove any formation of ice.

(e) sense compressor outlet temperature for indication or in some cases auto shutdown of system following overheat.

49. On some large jet transport aircraft it is necessary to supplement the air temperature in individual zones to satisfy the comfort of the occupants of that zone. Individual zone temperature requirements are satisfied by adding hot trim air to the output of the packs (ACM). However, trim air is not added to the zone controlling the output of the packs (i.e. demand for most cooling).

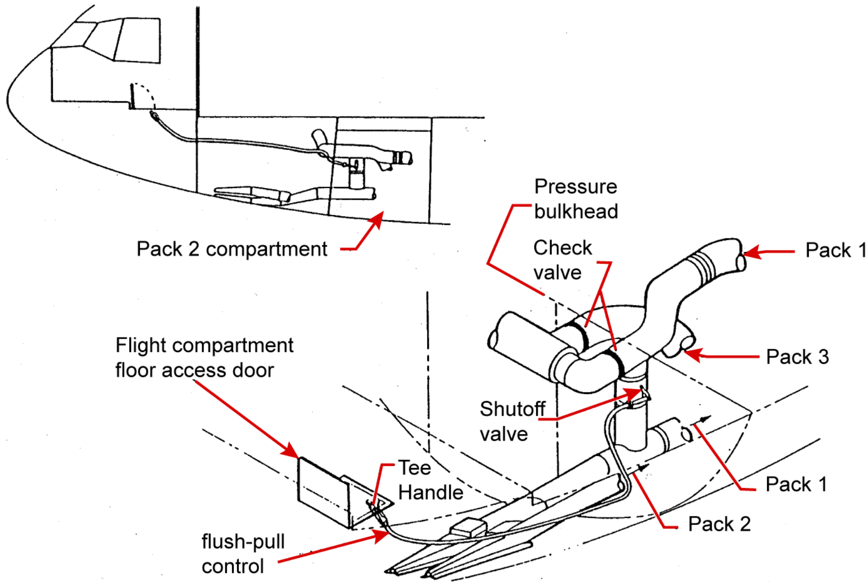
50. On all pressurised aircraft, provision must be made to ventilate the aircraft in flight in the unlikely event of total pneumatic supply or air conditioning systems failure.

51. A means is provided to admit ram air into the aircraft distributions system. There is no temperature control of this ram air, but a means of varying the flow may be provided.

52. The cabin pressure outflow valve is opened to ensure a ventilation flow.

53. Prior to operation of the unpressurised ventilation system, the aircraft must descend to below 10,000ft, terrain clearance permitting.

FIGURE 4-15
Ram Air
Ventilation



Vapour Cycle Cooling

54. In some large passenger transport aircraft, vapour cycle (refrigerator) air cooling systems are installed to supplement the air cycle systems whilst the aircraft is on the ground. Vapour cycle refrigeration systems make use of the fact that, during evaporation, a liquid absorbs latent heat from its surroundings. By using a liquid with a low boiling point, the latent heat removed from the cabin fresh air supply results in the temperature of the cabin air supply being chilled. The refrigerant typically used in vapour cycle systems is freon, with a boiling point of about 3°C at ISA msl pressure. A vapour cycle cold air unit is shown schematically at Figure 4-16.

55. Refrigerant at low pressure is drawn through the tubes of the evaporator (a heat exchanger) by the air turbine-driven compressor. The refrigerant enters the evaporator as a liquid, however the heat of the cabin air supply passing over the tubes of the evaporator causes the refrigerant to boil and the latent heat absorbed during this process of evaporation cools the cabin air supply.

56. The compressor raises the pressure, and therefore the boiling point, of the refrigerant before it enters the second heat exchanger in the cycle, the condenser. Ram air passing over the tubes of the condenser cools the refrigerant, which condenses back into a liquid, giving up latent heat to the ambient ram air as it does so.

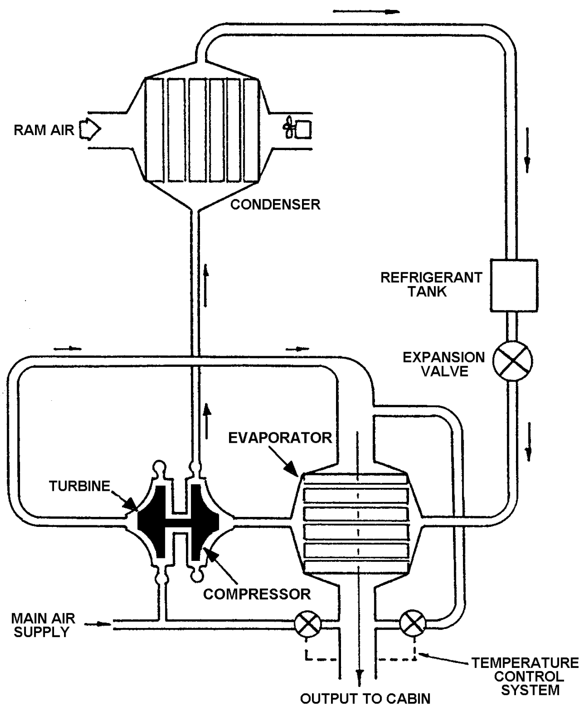
57. The pressurised liquid passes through an expansion valve which is an integral part of the evaporator situated at the inlet side. The valve causes the refrigerant to start evaporating the instant it enters the evaporator and to be completely evaporated before it leaves the coil. A thermal element is attached to the suction side of the evaporator and any change in temperature at the suction line causes a corresponding change in the thermal element. The temperature change in the thermal element is signalled to the expansion valve which then controls the refrigerant entering the evaporator according to temperature demand. The refrigerant tank acts as a reservoir.

58. Turbine outlet and inlet air is mixed with the chilled air output from the evaporator by means of a thermostatically operated series of valves. Air then enters the cabin at a controlled temperature of about 20°C.

59. In many aircraft systems the compressor is electrically-driven, the evaporator being capable of chilling the cabin air supply without the need of a pre-cooling turbine.

60. A major advantage of vapour cycle cooling is that it can be used on the ground without the need to run the engines. Units are lighter than air cycle units and do not require a high pressure air supply.

FIGURE 4-16
Vapour Cycle
Cold Air Unit



Pressurisation

61. In order to stay alive the human body requires oxygen. Insufficient oxygen, or hypoxia, results in loss of consciousness and, eventually, death. With increase in altitude both air pressure and air density decrease and this has a twofold effect:

(a) With decreased air density the proportions of the gases in the air will remain constant, but will quite simply be reduced in quantity. The amount of oxygen inhaled will therefore reduce.

(b) The body absorbs oxygen through the lung tissues during normal respiration, and this process is assisted by the pressure of air within the lungs. The lower the pressure, the more difficult it becomes for the lungs to perform efficiently.

62. Up to an altitude of 10,000 feet (3.3 km) there is sufficient air pressure and oxygen to enable the human body to perform efficiently. Above this level some artificial method must be employed to produce an environment inside the aircraft which equates to the conditions of air pressure and density at or below 10,000 feet. Cabin pressurisation normally achieves conditions equivalent to those of about 8,000 feet (2.6 km) or less. Under these circumstances the aircraft is said to be operating at a cabin altitude of 8,000 feet or less.

JAR-OPS 23.841 Pressurised Cabins

(a) If certification for operation over 25,000ft is requested, the aeroplane must be able to maintain a cabin pressure altitude of not more than 15,000ft in event of any probable failure or malfunction in the pressurisation system.

(b) Pressurised cabins must have at least the following valves, controls and indicators, for controlling cabin pressure.

(1) Two pressure relief valves to automatically limit the positive pressure differential to a predetermined value at the maximum rate of flow delivered by the pressure source. The combined capacity of the relief valves must be large enough so that the failure of any one valve would not cause an appreciable rise in the pressure differential. The pressure differential is positive when the internal pressure is greater than the external.

(2) Two reserve pressure differential relief valves (or their equivalent) to automatically prevent a negative pressure differential that would damage the structure. However, one valve is enough if it is of a design that reasonably precludes its malfunctioning.

(3) A means by which the pressure differential can be rapidly equalised.

(4) An automatic or manual regulator for controlling the intake or exhaust airflow, or both, for maintaining the required internal pressure and airflow rates.

(5) Instruments to indicate to the pilot the pressure differential, the cabin pressure altitude and the rate of change of cabin pressure altitude.

(6) Warning indication at the pilot station to indicate when the safe or pre-set pressure differential is exceeded and when a cabin pressure altitude of 10,000ft is exceeded.

(7) A warning placard for the pilot if the structure is not designed for pressure differentials up to the maximum relief valve setting in combination with landing loads.

(8) A means to stop rotation of the compressor or to divert airflow from the cabin if continued rotation of an engine-driven cabin compressor or continued flow of any compressor bleed air will create a hazard if a malfunction occurs.

63. The main problem with pressurised aircraft is not the complexity of the pressurisation system, but the fact that the airframe must be strong enough to withstand the considerable pressure differential. Pressure differential is the pressure difference between the inside and the outside of the pressurised hull of the aircraft. It imposes large stresses (known as hoop stress) which the hull must be capable of withstanding.

64. The flight deck, passenger areas and main cargo holds are pressurised, whilst areas such as the wheel wells and the tail cone are normally unpressurised. This reduces the number of potential leakages of cabin pressure, but introduces stress concentration points where, for example, the fuselage meets the fore and aft pressure bulkheads. These points undergo significant stress each time the aircraft is pressurised and de-pressurised (a pressurisation cycle). Just as continually bending and straightening a piece of wire will cause it to fatigue and break, so the repetitive stresses of pressurisation cycles will eventually lead to fatigue failure. Fortunately, those clever engineers can calculate the fatigue life of an aircraft structure and ensure replacement takes place in good time. The number of pressurisation cycles is a major element in assessing the fatigue life of an aircraft.

65. The rate of change of pressure from the surface upwards, through the first few thousand feet, is far greater than for the same number of thousands of feet at altitude. For this reason, a cabin pressure altitude of 8,000 feet or less is maintained, rather than mean sea level pressure. By this means the pressure differentials the hull must withstand are kept within the bounds of engineering feasibility. Table 1 illustrates this point.

Altitude	Air Pressure
Mean Sea Level	1013 mb (14.7 lbs/in^2)
8,000 feet	758 mb (11.0 lbs/in^2)
40,000 feet	186 mb (2.7 lbs/in^2)

66. If the cabin were to be maintained at mean sea level pressure whilst the aircraft was flying at 40,000 feet, the pressure differential across the skin of the hull would be 12 lbs/in^2. By maintaining a cabin altitude of 8,000 feet the pressure differential is reduced to 8.3 lbs/in^2.

67. Modern aircraft pressure hulls are normally designed to accommodate a maximum operating pressure differential of 8.6 lbs/in^2 to 8.9 lbs/in^2 .

68. Cabin pressurisation is not controlled, as one might expect, by governing the rate at which air is supplied from the air conditioning units, but by governing the rate at which cabin air is released to atmosphere. This is achieved by means of a pressure controller, which passes a signal to discharge valves. These valves restrict the rate at which cabin air is released to maintain the required cabin air pressure (cabin altitude).

69. As well as maintaining a constant cabin altitude in level flight, the pressure controller/discharge valve arrangement is designed to ensure a steady rate of change of cabin altitude as the aircraft climbs or descends. A rate of change of between 300 feet per minute and 500 feet per minute is normally considered to be comfortable. The rate of change is determined by the controller setting selected by the flight deck crew and by the rate of climb/descent of the aircraft. Maximum permissible rates of change of cabin altitude are discussed later.

70. In addition to the pressure controller and the discharge valves, pressure limiting safety valves and inward relief valves are also fitted to safeguard the structural integrity of the hull in the event of failure of the basic components.

71. Pressure controllers vary in construction and operation, but they basically consist of pressure sensing capsules and/or diaphragms (which are subjected to both cabin and external pressures) and metering valves and controls for setting the required cabin altitude and its rate of change. As the cabin pressure changes, the pressure controller automatically senses the change relative to the external pressure and transmits a signal to the discharge valves. This signal adjusts the discharge valve opening until the release of air from the cabin is just sufficient to achieve the desired pressure differential.

72. On some pressure controllers a ditching control is fitted. When activated, this will signal all discharge valves to fully close in order to minimise inflow of water. At the same time, all compressor output will be dumped, otherwise operation of the emergency exits would be impeded if the cabin remained pressurised. Appreciate that the force required to open a plug type door measuring 6 feet by 3 feet, with a positive differential pressure of just 1 lb/in^2, would be 1.16 tons.

73. Some pressurisation systems incorporate a manual/automatic dumping facility linked to the safety valve. This is to ensure that, whilst the aircraft is on the ground, the safety (dump) valve will be held in the open position preventing the pressurisation of the aircraft. Alternatively, the pressure dumping facility on some systems is linked to the discharge/outflow valves. In both cases manual switching is available to the pilot or automatic operation is via the aircraft weight (squat) switch.

74. Placards are displayed on flight deck pressurisation control panels, warning operators not to exceed nominated pressure differentials on take-off and landing. The restricted negative differential pressures typically range from .1 psi to .5 psi depending on aircraft type. This is to ensure that undue stresses are not imposed on the airframe structure which could occur during take-off and landing with excess negative differential pressure.

Safety Devices

75. The requirement for various safety devices is outlined in the following list:

(a) The aircraft must be fitted with devices in duplicate which will, at the maximum rate of flow, automatically limit the positive pressure differential to a given value. (If the aircraft is climbing at its maximum rate of climb it is permitted to exceed this maximum pressure differential value by 0.25 lb/in^2).

(b) The capacity of each half of the duplicated system shall be such that failure of one half would not cause appreciable fall in the pressure differential.

(c) Devices must be fitted in duplicate to prevent the negative differential pressure from exceeding 0.5 lb/in^2.

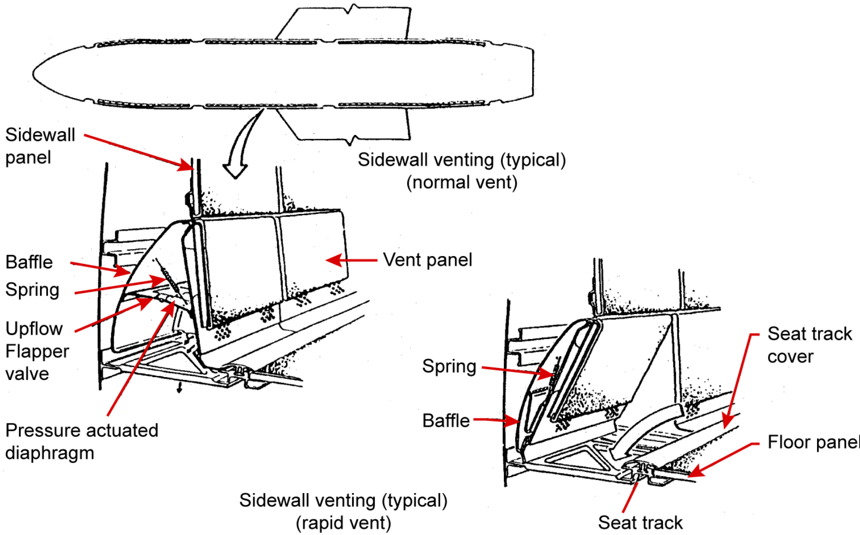
(d) A manually operated device must be fitted which permits the crew to reduce the pressure differential to zero.

(e) The system must have a suitable manual or automatic regulator (the pressure controller) for controlling both cabin pressure and rate of change of cabin pressure.

(f) Suitable instruments must be provided. If the system controls are not situated at the pilot station, an instrument indicating cabin altitude must be fitted at the pilot station. The crew member in charge of the cabin air supply shall have an indication of cabin altitude and differential pressure. The value of maximum differential pressure shall be indicated on or near the instrument.

76. Wide bodied jet transport aircraft are fitted with a means of equalising the pressure differential between sections inside the pressure hull which would result from a rapid loss of pressurisation. A rapid loss of pressurisation in one area could cause collapsing of floors, panels, walls etc. separating other areas in the pressure hull. A typical example would be the cabin floor and cargo department (see Figure 4-17).

FIGURE 4-17
Pressure Balance Panels



77. Of the items on the above list perhaps one is worthy of further discussion, namely the concept of negative differential pressure. The pressure hull is strengthened to withstand the forces imposed by relatively high pressure acting outwards on the pressure hull (positive differential pressure). If, during a descent, the cabin altitude stuck at 8,000 feet whilst the aircraft descended to mean sea level, the pressure outside the aircraft would be greater than the pressure inside (negative differential pressure). Hence the pressure loading would be the reverse of that for which the pressure hull is designed. In this condition, on the ground, the operation of inward opening doors would be extremely hazardous and the operation of outward opening doors impossible.

78. To prevent a condition of negative differential pressure from occurring, inward relief valves are fitted to the pressure hull. These valves open automatically at a differential pressure of, typically, 0.5 lb/in^2, whenever the outside air pressure exceeds cabin air pressure.

79. The action of the pressure limiting safety valves is to ensure that the absolute maximum pressure differential is never exceeded. This safety valve is usually combined with the inward relief valve and is spring-loaded to open at a differential pressure slightly higher then the controlled maximum differential pressure. In normal circumstances it is lightly spring-loaded in the shut position. In many aircraft the safety valve may be selected OPEN on the ground to ensure pressures are equalised. It is selected CLOSED prior to take-off.

Pressurisation System Description

80. Figure 4-18 shows the schematic arrangement of the pressurisation control system of a modern passenger transport aircraft. The aircraft is pressurised by a continuous inflow of conditioned air through the overhead distribution ducts, and cabin pressure is controlled by regulation of the outflow of air. The normal procedure is for a proportional relationship between ambient and cabin pressure to be maintained in the climb and descent, and a constant differential pressure to be maintained in the cruise. Two pressure-relieving safety valves (one only shown) and an inward-relieving negative relief valve are fitted.

81. Pressure control is achieved with the electric motor-operated main outflow valve. During normal operations an AC motor drives the valve, in emergency situations a DC motor takes over. The cabin pressurisation operates in one of three modes of control:

AUTO	The normal mode of operation.
STANDBY	A semi-automatic system used in the event of AUTO failure.
MANUAL	Manual control of cabin pressure by remote operation of the outflow valve.

82. Aircraft altitude is sensed barometrically direct from the static vents, with barometric connections fed from the Captain's altimeter (AUTO) or First Officer's altimeter (STBY). In standby mode an electrical altitude signal from the Air Data Computer (ADC) is also used, in the system illustrated. Cabin pressure is also fed to the pressure controllers, together with signals from the undercarriage squat switch and the flight/ground (FLT/GRD) switch.

AUTO Mode Operation The AUTO pressurisation control panel is ‘set up’ by inserting the intended flight or cruise altitude (FLT ALT) and the landing airport altitude (LAND ALT). Whilst on the ground the departure airport altitude (cabin altitude) is fed in continuously.

83. Whilst the aircraft weight is on the main wheels the undercarriage squat switch transmits a signal to the AUTO pressurisation controller. This signal prevents the controller from initiating cabin climb. The FLT/GRD switch, when selected to GRD, prevents any cabin pressurisation by holding the main outflow valve full open.

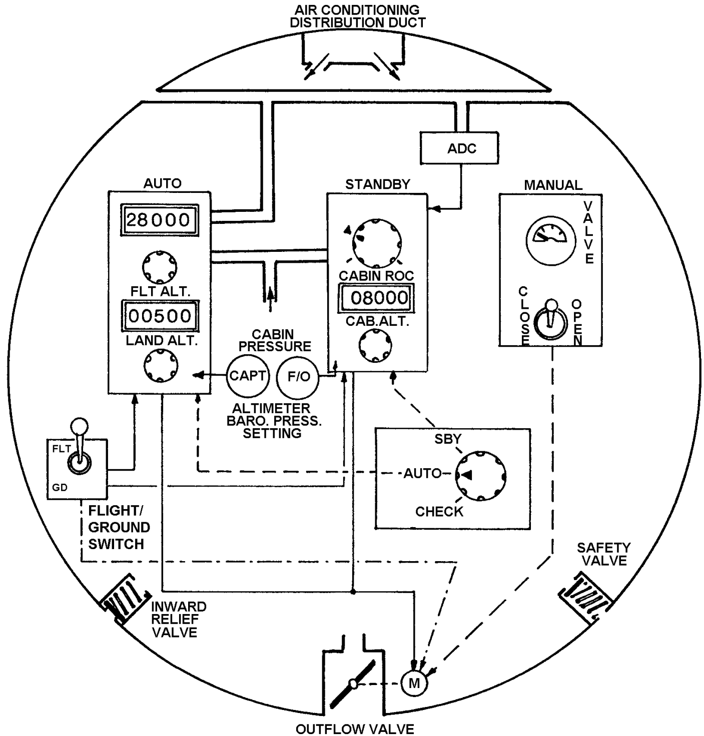


FIGURE 4-18
Pressurisation Control System

84. When taxiing commences the FLT/GRD switch is set to FLT and the controller moves the main outflow valve towards close, pressurising the cabin to 0.1 psi positive differential pressure (equivalent to airport altitude -200 feet). This ensures that the transition to pressurised flight will be gradual. Once airborne (signalled by the squat switch) the pressurisation controller switches to proportional control. Cabin altitude is increased at a rate proportional to aircraft rate of climb, but not greater than the maximum cabin rate of climb selected. Should the aircraft be required to hold during the climb, the cabin altitude will hold also, since cabin rate of climb is proportional to aircraft rate of climb.

85. When the outside air pressure is 0.25 psi above that of the selected flight altitude (FLT ALT) a cruise relay is tripped and the pressurisation controller switches from proportional to isobaric control. The controller now maintains a constant cabin altitude by maintaining a constant pressure differential (7.45 psi up to 28,000 feet, 7.8 psi above 28,000 feet) between cabin and ambient pressures.

86. If the aircraft begins to descend before the cruise relay has tripped (whilst still in proportional control), the pressurisation controller reduces cabin altitude, at a rate proportional to aircraft rate of descent, to land at the departure airfield elevation. If, during the climb, the pre-set value of FLT ALT is changed, this automatic reduction to departure field elevation facility is lost.

87. During isobaric (cruise) control, if aircraft altitude increases the controller will allow the pressure differential to reach a maximum of 7.9 psi in order to maintain constant cabin altitude. If the aircraft climbs further this maximum differential will be maintained and the cabin will climb maintaining maximum differential. This feature is adequate for minor excursions from cruise altitude, but for a higher cruise altitude the FLT ALT selection must be reset. If the differential pressure between FLT ALT and LAND ALT is less than 7.8 psi (7.45 psi at 28,000 feet or lower) the controller maintains the cabin altitude during the cruise at 300 feet below the selected LAND ALT.

88. At commencement of the descent, when outside air pressure is 0.25 psi above that of the selected FLT ALT, the descent relay trips and switches the pressurisation controller to proportional control. Cabin rate of descent is now kept proportional to aircraft rate of descent, but not greater than the maximum rate of change of cabin altitude selected. On touch-down, the squat switch signal, with the FLT/GRD switch still in the FLT position, causes the controller to adjust cabin pressure to 0.1 psi above ambient pressure (airfield altitude -200 feet).

89. During taxi in the FLT/GRD switch is put to GRD and the controller drives the main outflow valve to the full open position to equalise cabin and outside air pressures.

90. The pressurisation control system will automatically trip to STANDBY mode in the event of:

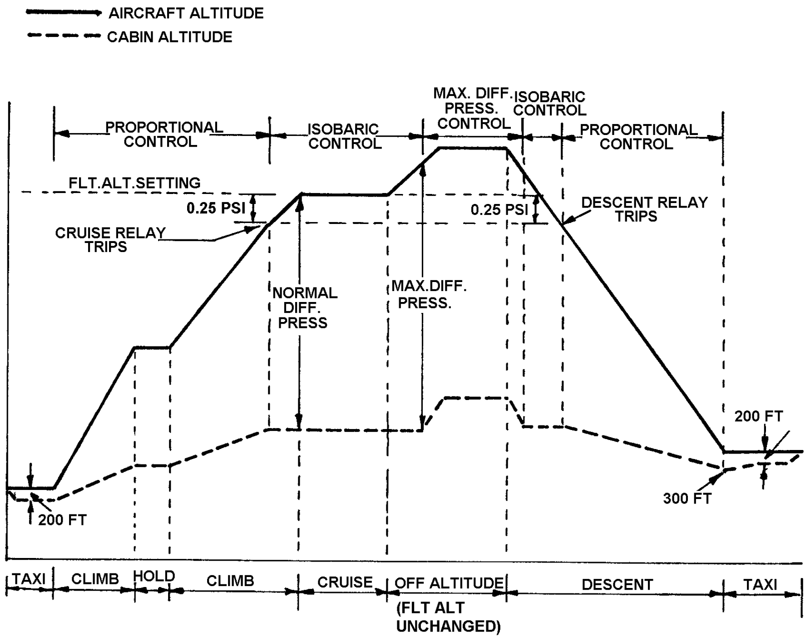
(a) Loss of AC power supply.

(b) Cabin rate of altitude change in excess of 1800 feet per minute.

(c) Excess cabin altitude (14,000 feet).

91. A typical flight profile in the AUTO pressurisation control mode is shown at Figure 4-19.

FIGURE 4-19
Auto Pressurisation Control Mode



Standby Mode Operation

92. On the ground, with the FLT/GRD switch set to GRD, the main outflow valve will be held full open. With the switch in the FLT position the standby pressurisation controller drives the main outflow valve to attempt to achieve the selected CABIN ALT. CABIN ALT should therefore be set to 200 feet below departure airfield altitude.

93. In flight a reference placard on the pressurisation control panel is used in selecting a cabin altitude based upon the intended flight altitude and pressure differential. Cabin rate of climb/descent is selected using the cabin rate selection knob, which has a range of settings from 150 fpm to 1800 fpm, and is detented to 300 fpm.

94. In the descent the cabin altitude is selected to 200 feet below destination airfield altitude, to ensure that the cabin is slightly pressurised on touch-down.

Manual Mode Operation If both AUTO and STBY modes fail, the main outflow valve opening can be modulated by means of the main outflow valve control, by reference to the valve position indicator and cabin altitude gauge.

95. The operating procedures described above are peculiar to the system illustrated at Figure 4-18. Different systems require different operating techniques and it is important that you are totally familiar with the normal and abnormal procedures which are relevant to your aircraft. For instance, some operating systems require the aircraft to be depressurised at all times whilst on the ground, with pressurisation commencing following take-off. Other systems allow for slight pressurisation prior to take-off and depressurisation on landing. In any system, manual operation of the main outflow valve imposes a high workload on the flight deck crew, especially during the climb and descent, and particularly during a stepped descent, when the workload is already high.

Instrumentation

96. All aircraft pressurisation systems contain the following minimum instrumentation:

Vertical speed indicator. This shows the rate of cabin climb or descent.

Cabin altitude warning light. A red light (often associated with an aural warning) which illuminates if cabin altitude reaches 10,000 feet.

Differential pressure gauge. This shows the difference between cabin pressure and ambient atmospheric pressure.

Cabin altimeter. A gauge reading cabin pressure, but calibrated to display this in terms of equivalent altitude.

Pressurisation System Procedures

97. The operations manual for the aircraft contains the procedures for the various modes of pressurisation control. The procedure to be followed in the event of rapid cabin depressurisation can be found in the emergency procedures section.

Limitations

98. Because of the structure of the human ear and its susceptibility to the common cold by virtue of inflammation of the eustachian tubes, it is important that cabin rates of change are kept to moderate values, especially in the descent. Because it is not common for passengers to be tested for runny noses before flight, cabin altitude rates of change should be carefully monitored and wherever possible descent rates in particular kept to a minimum.

Ground Testing

99. In attempting to limit the escape of pressurised air through uncontrolled means, any mechanical linkage which passes through the pressure hull must be adequately pressure sealed, irrespective of the type of linkage.

100. Pressurisation systems must be checked at appropriate times to ensure that there are no serious leaks and that the pressure control equipment and safety devices are functioning correctly.

101. The occasions on which tests are carried out are:

(a) When specified in the maintenance manual.

(b) After repairs or modifications which affect the structural strength of the pressure hull.

(c) After suspected damage to the fuselage.

102. The exact procedure to be followed when carrying out leak rate tests and functioning tests is laid down in the maintenance manual for the particular type of aircraft. A general guide is given below.

103. The functional testing can be carried out either by running the engines or by using an external source of supply. It is generally preferred to run the engines when functional testing, as this checks the whole system. An external supply is preferred when leak rate testing, since this removes the hazard of checking for leaks in the proximity of engine intakes, jet pipes and propellers. It also has the advantage that leaks can often be heard.

Functional Test

(a) Refer to the maintenance manual. Override dump valve/squat switch and ensure that the dump valve is fully closed.

(b) Close all external doors and windows.

(c) Open internal doors (unless otherwise specified by the manufacturer). This includes cupboards, galleys and so on.

(d) Electrical power on, pressurisation and air conditioning controls set as specified by the manufacturer.

(e) Introduce air supply at the appropriate rate.

(f) Watch pressure increase until it stabilises at maximum differential working pressure and remains stable.

(g) If more than one pressure controller is fitted, check each one separately.

(h) Operate flying controls to check the integrity of seals.

(i) Check automatic action of Safety Valves.

(j) On conclusion of test, allow pressure to drop slowly to prevent moisture precipitation in the cabin.

Leak Rate Check

(a) Introduce air into the cabin at the appropriate rate. Allow pressure to stabilise, then shut off air supply and check time taken for pressure to drop over the range appropriate to the aircraft type. Check against the maintenance manual.

(b) If leak rate is excessive, investigate, rectify and repeat.

(c) Detection of leaks can usually be carried out by ear, but a soapy water solution can be used to produce bubbles at the point of leakage.

(d) On conclusion of the test shut off air supply. Before opening doors and windows ensure pressure differential is zero.

(e) Restore all pressurisation control components to the condition they were in before testing.

(f) Examine fuselage for deformation and damage. Open and close all hatches, windows, internal doors and so on, to check for freedom of movement.

(g) Examine pressure bulkheads for deformation, and transparencies for crazing.

104. It should be noted that in both tests observers are required in the cabin. A minimum of two operators should be inside the pressurised area during pressure testing. If the test is being carried out with the engines running, a third person is required to supervise the engines. All operators must be certified fit, this includes freedom from colds and sinus problems.

Ice and Rain Protection

Ice and Rain Protection

1. In this chapter the systems employed to ensure that the airframe and the power units are protected from the effects of ice and rain are considered.

Conditions Conducive to Aircraft Icing

2. Aircraft on the ground or in flight are susceptible to accumulation of ice accretions under various atmospheric and operational conditions. Aircraft in flight can encounter a variety of atmospheric conditions which will individually or in combination produce ice accretions on various components of the aircraft. These conditions include:

(a) Supercooled clouds. Clouds containing supercooled water droplets (below 0°C) that have remained in the liquid state. Supercooled water droplets will freeze upon impact with another object. Water droplets can remain in the liquid state at ambient temperatures as low as -40°C. The rate of ice accretion on an aircraft component is dependent upon many factors such as droplet size, cloud liquid water content, ambient temperature, and component size, shape, and velocity.

(b) Ice Crystal Clouds. Clouds existing usually at very cold temperatures where moisture has frozen to the solid or crystal state. Airframe icing does not occur in these conditions.

(c) Mixed Conditions. Clouds at ambient temperatures below 0°C containing a mixture of ice crystals and supercooled water droplets.

(d) Freezing Rain and Drizzle. Precipitation existing within clouds or below clouds at ambient temperatures below 0°C where rain droplets remain in the supercooled liquid state.

Icing Problems on the Ground

3. Aircraft on the ground, during ground storage or ground operations, are susceptible to many of the conditions that can be encountered in flight in addition to conditions peculiar to ground operations. These include:

(a) Supercooled ground fog and ice clouds.

(b) Operation on ramps, taxiways, and runways containing moisture, slush, or snow.

(c) Blown snow from snow drifts, other aircraft, buildings, or other ground structures.

(d) Snow blown by ambient winds, other aircraft or ground support equipment.

(e) Recirculated snow made airborne by engine, propeller, or rotor wash. Operation of jet engines in reverse thrust, reverse pitch propellers, and helicopter rotor blades are common causes of snow recirculation.

(f) Conditions of high relative humidity that may produce frost accretions on aircraft surfaces having a temperature at or below the frost point. Frost accumulations are common during overnight ground storage and after landing where aircraft surface temperatures remain cold following descent from higher altitudes. This is a common occurrence on lower wing surfaces in the vicinity of fuel cells. Frost accretions can also occur on upper wing surfaces in contact with cold fuel (with the wing tanks full of cold-soaked fuel).

Anti-icing and De-icing

4. Anti-icing is the prevention of ice formation and may be achieved on the ground by repeatedly coating the aircraft with a freezing point depressant (FPD) fluid. In the air, surfaces prone to ice formation (leading edges of wing and tail surfaces, propeller blades, engine intakes) may be heated electrically, by hot bleed air or by heated fluids.

5. De-icing is the removal of ice or frost which has formed on the aircraft surface. On the ground this may be done by spraying the aircraft with hot FPD fluid. In flight the susceptible surfaces may be heated as described above. Alternatively, flexing leading edges (de-icing boots) may be fitted on lower performance aircraft.

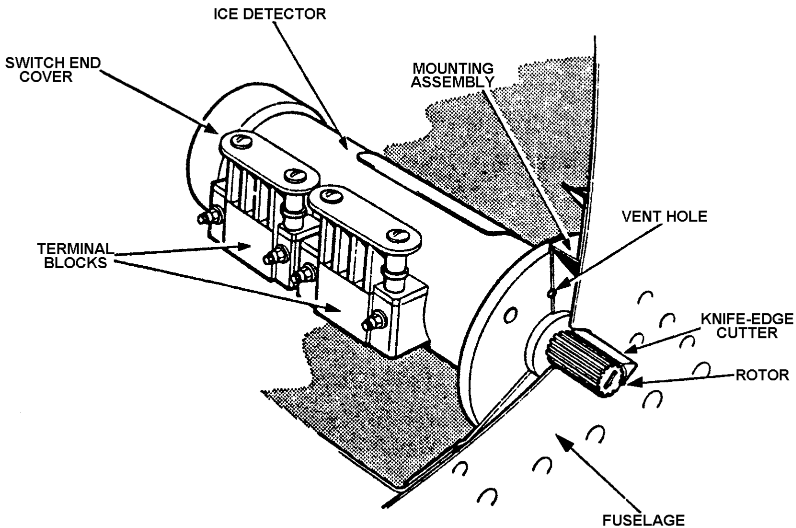
Ice Warning

6. In-flight warning of ice formation is provided by ice detection devices which activate a warning indicator on the flight deck. The traditional automatic ice detection system consists of an electric motor driving a serrated rotor which projects into the airstream from the side of the fuselage. Adjacent to the rotor is a fixed cutting edge.

7. As ice builds up on the serrated rotor it is scraped off by the cutting edge, the greater the ice build-up the greater the torque loading on the motor. The torque reaction trips a microswitch, activating a cockpit warning. The system is shown diagrammatically at Figure 5-1.

8. Pressure operated ice detector heads consist of a short aerofoil section tube mounted at right angles to the air flow and having four small holes drilled in the leading edge. During flight air enters these holes and can partially escape through smaller holes in the trailing edge, hence there is a pressure build-up in the tube and this pneumatically holds open the points of an electrical relay. In icing conditions the leading edge holes become blocked, the pressure in the tube falls and the relay points close to illuminate a warning indication on the flight deck. Vibrating rod ice detectors consist of a probe in the airstream which vibrates at approximately 40 KHz. Ice forming on the probe reduces the vibration frequency and activates the ice detector warning circuit. Both devices are illustrated at Figure 5-2.

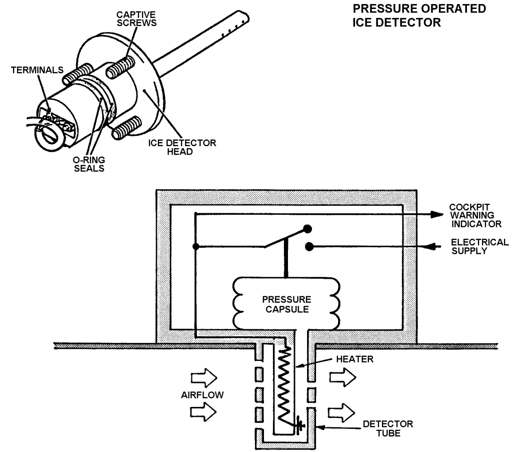
FIGURE 5-1
Ice Detection Device - Serrated Rotor

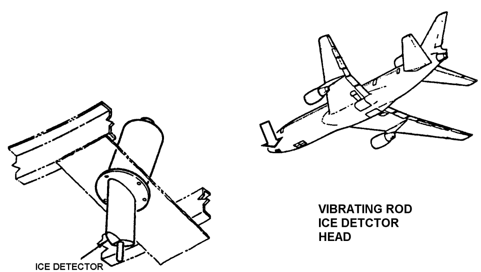


9. Visual ice detectors, in the form of small aerofoils situated externally and easily visible from the flight deck are often fitted. In icing conditions, ice will readily form on these, warning the flight crew that it will also be forming in more critical locations. These visual detectors are fitted with internal heaters, so that they can be cleared for further ice evaluation.

FIGURE 5-2

Ice Detection Devices - Pressure Operated / Vibrating Rod





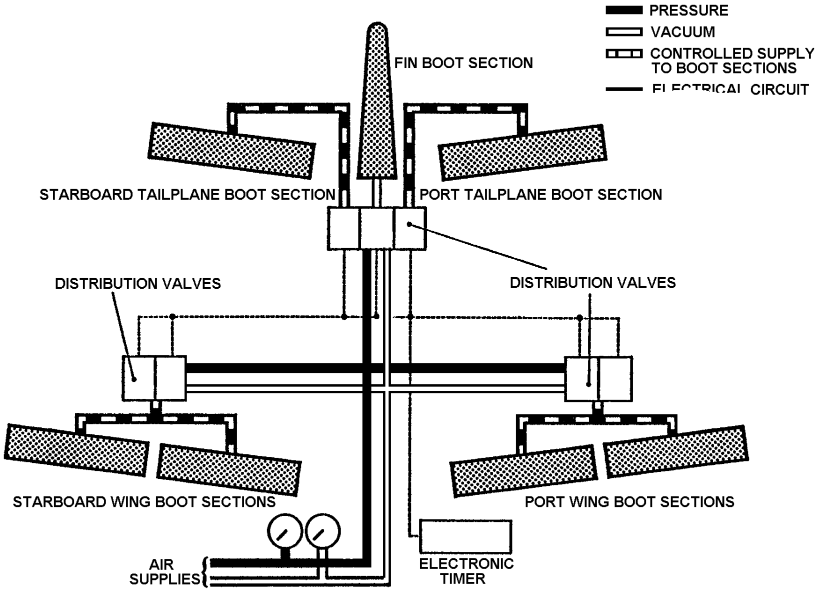
De-Icing Systems

Pneumatic De-Icing Systems

10. Some piston-engined and turbo-prop aircraft use pneumatically-operated de-icing systems. These take the form of flat inflatable tubes, closed at the ends, attached to the leading edges of wings, tailplanes and fin. The rubberised fabric tubes normally run parallel to the span of the flying surface and they are inflated and deflated cyclically with air from the aircraft's compressed air system.

11. Inflation of the tube, or boot, causes any ice layer formed to break away. When not in use a vacuum source is applied to the boots to hold them flush with the leading edge. De-icer boots may be attached to the airframe by screw fasteners or by bonding cement. A diagram of a pneumatic de-icing system is shown at Figure 5-3.

FIGURE 5-3
Pneumatic De-icing System



12. De-icing boots are only suitable for relatively low performance aircraft, since they obviously disturb airflow over the wing to some extent.

13. Flight deck controls and indications typically comprise an automatic alternating time sequence of pressure and vacuum to the de-icing boots, with pressure and vacuum gauges or lights and an on/off switch. The rubberised fabric is susceptible to damage from oils, greases, knocks, abrasions and exposure to strong sunshine.

Fluid De-Icing Systems

14. De-icing fluid is distributed to porous metal panels by an electrically-driven pump. The escaping fluid destroys the bond between ice and aircraft and is carried rearward over the aerofoil to prevent further ice build-up. A typical fluid de-icing system is shown at Figure 5-4.

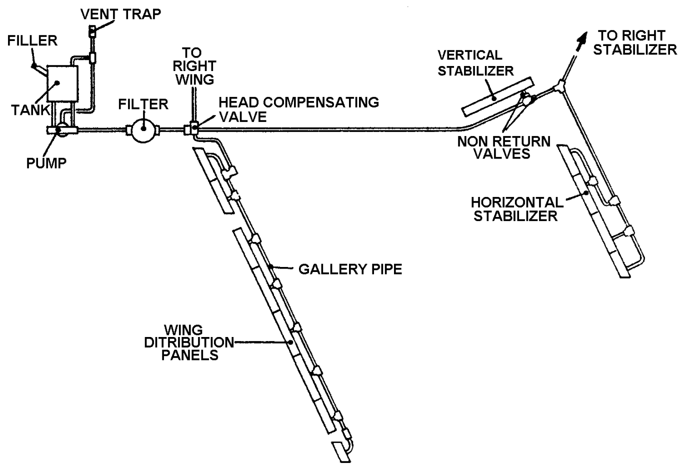


FIGURE 5-4
Typical Fluid De-icing System

De-Icing Paste

15. A hand applied de-icing paste is sometimes used on aircraft with no aerofoil de-icing system. It is spread onto the leading edges of wings and empennage and produces a surface upon which ice may form, but to which it will not bond. The ice then blows off in flight. It must be emphasised that the use of de-icing paste does not provide reliable protection for flight into known or forecast icing conditions.

Initiation / Timing of De-icing System Usage

16. Where possible the icing situation should be avoided, but where it is unavoidable pilots must make a critical analysis of the situation. Whatever system is available, be it thermal, fluid or electrical, it must be fully utilised, requiring a judgement by the pilot for initiation and duration of use. Type operations manuals will lay down recommended procedures for this eventuality. Since de-icing systems are designed to remove ice, once it has been formed, the decision for activating the system is vital to ensure adequate removal. Once switched on, most systems incorporate a cyclic timing device, which automatically controls the de-icing medium to de-ice aerofoil section leading edges or propellers. This automatic control is programmed to de-ice the sections using a timed sequence selected alternately.

17. The system is capable of continued operation with loss of associated systems (redundancy) and will have warnings of normal operation and any exceedance of limits.

18. The type operations manual will contain the procedures for operation and control dependent on flight profile i.e. some aircraft may require system off for approach and landing phase.

Propeller De-Icing

19. If ice is allowed to build-up and remain upon a propeller, the propeller becomes heavier and, more importantly, unbalanced. Vibration is generated which ultimately will cause damage to the engine and the airframe.

20. De-icing systems are used to remove ice after it has built-up. The removal is carried out by means of electrical heating elements in the blades, to melt the ice and to remove it by centrifugal and aerodynamic forces. After a short period of time, the electrical current is switched off, the ice begins to form again, the current is switched on, the ice melts, and so on.

21. The pilot controls the propeller de-icing by means of a control switch on the control panel, see Figure 5-5. Also on the control panel will be found a load meter which indicates the current being drawn. On early systems, one load meter served all of the propellers. Modern systems tend to use one load meter for each set of propellers.

22. Since the propeller blades of only one engine are being de-iced at any one moment, a timer or cycling unit is required to arrange for the application of current to the propeller blades of the appropriate engine at any particular moment. The timer in fact controls power relays located in the engine nacelles. In this way, the length of heavy current wiring is kept to the minimum.

23. Mounted on the engine front casing behind the propeller hub will be found the slip ring from which the current to the heating elements on the propellers is passed via brushes attached to the rear of the propeller hub (but the opposite arrangement may also be found on some installations). Since the propeller has remained unheated for a relatively short period of time, only a thin film of ice will have built up, without interfering with the aerodynamic characteristics of the propeller blades. The deposited ice acts as a thermal insulator. As the ice in contact with the heated propeller blade melts, the main body of ice is removed under the influence of aerodynamic and centrifugal forces.

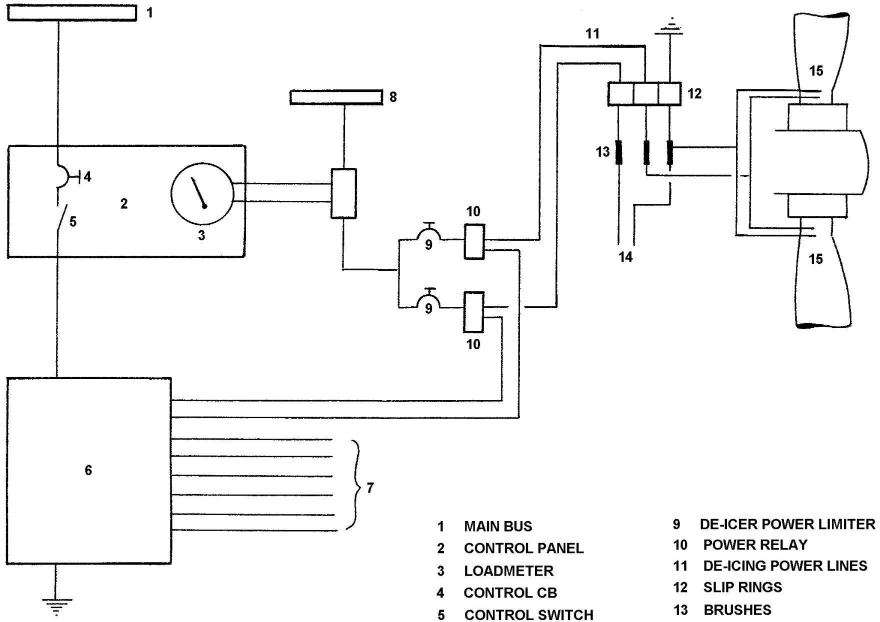
24. According to aircraft type, the current used may be either DC or AC. Although only one set of propellers is being de-iced at any one time, the timer will ensure that there is a null period of at least one second between the de-icing of one set of propellers being switched off and the de-icing of the next set of propellers being switched on. Heating current is applied to a set of propellers over a typical period of 30 seconds.

25. In the case of propellers attached to a turbo-prop engine, the propeller de-icing is part of the engine de-icing system, and the timer is part of the engine intake heating circuit. The spinner is also de-iced on turbo-prop installations.

26. Figure 5-5 shows a schematic diagram of an electrical propeller de-icing system.

27. Some turbo-prop aircraft employ thermal de-icing systems using exhaust gas heat exchangers to produce the hot air required. An alternative, combustion heater, system is used in some piston-engined aircraft.

FIGURE 5-5
Propeller De-icing System



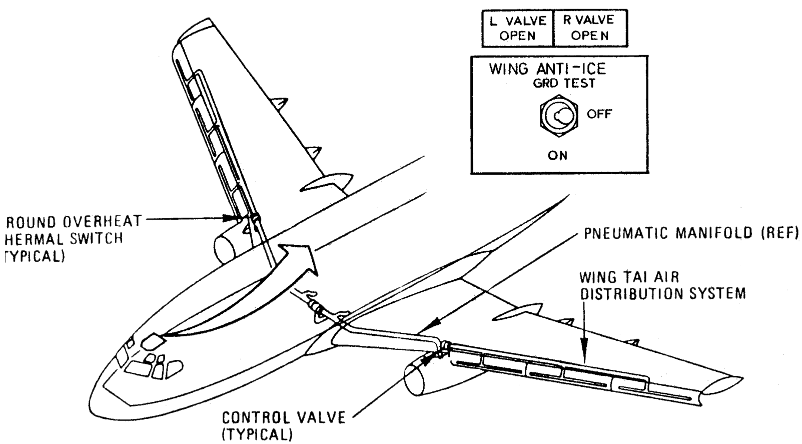
1	MAIN BUS	9	DE-ICER POWER LIMITER CB
2	CONTROL PANEL	10	POWER RELAY
3	LOADMETER	11	DE-ICING POWER LINES
4	CONTROL CB	12	SLIP RINGS
5	CONTROL SWITCH	13	BRUSHES
6	TIMING OR CYCLING UNIT	14	TO OTHER PAIR OF BLADES
7	TO OTHER PROPELLERS	15	INTERNAL OR EXTERNAL HEATING ELEMENTS
8	ENGINE NACELLE BUS		

Anti-Icing

Thermal Anti-icing Systems

28. The leading edges of wings, tailplanes and fins may be anti-iced by hot compressor bleed air supplied by the aircraft pneumatic system. A motorised shut-off valve controls the air flow into a titanium distribution duct routed through the fixed leading edge of the wing. Telescopic ducts feed the hot air into a perforated duct (piccolo tube) inside the slat surface.

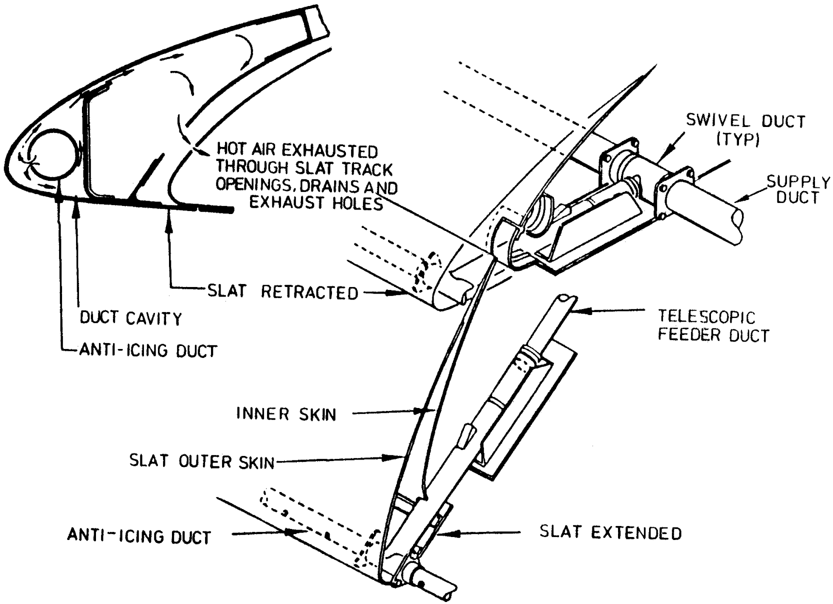
FIGURE 5-6
Thermal Anti-icing System



29. The hot air anti-ices the slat leading edge and flows into the remainder of the slat structure. The hot air is then exhausted overboard through the slat track openings and under wing drain holes. If the slats are retracted the exhausting air warms the wing leading edge section.

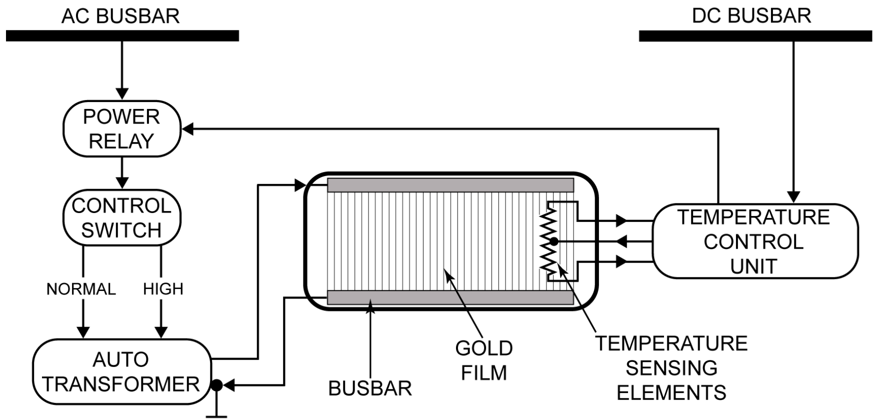
30. The temperature of the air supply to the leading edges is thermostatically controlled by a system of leading edge temperature sensing units and duct control valves. The major source of failure in these systems is leakage of the very hot bleed air from the distribution ducting, which can cause damage to other systems, particulary electrical circuits. High temperature sensing probes are usually positioned at strategic locations adjacent to the ducting.

FIGURE 5-7
Slat Anti-icing Duct



Windshield

FIGURE 5-8
Typical Electrical Anti-ice System



31. The use of electrical heating elements built into flight deck windscreens is widely used in modern transport aircraft. A layer of transparent conductive material (can be referred to as Gold Film) is supplied from the aircraft A/C electrical system. The heating process will provide a non-shattering quality to the window and the flight crew are provided with normal and failure indications. The (inner) glass panel is the load-bearing agent. The vinyl interlayer is the 'fail-safe' load carrying member and prevents the window shattering if the inner panel should fail. The outer glass panel has no structural significance, it provides rigidity and a hard scratch resistant surface. A

conductive film is applied to the inner surface of the outer glass panel to permit electrical heating for anti-icing and de-fogging. However damage of the outer panel due to arcing can lead to visibility problems. For an electrical supply failure a limited amount of de-fogging can be gained from the windscreen warm air de-misting supply. A conductive coating on the outer panel also assists in dissipating static electricity from the windscreen. Figure 5-8 shows a typical electrical windshield anti-ice system.

32. Specific problems associated with this type of system are:

Delamination Separation of the vinyl windscreen plies which if neglected can spread and cause visibility or electrical problems. Limitations of allowable delamination is laid down by manufactures.

Arcing Indicated by a breakdown in the conductive coating resulting in local overheating causing further damage to the panel. This can lead to loss of temperature control and in the extreme, panel failure.

Propeller Anti-Icing

33. Anti-ice systems are used to prevent ice build-up, as distinct from removing ice build-up. Anti-ice systems are generally ineffective once ice has built-up. Any anti-icing system must therefore be switched on before it is anticipated that the aircraft is about to enter icing conditions. When fluid anti-icing systems are used, the fluid used must have a low freezing point, and it must also combine readily with water. For this reason, isopropyl alcohol is most commonly used, since it is readily available and of relatively low cost. The major drawback of isopropyl alcohol is its high flammability.

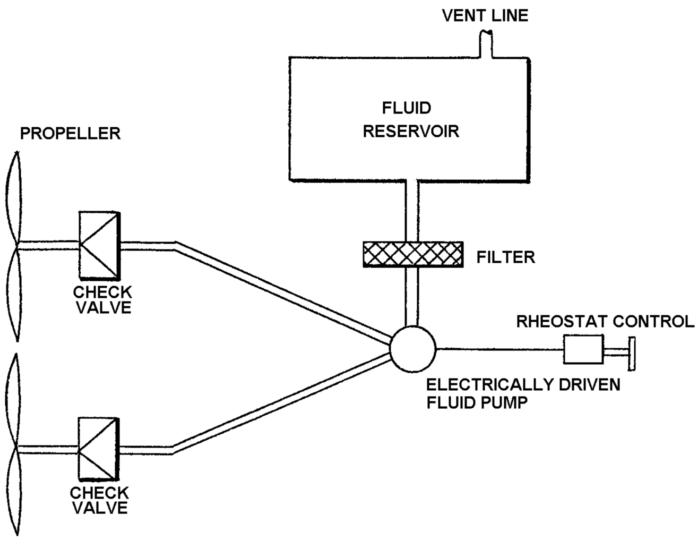
34. Figure 5-9 shows diagrammatically the main features of a typical propeller anti-icing system. The reservoir is positioned remotely from the flight deck, and so a contents gauge will be found on the flight deck. The tank must be vented to atmosphere, to ensure availability of the fluid at all contents levels. The position of the fluid reservoir must be such as to permit gravity feed to the fluid pump(s), via a filter. The fluid pump is controlled by a rheostat, so that rate of flow can be controlled to match existing conditions. Pump pressure is quite low, around 10 psi, but sufficient to open the check valves which have been found necessary to prevent syphoning of the anti-icing fluid when the system is not in use. On reaching the propeller, the fluid leaves the delivery pipe mounted in the front end of the engine, and enters the slinger ring mounted at the rear of the propeller hub. The fluid is distributed to the propeller blades from the slinger ring under the influence of centrifugal force.

35. On some installations, air pressure may be used in place of pump pressure, a relief valve is installed in the air supply line together with a control valve for regulating the fluid flow.

36. Additionally, propeller blades may have overshoes fitted, in order to assist in the efficient distribution of the fluid. These overshoes, sometimes referred to as anti-icing boots, extend out along one third of the propeller blade.

37. Normally one pump will supply two propellers.

FIGURE 5-9
Propeller Anti-icing



Power Plant and Air Intakes

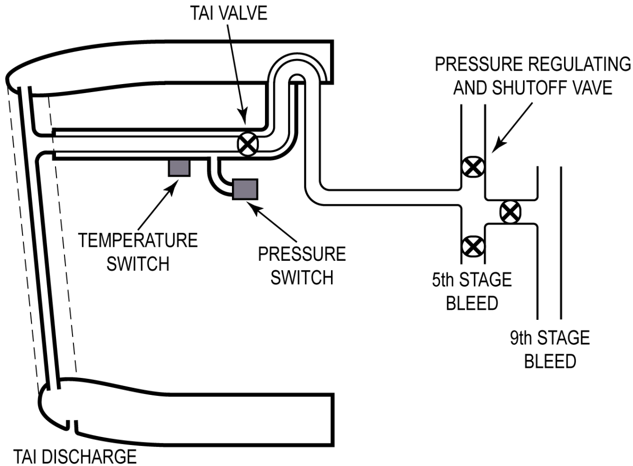
38. **Powerplant.** The Engine air intake nacelle and spinner of a gas turbine or turbo fan engine is anti-iced using the aircraft pneumatic air supply. Some systems will utilize the pneumatic air supply to additionally anti-ice the initial stage LP compressor stator vanes. Engines which use a probe for sensing intake pressure (P_1 PROBE) will use the engine ant-ice air to prevent build up on the probe.

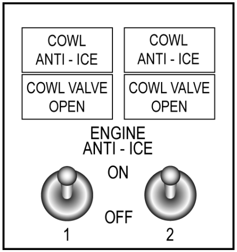
39. Turbo Prop aircraft use an electrical system employing a cyclic system to anti-ice intakes.

40. Switching for anti-ice systems is initiated by the pilot prior to entering icing conditions on the ground or when airborne. Some systems may incur a performance penalty which will be annotated in the type operations manual.

41. **Air Intakes.** Engine compressor bleed air is controlled by switching on the flight deck and is ducted to the powerplant intake cowl and, in some installations, the engine spinner. The system can be operated on the ground or during flight in known or anticipated icing conditions. Switching on the system activates engine cowl anti-ice valves that are electrically controlled and pressure operated. System normal operation and failure warning is annunciated on the flight deck. Correction to engine maximum EPR (engine pressure ratio) indication and limits is automatic upon activation of the anti-icing system. As a protection against flame-out during use of anti-icing, the engine ignition is selected ON either manually or automatically. The aircraft Operating Manual will contain procedures for normal and abnormal operation. On some installations the engine RPM must be maintained within certain parameters, dependent upon altitude, to give adequate protection. Figure 5-10 shows a schematic illustration of a gas turbine bleed air anti-icing system.

FIGURE 5-10
Anti-icing B757





Pitot Static Pressure Sensors and Stall Warning Devices

FIGURE 5-11
Anti-ice Probe Locations

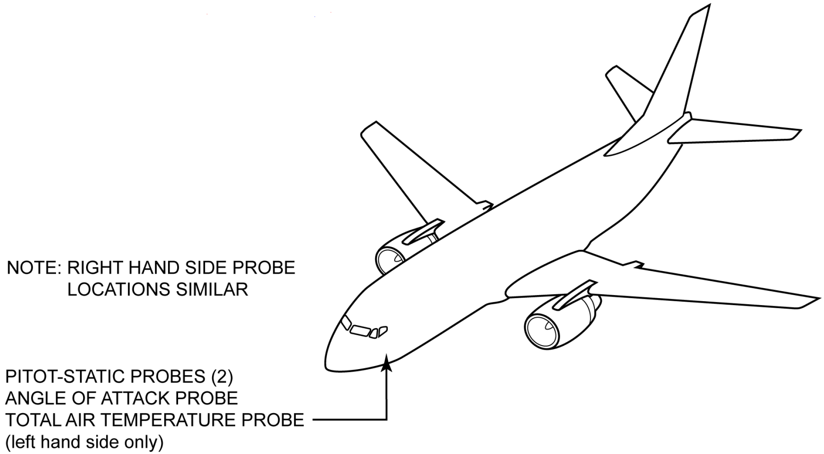


Figure 5-11 shows a schematic illustration of a probe locations.

42. Pitot probes contain an electrical heating element supplied by aircraft electrical system power. Once switched on they are thermostatically controlled. Normal and failure warnings are provided on the flight deck. Some provision is made for lower voltage applied on the ground to protect against burning out the element. This type of control is normally activated by the undercarriage weight (squat) switches. Stall warning detectors are anti iced in a similar manner.

43. Where a TAT (Total Air Temperature) system is used this also will be anti-iced by electrical elements.

Rain Removal

44. During flight in heavy rain it may be necessary to cease the operation of the windshield wipers with application of a rain repellant. The main thing to remember about rain repellant fluid is that, when inadvertently selected with a dry windscreen, it will obscure the view through the windscreen rather than improving it.

Rain Repellent

45. During heavy rain a chemical is sprayed onto the windscreen which forms on the surface and causes the rain water to form beads, leaving large areas of dry glass in between. The beads of water are easily removed by the natural airstream.

46. Activation of the system is by switch or push button on the flight deck. Irrespective of switch/button operation only a measured amount of fluid will be applied.

47. The system should not be applied to dry windscreen surfaces because it can cause smearing and restrict visibility. After application the film will gradually deteriorate with further rain impingement. The length of time between applications is dependent on rain intensity, type of repellent and windscreen wiper usage.

Fuel & Fuel Systems

Fuel & Fuel Systems

Aircraft Fuels

1. Aircraft engines use either gasoline (piston engines) or kerosene (gas turbines). It is vital to safety that the correct fuel is used.

Aviation Gasoline (AVGAS)

2. Aviation gasoline (AVGAS) is only generally available in the UK as AVGAS 100 LL (low lead). It is coloured blue, has a specific gravity of approximately 0.72, and must comply with Directorate of Engineering Research and Development (DERD) specification 2485. 100 LL has an octane rating of 100/130, the 100 corresponding to the anti-knock qualities when a lean mixture is used and the 130 to the anti-knock qualities when a rich mixture is used.

3. There are two other categories of AVGAS which may be available. 100 AVGAS is coloured green and has identical anti-knock characteristics to 100 LL (100/130 octane rating under lean/rich conditions) but obviously contains more tetra-ethyl-lead (TEL) than 100 LL and is therefore considered to be environmentally unfriendly. Avgas 80 is coloured red and has a lean/rich octane rating of 80/87.

Automobile Gasoline (MOGAS)

4. Automobile gasoline (MOGAS) may be available, but its use implies limitations. It is more volatile than AVGAS and thus more prone to cause vapour lock in fuel lines and pumps, especially at high temperatures and/or altitudes. It does not possess the anti-knock qualities of AVGAS and may lead to pre-ignition and detonation in some aircraft piston engines. Detergents used in MOGAS may be harmful to aircraft fuel system components and alcohol, found in some grades of MOGAS, will cause deterioration of the seals in most aircraft fuel systems, with the obvious hazard to flight safety.

5. Most major aircraft manufacturers have refused to approve the use of MOGAS, using it may invalidate the engine manufacturer's guarantee. It is used in certain light aircraft/engine combinations, but limited by conditions of fuel specification, source of supply, storage and operating precautions. In particular, in order to minimise the risk of vapour lock, the temperature of fuel in the aircraft tanks must be less than 20°C and the aircraft must not be flown above 6,000 feet. Because of the higher volatility of the fuel (and possible water content) carburettor hot air systems must be functional and operated, since the risk of carburettor icing is greater than with AVGAS.

Turbine Fuel (AVTUR)

6. Turbine fuels (Avtur) for use in turbo-prop and turbo-jet engined aircraft may be categorised into two types. These are Kerosene and wide-cut gasoline fuels. Kerosene type fuel has a higher density but slightly lower heating value than gasoline. Kerosene has an extremely low vapour pressure, resulting in neglible fuel boiling and vapour losses.

7. An advantage of wide-cut gasoline is its low freezing point and higher volatility than kerosene and its ability to produce flammable mixtures over a wider range of temperature and altitude conditions.

8. Turbine fuel (Avtur) is not colour coded and is naturally colourless or straw coloured. The commonly used types of AVTUR are:

JET A: Primarily available for commercial use in the USA. It is also known as 'Civil Aviation Kerosene' and contains no gasoline blend. Jet A has a specific gravity of 0.807 at +15.5°C and has a maximum specified freezing limit of -40°C.

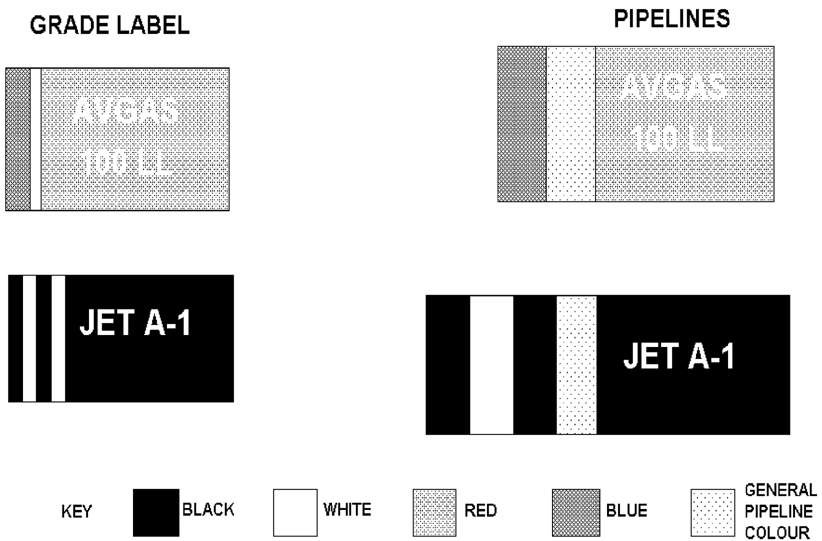
JET A-1: The fuel most commonly used by international airlines. A kerosene fuel (again with no gasoline blend) with a lower freezing point than JET A. JET A-1 has a specific gravity of 0.807 at +15.5°C, has a maximum specified freezing limit of -47°C and must comply with DERD specifications 2494. It is also available with fungus suppressant and icing inhibitor (FSII) additives, when it meets DERD specification 2453.

JET B: A blend of approximately 30% kerosene and 70% gasoline, known as a wide-cut fuel. It has a very low freezing point of -60°C but also a low flash point. Primarily used by US (and some other) military aircraft. It has a DERD specification of 2486. Because of its greater volatility it presents a considerably greater fire hazard than does JET A-1. This applies particularly to the vapour which forms in partly-filled fuel tanks and which is more flammable than either JET A-1 or AVGAS vapour. JET B has a lower SG than JET A-1 and when used adjustment of the engine fuel control unit (FCU) is necessary.

Colour Coding and Labelling

9. Aircraft fuelling points, re-fuelling vehicles, pipes and ground installations are marked with distinctive colour codes to distinguish gasoline and kerosene supplies. The labelling and colour coding used is illustrated at Figure 6-1.

FIGURE 6-1
Fuel Labelling and Colour Coding



Fuel Sampling

10. Before flight each day a small quantity of fuel should be drawn from each of the aircraft tank drain valves and inspected in a glass container. Fuel should be considered unfit for aircraft use if visual inspection shows:

(a) More than a trace of sediment.

(b) Globules of water.

(c) Cloudiness.

(d) A positive reaction to water finding paste, or a paper or chemical detector.

11. The following should serve as a guide to the visual assessment of aircraft fuels:

Colour. AVGAS 100LL is dyed blue, AVTUR JET A-1 is undyed and can vary in appearance from colourless to straw yellow.

Undissolved water (free water) will appear as droplets on the sides of the sampling vessel, or as bulk water in the bottom.

Suspended water (water in suspension) will cause the fuel to appear hazed or cloudy.

Solid matter (rust, sand, dust, scale, etc) may be suspended in the fuel or settle on the bottom of the sampling vessel.

Clear and Bright. The fuel sample should appear clear and bright. Clear refers to the absence of sediment or emulsion and bright refers to the sparkling appearance of fuel which is free from cloudiness or haze.

12. Because of its lower SG, fuel floats on water. If there is no obvious separation of fluids, test the sample by smell, colour and finally by evaporation test. Empty the sample onto a smooth surface and see how quickly it evaporates. The fluid in the container could be all water!

Stored Fuel Quality Control

13. Refuelling vehicles and bulk fuel storage facilities must be regularly checked for fuel quality. Water-finding paste (or tip tape) applied to the end of a dipstick is used for direct checking. Fresh paste (or tape) must be used for each check and the dipstick allowed to rest on the bottom of the container for a short period, not exceeding 10 seconds.

14. More extensive quality control checks are also conducted regularly. The size of fuel sample taken from the storage facility should be sufficient to complete a full and conclusive check of the state of the fuel and should typically be approximately 1 Imperial Gallon (approximately 4.5 litres). Fuel samples are drawn up from buried tanks and barrelled supplies by means of a thief pump. These samples are checked for colour, sediment, water globules, cloudiness and general cleanliness.

Refuelling

Precautions During Fuelling Operations

15. **Refuelling.** Refuelling of aircraft is carried out either by the overwing method or pressure refuelling. The overwing method involves placing the refuel nozzle of the refuel hose into the overwing refuel opening in the top of the tank. The fuel is then transferred under low pressure from bowser to tank. The pressure refuelling method is used on most jet transport aircraft Fuel under pressure and high flow rate is transferred from bowser/dispenser via a coupling at a refuel panel usually located under one wing of the aircraft. The panel houses refuel valve selection switches and fuel tank gauges. Pre-set contents or full tank selections can be made at the refuel panel so that the operator can select which tanks are to be replenished. When a tank becomes full, a float operated shut off valve is closed by the rising fuel level.

16. The following general precautionary measures should be taken during aircraft fuelling operations:

- The main aircraft engine(s) should not be operated. The main aircraft engines should not be used to power the aircraft electrical systems during fuelling.
- Bonding, as appropriate, should be carried out in accordance with the appropriate guidelines.

17. Fuelling vehicles and equipment should be positioned so that:

- Access to aircraft for rescue and fire fighting vehicles is not obstructed.
- A clear route is maintained to allow their rapid removal from the aircraft in an emergency.
- They do not obstruct the evacuation routes from occupied portions of the aircraft in the event of fire including chute deployment areas.
- Sufficient clearance is maintained between the fuelling equipment and the aircraft wing as fuel is transferred.
- They are not positioned beneath the wing vents.
- There is no requirement for vehicles to reverse before departure.
- All other vehicles performing aircraft servicing functions should not be driven or parked under aircraft wings while fuelling is in progress.
- All ground equipment such as rostrums, steps etc., should be positioned so that the aircraft settling under the fuel load will not impinge on the equipment.
- If an auxiliary power unit located within the fuelling zone or which has an exhaust efflux discharging into the zone is stopped for any reason during a fuelling operation it should not be restarted until the flow of fuel has ceased and there is no risk of igniting fuel vapours.

- Aircraft batteries should not be installed or removed nor should battery chargers be connected, operated or disconnected.
- The practice of connecting and disconnecting ground power generators and the use of battery trolleys to supply power to an aircraft during the fuelling process within the fuelling zone should be prohibited. No aircraft switches, unless of the intrinsically safe type, should be operated during this time. However, connections may be made prior to the start of fuelling and the circuit should then remain unbroken until fuelling has ceased.
- No maintenance work which may create a source of ignition should be carried out in the fuelling zone.
- Oxygen systems should not be replenished.
- The Aerodrome Authority - Air Traffic Control should issue guidance, depending on local conditions, as to when fuelling operations should be suspended due to the proximity of severe electrical storms.
- Aircraft external lighting and strobe systems should not be operated.
- Aircraft combustion heaters should not be used.
- Only checking and limited maintenance work such as the exchange of units should be allowed on radio, radar and electrical equipment. Any use or testing of such equipment should be deferred until fuelling is completed.
- When passengers are embarking or disembarking during fuelling their route should avoid the fuelling zone and be under the supervision of an airline official. The use of personal hand held telephones by passengers should not be permitted. The 'NO SMOKING' rule should be strictly enforced during such passenger movements.

Fuelling Operations with Passengers on Board

- To reduce turnround time and for security reasons, airline operators of fixed wing aircraft may allow passengers to embark, disembark or remain on board during fuelling operations provided the following safety procedures are followed.
- Fixed wing aircraft with a seating capacity of less than 20 should not be permitted to be fuelled with passengers on board.
- When wide cut turbine fuels (e.g. Jet B, JP4, Avtag) are involved and the fuel being supplied does not contain an anti-static additive, it is advisable that passengers should disembark before fuelling. Passengers should always be required to disembark when AVGAS is involved.
- Cabin attendants, passengers and other responsible staff should be warned that fuelling will take place and that they must not smoke, operate electrical equipment or other potential sources of ignition.
- The aircraft illuminated 'NO SMOKING' signs should be on together with sufficient interior lighting to enable emergency exists to be identified. Such lighting should remain on until fuelling operations have been completed. The 'Fasten Seat Belts' signs should be switched off and passengers should be briefed to unfasten their seat belts.
- Provision should be made, via at least two of the main passenger doors, (or the main passenger door plus one emergency exit when only one main door is available), and preferably at opposing ends of the aircraft, for the safe evacuation of passengers in the event of an emergency. Throughout the fuelling operation these doors should be constantly manned by a cabin attendant.
- Ground servicing activities and work within the aircraft, such as catering and cleaning should be conducted in such a manner that they do not create a hazard or obstruct exits.

- Inside the aircraft cabin the aisles, cross aisles, all exit areas and exit access areas should be kept clear of all obstructions.
- Whenever an exit with an inflatable escape slide is designated to meet the above requirements, the ground area beneath that exit and the side deployment area should be kept clear of all external obstructions and the fuelling overseer informed accordingly.
- Access to and egress from areas where other slides may be deployed should also be kept clear.

Wide Bodied Aircraft And All Other Aircraft Equipped With Automatic Infalatable Chutes

NOTE:

When a loading bridge is in use no additional sets of aircraft steps need be provided. However, either the left or right rear door should be manned constantly by a cabin attendant and should be prepared for immediate use as an emergency escape route using the automatic inflatable chute. Where slide actuation requires the manual fitting of an attachment to the aircraft e.g. girt bar, the slide should be engaged throughout the fuelling process.

18. As a precautionary measure when a loading bridge is NOT available for use one set of aircraft passenger steps should be positioned at the opened main passenger door normally used for the embarkation and/or disembarkation of passengers.

Aircraft Not Equipped With Automatic Inflatable Chutes

NOTE:

When a loading bridge is in use, one set of aircraft steps should be positioned at another opened main passenger door and preferably at the opposing end of the aircraft.

When a loading bridge is NOT available for use, aircraft passenger steps should be positioned at two of the main passenger doors (i.e. preferably one forward and one aft) which are to be open.

Where aircraft are fitted with integral stairways and these are deployed, each may count as one means of egress.

Concorde Aircraft

19. Whether passenger steps or a loading bridge is in use, the area on the ground in the vicinity of the left hand centre passenger door (immediately forward of the leading edge of the wing) should be unobstructed by ground equipment to allow the escape chute from this door to be deployed.

Cabin Attendants

20. Cabin attendants are required to supervise passengers and to ensure aisles and emergency doors are unobstructed.

21. The aircraft operator should ensure that at all times during aircraft fuelling with passengers on board, that there are sufficient cabin attendants on board the aircraft to secure the rapid safe evacuation of passengers if an incident occurs. In determining the minimum number of cabin attendants required, aircraft operators are to take into account the number of passengers on board the aircraft, their location within the cabin, the size of the aircraft and the emergency exits and escape facilities. The following minimum cabin attendants should be on board:

22. 1 cabin attendant for every 50 seats (or fraction thereof) on the aircraft, with at least one cabin attendant for each separate passenger compartment.

23. If during fuelling, the presence of fuel vapour is detected in the aircraft interior or any other hazard arises, the Fuelling Overseer (who should ensure that he/she has adequate means of communication) should be informed. Fuelling should be stopped until, in the opinion of the Fuelling Overseer, it is safe to resume.

NOTE:

These precautions may not be observed at some locations throughout the world. Knowledge of local precautions is necessary.

De-fuelling

24. When an aircraft is de-fuelled, in whole or in part, that fuel must not be returned to aircraft tanks unless satisfactory quality checks are obtained. Fuel taken from an aircraft's tanks may be contaminated with water or sediment and may not conform to the fuel specification normally provided at that aerodrome.

25. Consequently, de-fuelling should be into an empty vehicle or storage tank segregated from other parts of the installation. The fuel remains the property of the aircraft operator and its disposal is his responsibility.

Definitions

Anti-Knock Value	A fuel's resistance to detonation. A fuel said to have a good anti-knock value when its detonation resisting quality is good, compared with other fuels under the same operating conditions.
Flash Point	The lowest temperature at which fuel produces sufficient vapour to be ignited by a small flame or spark.
Freeze Point	The temperature at which visible solid fuel particles (crystals) disappear on warming.
Spontaneous Ignition Temperature	The temperature at which fuel vapour will ignite in air at atmospheric pressure, even though an external ignition source is not present. Also known as 'auto ignition temperature'.
Viscosity	A measure of a fluid's internal friction, or resistance to motion. The lower the viscosity, the more freely a fluid will flow or pour.
Volatility	The tendency of a fuel to vaporise (change from liquid to vapour). An ideal aircraft fuel is one which has volatility high enough to permit easy engine starting, but not so high as to readily form excessive vapour in the fuel system.

Fuel Systems

26. In this section the major components of a typical fuel system are considered.

Fuel Tanks

27. Fuel tanks are found in three types; integral, flexible and rigid. Integral tanks are formed by sealing part of the aircraft structure (usually wings), whereas flexible tanks are manufactured to fit into an available space in wing or fuselage. Rigid tanks are sometimes used for extra fuel storage in the aircraft fuselage or, in military aircraft, may be carried externally.

28. **Integral** tanks are incorporated in the aircraft wing structure during manufacture. In large aircraft the whole of the wing torque box (the 'box' formed by the front and rear spars and the upper and lower wing skins) is sealed and compartmented to contain fuel. The fuel-tight seal is achieved by inserting a sealant compound between mating surfaces and by coating the whole of the tank interior with a fuel-proof coating. In light aircraft often only the wing centre section is used as an integral fuel tank, many do not incorporate integral tanks at all. Semi bulkheads (baffles) are frequently fitted internally within large fuel tanks in order to prevent fuel surge in the tanks during aircraft manoeuvres.

29. **Flexible** tanks are made from re-inforced rubber or plastic sheeting and they are attached by means of cords or buttons to the structure surrounding the tank bay. Wing fuel tanks in light aircraft are often of the flexible type.

30. **Rigid** tanks are sometimes used for fuel storage in light aircraft, mounted in the fuselage. They are generally constructed of aluminium alloy.

31. In single-engined high winged aircraft a single, gravity-feed, fuel tank is often adequate. A filter and shut-off valve between tank and engine are the only essential system requirements. Low-wing monoplanes will require a fuel pump to supply fuel from tank to engine. The tank, or tanks, are usually mounted at the aircraft's longitudinal centreline as near the centre of gravity as possible.

32. Multi-engined aircraft normally have the fuel tanks contained in the wing structure (the fuselage being required for payload). The tanks will be compartmented, typically on the basis of one tank per engine plus a centreline tank.

33. Fuel tanks must be vented to atmosphere (via a non-return valve) in order to maintain ambient atmospheric pressure inside the tank. This prevents a partial vacuum forming in the tank as fuel is used. Provision is made for draining off fuel/water from the lowest point in the tank, in the form of a sump drain.

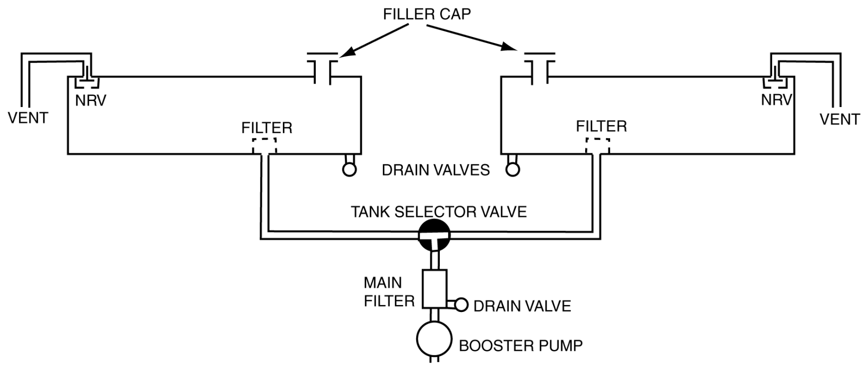
34. The fuel tank system on large jet transport aircraft incorporate vent surge tanks. They are constructed the same as integral fuel tanks and form part of the fuel tank venting system. Surge tanks are normally empty and are designed to contain fuel overflow and prevent spillage, especially during refuelling to full tank loads. Some vent surge tanks are fitted with a pump to transfer the accumulated fuel back into a main tank.

35. Fuel tank vent systems also contain overpressure relief valves to prevent any excessive positive pressure build up in the vents during ram air venting. Some systems incorporate positive and negative relief valves to prevent wing damage due to positive or negative pressure in the vent system at all aircraft attitudes.

Fuel Feed Systems

36. A simple fuel system of the type found in modern light aircraft is shown at Figure 6-2.

FIGURE 6-2
Simple Fuel System



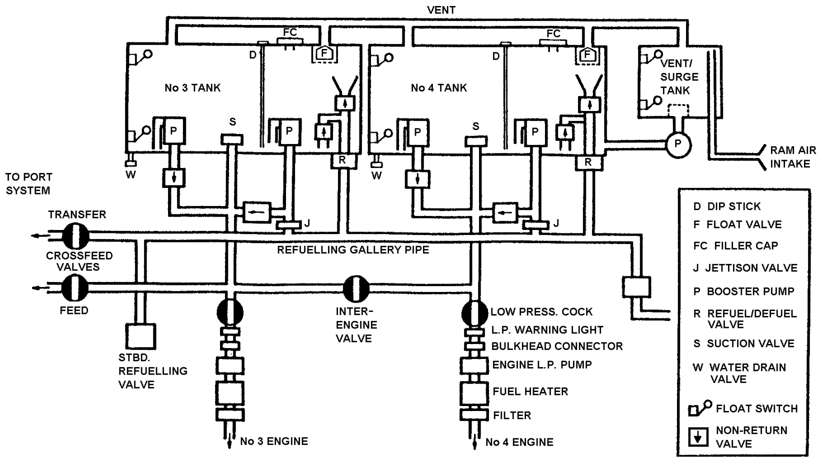
37. The booster pump serves to provide positive fuel supply to the engine-driven fuel pump during critical flight phases such as take-off and climb, approach and landing. The booster pump may be fitted in the fuel feed line, as shown, or it may be immersed at the bottom of the fuel tank. Some high-winged light aircraft rely upon gravity feed instead of a booster pump.

38. The fuel tanks are vented to atmosphere in order to prevent a partial vacuum forming in the tanks as fuel is drawn off. Were this allowed to occur, fuel flow from the tanks would eventually cease. It is good practice to completely fill the tanks of aircraft which are to be parked in the open for any length of time. This reduces the risk of condensation forming in the tanks, but could lead to fuel venting overboard when ambient temperatures subsequently rise.

39. Multi-engined aircraft require more complex fuel supply systems to accommodate re-fuelling and tank venting in addition to provision for cross-feeding. In some cases, the ability to jettison (dump) fuel in flight may be necessary. The starboard half of a fuel system for a four-engined aircraft is shown at Figure 6-3.

40. Each tank contains two booster pumps, either of which can supply the needs of any one engine. In the event of failure of both pumps the engine-driven low pressure (LP) fuel pump will continue to supply the engine, but aircraft operating altitude may have to be reduced in order to prevent cavitation in the feed line. The booster pumps are electrically driven, usually by an AC induction motor, whereas in light aircraft the booster pump is usually operated by a DC motor.

FIGURE 6-3
Fuel Cross-feed System



41. From Figure 6-4 it can be seen that, by use of the cross-feed and inter-engine valves, the system can be operated so that each engine draws from its own tank (cross-feed and inter-engine valves closed), port engines from port tanks, starboard engines from starboard tanks (cross-feed valve closed, inter-engine valves open) or all engines from all tanks (cross-feed and inter-engine valves open).

42. The booster pumps draw fuel from above the bottom of the tank, to avoid water residue reaching the engines. Consequently there is always a quantity of fuel at the bottom of the tanks which cannot be pumped and this forms part of the unusable fuel.

Fuel Jettisoning

43. In large transport aircraft it may be necessary to jettison fuel in a flight emergency, in order to reduce all-up weight for landing. Some aircraft are equipped with provision for this. Typically, the re-fuelling gallery is extended to an open-ended pipe near the wingtip, protected by a jettison valve. One of the booster pumps in each tank draws fuel from a standpipe, and it is this pump which is used when fuel jettison is selected. This ensures that a specified quantity of fuel remains in the tanks. Such a system is illustrated at Figure 6-3. In some aircraft gravity-draining replaces the use of the booster pumps during fuel jettisoning.

44. In another type of system, fuel is jettisoned through a pipe in each wing, the pipe being lowered into the airstream by an electrically-operated actuator. A short manifold is fitted between the main tanks in each wing, and a jettison valve controls flow from each tank into the manifold. Auxiliary tanks are fed into the main tanks by the normal transfer valves, the transfer pumps being inter-connected with the circuits operating the jettison valves. When the jettison pipe is in the retracted position it forms a seal at the manifold, and acts as a master jettison valve. The circuits controlling the jettison valves are not armed until the jettison pipe is locked in the extended position.

45. National legislation stipulates the aircraft operating conditions under which fuel may be jettisoned. Below certain altitudes and in certain climatic conditions it is only safe to jettison fuel over the sea and fuel dumping may only be carried out with the expressed authorization of ATC. As jettisoned fuel escapes from the aircraft it quickly breaks up and vaporises, hence it presents a fire hazard. To reduce this, jettisoning should never be attempted near electrical storm activity. In aircraft with underwing engines, dumping is carried out with flaps retracted to prevent any possibility of fuel vapour being directed into the exhaust gas stream.

46. A means must be provided in the fuel system to prevent jettisoning the fuel in the tanks below a level that would permit a climb from sea-level to 10,000ft and thereafter allowing 45 minutes cruise at a speed for maximum range.

47. A fuel jettison system must be installed on each aeroplane unless it can be shown that the aeroplane meets a climb gradient of 3.2% at maximum take-off weight less the fuel necessary for a 15 minute flight comprised of a take-off, go-around and landing at the airport of departure with the aeroplane configuration, speed, power and thrust the same as that used in meeting the applicable take-off, approach and landing climb performance requirements.

Fuel System Monitoring

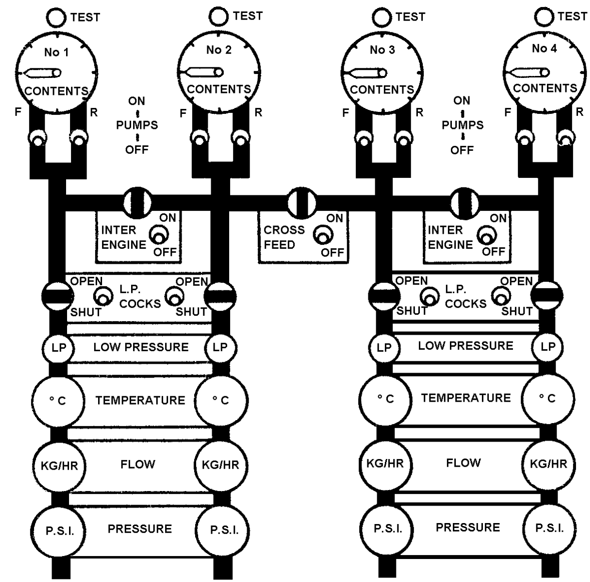
48. It is important that the fuel system is properly managed in order to maintain supply of fuel to the engines, whilst avoiding excessive movement of the aircraft centre of gravity or undue stresses on the wings. Consider, for example, a large swept-wing aircraft with two tanks in each wing (inboard and outboard) and a fifth, centre-section tank.

49. At high all-up weights there is structural benefit in concentrating weight in the outer wings to reduce the bending moment of high wing loads. As fuel is used, however, the centre of gravity will shift because of wing sweep. If all the usable fuel in the wing tanks is burned off, the landing weight distribution will be wrong. Consequently, management of the fuel to meet the various criteria is a complex business.

50. To facilitate fuel system management the controls and indications are grouped together on a control panel. Figure 6-4 illustrates a control panel which would be associated with the fuel system at Figure 6-3. It will be seen that this provides control of booster pumps, engine LP cocks and the valves in the cross-feed system. Fuel temperature, pressure and flow rate are indicated on gauges, with warning lamp indication of low fuel pressure (booster pump failure). Fuel tank contents are usually measured by a capacitive system in terms of weight, rather than volume. This is convenient since the energy available in the fuel (calorific value) is directly proportional to its weight (mass).

51. Normal operation of a multi-engined aircraft fuel system is with the cross-feed closed and fuel feed from tank to engine. In aircraft fitted with a centre wing tank (CWT) in addition to the main wing tanks it is normal practice to switch to this tank after take-off and empty it before switching back to the main tanks. The cross-feed is used to transfer fuel if a lateral imbalance occurs, or to feed an engine from other than its 'own' fuel tank. The 'engine out' configuration, especially in a twin-engined aircraft, would be with cross-feed open.

FIGURE 6-4
Fuel System
Control Panel



Example of Fuel Management

52. A four engined aircraft with wing and centre section (auxillary) tank(s) will require the centre section tank contents (except for any fuel required as ballast) to be consumed first.

53. For take-off it is normal to select direct tank-to-engine feed. When established in the climb, the fuel system will be configured to supply fuel from the centre tank, backed up by booster pumps in the inboard wing tanks. The outboard wing tank booster pumps remain off.

54. By arranging the centre tank pumps to have a higher output pressure than the wing tank booster pumps, fuel will flow from the centre tank. With the inter-engine valves selected open, all engines will be fed from the centre tank.

55. When the centre tank is empty, the inboard tank booster pumps will continue to supply all engines, and the centre tank pumps can be turned off. A tank pump low pressure warning light illuminates for each pump when the tank is empty.

56. When the quantity of fuel in the inboard tanks is the same as the outboard tanks, the outboard tank booster pumps are turned on and the inter engine valves closed. The fuel system is now configured for tank-to-engine feed for the remainder of the flight, unless uneven fuel burn or an engine failure results in fuel load asymmetry.

57. The booster pumps are located at the low point of the fuel tanks, one booster pump forward, one aft. During take-off and climb, the aft booster pumps are used. During descent with a low fuel quantity, the rear booster pump inlets may be uncovered and the forward booster pumps should be used. In the event of a go-around at a low fuel quantity, the aft booster pumps should be used, as aircraft acceleration and climb may uncover the forward booster pump inlets.

Fuel Heating

58. It is usual in large aircraft for the fuel to be heated before it passes through the low pressure filters. This is to ensure that any droplets of water in the fuel do not enter the filters as ice and cause blockage of the fuel supply to the engines. Solid, wax-like particles begin to form in kerosene (Jet A) at temperatures below -40°C. These particles are capable of clogging fuel filters and again heating the fuel prevents this.

59. In many cases the fuel is heated by passing it through oil/fuel heat exchangers in which heat from the engine lubricating oil is transferred to the fuel. An alternative, and in some instances additional, method employs air/fuel heat exchangers in which the heating medium is compressor bleed air. Both are usually thermostatically controlled, although manual control of air/fuel heat exchangers is sometimes employed.

60. Manually controlled fuel heating systems incorporate a warning lamp, activated by a pressure differential switch in the fuel filter. If fuel flow through the filter is restricted by ice or wax the differential pressure across the filter increases until it is sufficient to operate the icing warning lamp. The heat exchanger valve is then opened to admit hot compressor bleed air to heat the fuel until the fuel temperature gauge reaches a limiting value. Thermostatically-controlled systems are fully automatic, maintaining fuel temperature within pre-set limits above 0°C.

Fuel Quantity Measurement Systems

Capacitive

61. The principle of electrical capacitance is described in the AC Electricity section. A capacitor consists of two conducting plates in an electrical circuit, separated by a material resistive to current flow called the dielectric. The device will store a charge of electricity supplied to it, and will discharge that stored charge back into the circuit if the charging current is removed. The magnitude of the stored charge and discharge (the capacitance), depends upon the dielectric strength of the material which separates the conducting plates of the capacitor.

62. Capacitive fuel quantity measurement systems utilize a capacitive probe which is immersed in the fuel tank. The probe consists of two conducting tubes, with one tube inside the other. These extend from the top of the fuel tank to the bottom and they form the conducting 'plates' of a capacitor. When supplied with alternating current, the capacitor alternately stores an electrical charge and then discharges it back into the circuit. The magnitude of the discharge is determined by the dielectric strength of the separator. In this case the dielectric is the fuel in the tank or, if the tank is empty, the air in the tank. Consequently, as the fuel level in the tank rises or falls the dielectric strength of the capacitor varies. Fuel has a dielectric strength more than twice that of air so the capacitance, or charge-storing capability, of the capacitor varies in proportion to the fuel level in the tank. The magnitude of the capacitor discharge is measured and produces a signal whose strength is proportional to tank contents.

63. The output from the capacitive probe is used in a bridge circuit to provide an amplified signal, which actuates the tank fuel contents gauge. The principle of a bridge circuit is explained shortly in the description of resistive fuel quantity measurement.

64. Capacitance systems measure fuel quantity by weight, rather than volume. The probes measure the volume of fuel in the tank, a compensator applies a correction for fuel temperature and a densitometer adjusts for different fuel specific gravity or type (Jet A-1 or Jet B). This is advantageous because the fuel energy available is dependent upon the weight of fuel, not its volume (the number of molecules present in a given quantity of a substance depends upon the mass of that quantity, regardless of the volume occupied by it).

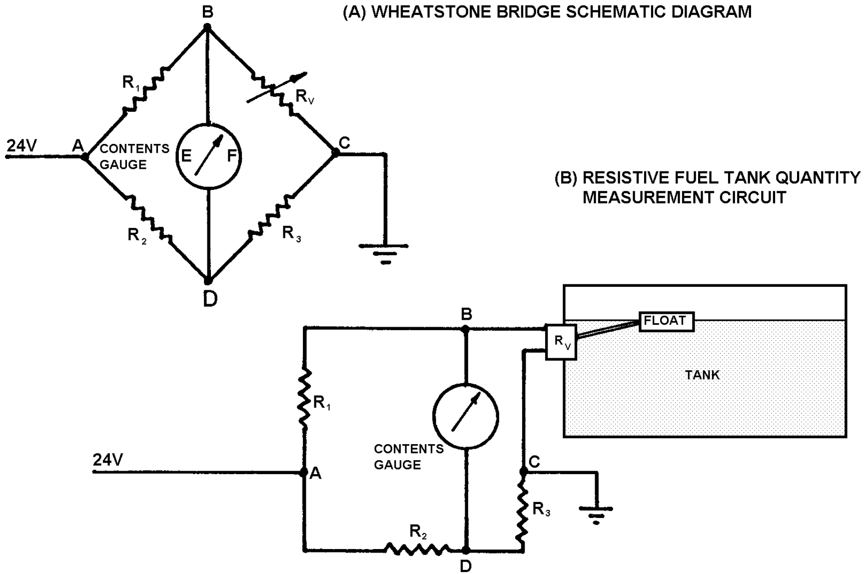
65. In practice a number of capacitive probes, or sensors, are connected in parallel in each fuel tank in order to compensate for fuel movement in the tank due to aircraft attitude change and wing flexure. Each fuel tank indicating system is provided with a parallel circuit carrying a continuous 'empty' signal. In normal circumstances this is suppressed by the amplified tank probe signal. In the event of failure of the indicating signal, the current in the parallel circuit drives the tank contents gauge to the 'empty' position.

66. Operation of the TEST circuit simulates an empty signal in the indicator circuit, again causing the tank contents gauge to read empty if the indicating circuit is functioning correctly. Alternatively, pressing the test button may cause each tank contents gauge to read a given value, as specified in the amplified checklist and the manufacturer's system description.

Resistive

67. Many light aircraft fuel tank quantity indicating systems are of the resistive type. These employ a float-operated variable resistance in the tank, connected in a wheatstone bridge circuit containing reference fixed resistances and the fuel quantity indicator gauge. The principle is illustrated at Figure 6-5.

FIGURE 6-5
Wheatstone Bridge Fuel Quantity Indicator



68. The three fixed resistances (R_1, R_2 and R_3) are each of the same resistive value (say 100 ohms). If the variable resistance is also 100 ohms, we can determine by Ohm's law that the current flow through each side of the bridge circuit (sides ABC and ADC) will be the same. Under these circumstances there will be no current flow through the indicator and it will read zero (tank empty). As the level of fuel in the tank rises and lifts the float, the float-operated variable resistance increases its resistive value. Suppose, for example, at maximum fuel level its resistance has risen to 200 ohms. The current flow through side ABC of the bridge (which contains the variable resistance) will be less than that through side ADC. This will create a voltage difference between points B and D such that the voltage at B will be higher than that at D (it is possible, using Ohm's Law, to show that the voltage difference in this instance would be 4v). This will cause current to flow from B to D, through the indicator, which would be calibrated to show a 'tank full' reading under these circumstances.

69. Resistive fuel quantity indicators are calibrated to correctly indicate tank contents with the aircraft in a level attitude. At other aircraft attitudes the indication provided is obviously inaccurate. Since they indicate tank fuel level, expansion or contraction of the fuel due to temperature changes will result in erroneous readings if the indicator is calibrated for fuel weight rather than volume.

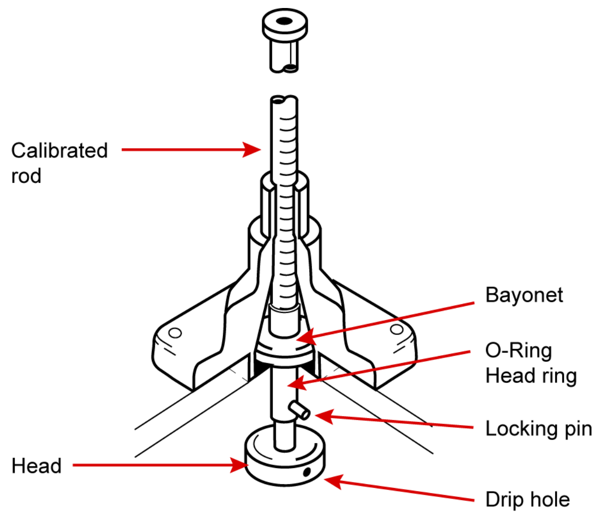
Fuel Measuring Sticks

70. In order to confirm that a quantity of fuel has been uplifted or to obtain the contents in the event of indication system failure, fuel measuring sticks may be used. These may be dipsticks, dripsticks or magnetic level indicators.

71. The dipstick is simply a calibrated stick which is fitted in the top of the tank and when removed will indicate by wetness, the level of fuel in the tank.

72. The dripstick is a calibrated tube which is fitted in the bottom of the tank and when pulled down, will indicate fuel level in the tank when fuel flows through the tube and exits at the drip hole. (Figure 6-6).

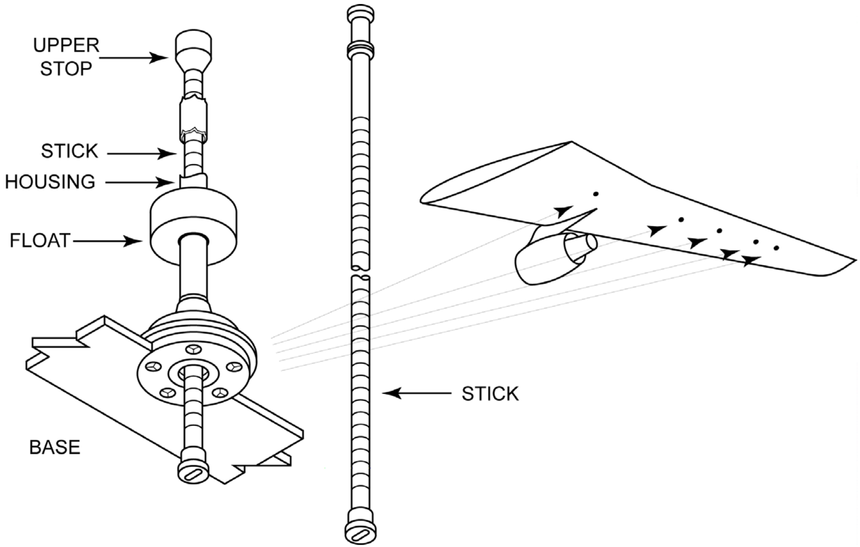
FIGURE 6-6
Fuel Level Dripstick



73. The magnetic level indicator system consists of a housing inside the tank surrounded by a float containing a magnet. The calibrated indicator rod has a magnet at the upper end. When the indicator rod is lowered out of the bottom of the tank, magnetic coupling occurs between the rod and float. The level of fuel in the tank may be read off from calibrations on the indicator rod. (Figure 6-7).

74. Since all the fuel measuring sticks indicate volume of fuel, the stick readings are converted to fuel weight using tables or charts and applying a connection for fuel specific gravity. The aircraft attitude is also required at the time of stick readings i.e. nose up/nose down, pitch and roll attitude. An aircraft spirit level indicator is provided for this purpose.

FIGURE 6-7
Magnetic Level
Indicator



Self Assessed Exercise No. 2

QUESTIONS:

QUESTION 1.

What are the main components in a pneumatic system for a piston engine aircraft.

QUESTION 2.

Identify the aircraft systems which use compressed air as a power source.

QUESTION 3.

Identify the main components of an air conditioning system.

QUESTION 4.

What heating sources are used in air conditioning systems.

QUESTION 5.

How is cabin temperature controlled.

QUESTION 6.

Identify the main components of a pressurisation system.

QUESTION 7.

What is cabin altitude.

QUESTION 8.

What is differential pressure.

QUESTION 9.

What is the maximum cabin altitude under normal conditions.

QUESTION 10.

How does the maximum differential pressure limit maximum operating altitude.

QUESTION 11.

What areas of the fuselage are pressurised.

QUESTION 12.

What is the principle of operation of a cabin pressure altitude control system.

QUESTION 13.

Identify and explain the principle of operation of the safety devices in pressurisation system.

QUESTION 14.

Where on the aircraft are pneumatic de-icing systems located.

QUESTION 15.

Describe the working principle of inflatable rubber boots.

QUESTION 16.

In a pneumatic de-icing system, how is inflation and deflation controlled.

QUESTION 17.

When should the pneumatic de-icing system be operated.

QUESTION 18.

List the power sources which drive pneumatic systems.

QUESTION 19.

What is the purpose of a pneumatic system.

QUESTION 20.

In a pneumatic system what is the function of an isolation valve.

QUESTION 21.

What is the function of the pressure regulating valve.

QUESTION 22.

What is the function of the system bleed valve.

QUESTION 23.

State how the pneumatic system is controlled and monitored.

QUESTION 24.

What are two possible pneumatic duct failures that can occur.

QUESTION 25.

What is the purpose of an air conditioning system.

QUESTION 26.

Explain how air temperature is controlled.

QUESTION 27.

What is the function of the ram air valve in the air conditioning system.

QUESTION 28.

What is the function of the re-circulation fan.

QUESTION 29.

Describe the use of hot trim air.

QUESTION 30.

Describe the construction of the 3 types of fuel tanks.

QUESTION 31.

What is the function of the fuel tank baffles.

QUESTION 32.

What is the function of an overpressure relief valve.

QUESTION 33.

What is the function of the surge vent tank.

QUESTION 34.

What is the refuelling sequence for tanks that are only to be partially filled.

QUESTION 35.

What is meant by unusable fuel.

QUESTION 36.

Describe the various methods of refuelling.

QUESTION 37.

What is the operating principle and function of a fuel screen.

QUESTION 38.

What is the operating principle and function of the cross feed valve.

QUESTION 39.

What method is used to control the temperature of the fuel feed to the engine.

QUESTION 40.

Describe the typical controls of a fuel system.

ANSWERS:

ANSWER 1.

Engine driven compressor.

Pressure regulator.

Main storage bottle.

Pressure reducing valve.

Oil and water trap.

Indication and warning.

ANSWER 2.

Landing gear, main and alternate.

Wheel Brakes.

Radiator shutters.

Aerofoil de-icing.

ANSWER 3.

Ram air inlet.

Heat exchanger.

Refrigeration unit.

Water separator.

ANSWER 4.

Exhaust heater muff-Ram air.

Combustion heater-Ram air.

Engine driven blower.

ANSWER 5.

Cabin air temperature is controlled by mixing the hot air with cold air in response to the temperature controller.

ANSWER 6.

Pressurisation controller.

Outflow (Discharge) valves.

Safety valves.

Inward relief valves.

Dump valve.

Ditching Control.

ANSWER 7.

Cabin altitude is the altitude in the pressure cabin corresponding to the prevailing cabin air pressure.

ANSWER 8.

Differential pressure is the difference in air pressure between the cabin pressure compared to the aircraft ambient pressure. It is a positive value when the cabin pressure is higher than ambient and negative value when the ambient is higher than cabin pressure.

ANSWER 9.

10,00ft

ANSWER 10.

The pressurisation controller will limit the differential to a maximum value and because maximum cabin altitude is also limited it means that the aircraft operating altitude will be limited.

ANSWER 11.

Passenger Cabin and Flight Deck

Cargo Holds

ANSWER 12.

Air entering the pressure hull is mass flow controlled and an automatic pressurisation controller governs the amount of air leaving the hull by positioning the outflow valve(s).

ANSWER 13.

Safety valves. Relieve pressure from the pressure cabin when maximum design structural differential pressure is reached.
Inward relief valves. Equalise pressure between ambient and cabin should ambient pressure exceed cabin pressure.

ANSWER 14.

On the leading edges of the mainplane, fin and tailplane.

ANSWER 15.

Rubberised fabric tubes are inflated and deflated cyclically with air and vacuum respectively which will break up the ice layers above them.

ANSWER 16.

Inflation and deflation is controlled by electrically operated valves on a cyclic time sequence allowing pressure or vacuum into the inflatable tubes.

ANSWER 17.

It should be switched on when the recommended depth of ice has formed on the aerofoil surfaces.

ANSWER 18.

Engine compressor bleed air

Engine driven blower

APU bleed air

Ground air supply

ANSWER 19.

Air conditioning and pressurisation.

Engine starting.

Thermal anti-icing.

Pressurising hydraulic reservoir and potable water supply.

Leading edge flaps.

Emergency lowering of landing gear.

ANSWER 20.

To shut off the engine bleed air supply from compressor to system.

ANSWER 21.

To control the designed pneumatic system pressure.

ANSWER 22.

To admit air from the appropriate compressor stage into the pneumatic system.

ANSWER 23.

It is controlled by the pressure regulating valve which is active once the appropriate bleed air selection has been made on the flight deck. The system is monitored by reference to duct pressure gauges on the pneumatic system display with warning lights indicating pneumatic duct leaks.

ANSWER 24.

OVER PRESSURE. This could cause duct leakage of hot air.
MECHANICAL. Duct joints can become loose or detached due to vibration or overflexing causing hot air to leak from the duct.

ANSWER 25.

To maintain an adequate supply of air for ventilation and pressurisation at a comfortable temperature and humidity.

ANSWER 26.

The temperature of the air supply to the cabin is controlled by mixing hot and cold air in variable proportions to maintain the cabin air temperature within prescribed limits.

ANSWER 27.

To provide a means of admitting ram air for ventilation in the event that there is total loss of air conditioning.

ANSWER 28.

To reduce the air conditioning load, by recirculating some cabin air back to the mix manifold, thus supplementing incoming fresh air.

ANSWER 29.

Individual zone temperature requirements are satisfied by adding hot trim air to the affected zone.

ANSWER 30.

Rigid-Generally constructed of Aluminium Alloy. B. Flexible-Constructed from re-inforced rubber or plastic sheeting in a non rigid form. C. Integral-Built into the aircraft structure. Can be in the lower centre fuselage or commonly in the wing.

ANSWER 31.

To prevent fuel surging in the tank during aircraft manoeuvres.

ANSWER 32.

Where the area above a fuel tank is pressurised a relief valve is fitted to relieve excess pressure. Located in the vent system it will relieve any positive or negative at all aircraft attitudes.

ANSWER 33.

To collect any fuel which flows into the vent system and pump it back into the main tank system.

ANSWER 34.

The sequence of refuelling is to commence refuelling the inboard tanks to the required level according to requirements and continue outboard for all remaining tanks.

ANSWER 35.

A quantity of fuel at the bottom of the tanks which cannot be removed by pumping. It is determined by the shape or position of the tank.

ANSWER 36.

OVERWING. A method whereby a tank is refuelled from an external source by placing a fuel nozzle directly into the tank filler opening. PRESSURE. Fuel under pressure from an external source is pumped into the fuel tank via a connection at the refuel panel.

ANSWER 37.

A fuel screen is a filter which is required to prevent foreign matter contamination of the fuel system. Eg. A fuel filter is fitted between the fuel tank and the inlet to the pump.

ANSWER 38.

A cross feed valve will give flexibility of fuel management and the ability to maintain fuel balance between tanks. It allows any tank to supply fuel to any or all engines.

ANSWER 39.

Prior to entering the main fuel filter and engine fuel control, the fuel is maintained at a minimum temperature above zero degrees centigrade. The fuel is passed through a heat exchanger using a medium of hot bleed air or oil to warm the fuel.

ANSWER 40.

Typical controls consist of; Fuel system shutoff valves. Booster Pump switching. Fuel heater selection. Fuel tank transfer valves switching.

Electrics-DC

Electrics-DC

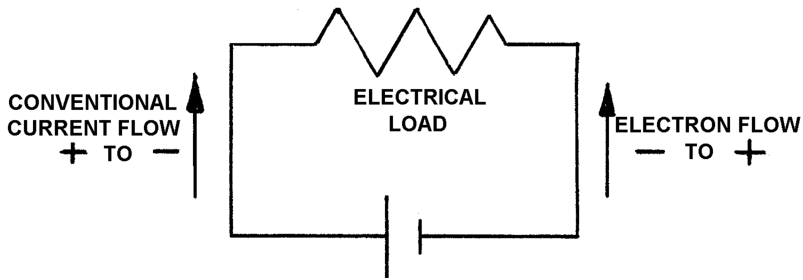
Basic Principles of DC Electricity

1. All matter is made up of atoms, which consist of a positively charged nucleus and negatively charged orbiting electrons. The nucleus is formed by an equal number of protons and neutrons, the protons being positively charged and the neutrons having no charge. An oxygen atom, for example, comprises eight electrons (negative), eight protons (positive) and eight neutrons (no charge). Hence such an atom is electrically neutral, since the total negative charge of the electrons is balanced by the total positive charge of the protons.

2. An electrical current is quite simply a flow of electrons through a conductor. When matter gives up electrons it becomes positively charged, and when it gains electrons it becomes negatively charged.

3. During the early experiments with electricity the true nature of electrical current flow was not fully understood and the direction of current flow was arbitrarily assumed to be from positive to negative, as illustrated at Figure 7-1. This convention with regard to current flow has persisted, although it was subsequently proved that a current flow involves the transference of negatively charged electrons to a positively charged atom (that is, an atom which has a deficiency of electrons). For the sake of conformity, these Notes will maintain the convention of current flowing from positive to negative, preferably through a load or resistance, which will limit the rate of current flow.

FIGURE 7-1
Direction of
Current Flow



4. A substance consisting of atoms that readily give up their electrons is called a conductor, and will offer little resistance to a flow of electricity. Copper and Gold are both particularly good conductors. Conversely, a substance consisting of atoms that are reluctant to release electrons is called an insulator, because it offers a very high resistance to the flow of electrical current. Insulators are used to protect conductors, insulating them from other conductors. For example, electrical current-carrying cables are covered with insulating material if they are to be located where other conductors, such as the metal case of a device or a human being, might otherwise come into contact with them. Natural rubber and most thermoplastic materials are good insulators. Polyvinyl chloride (PVC) is a well-known example. A semiconductor is a material whose resistance is midway between that of a good conductor and that of a good insulator. Commonly used semiconductor materials include silicon and germanium. (diodes, transistors and integrated circuits.

5. Static electricity occurs when an excess of electrons are induced into a body, which then stores this electrical charge until able to discharge itself to another body. A good example of this occurs with an aircraft in flight. Since the airframe is used as the earth for all of the aircraft electrical systems it develops its own static electrical potential. An attempt is made to discharge this potential during flight to atmosphere via static wick dischargers. On the ground it is discharged to the tarmac via a conducting bead in the tread of the tyres (usually on the nosewheels).

6. It is because of static electricity that it is necessary to bond an aircraft and a fuel bowser to each other during refuelling, and to bond the fuel pipe nozzle to the tank filler pipe. Were this not done, static electricity could build up and cause a spark with obviously disastrous consequences. For the same reasons it is necessary to bond together the various parts of the aircraft structure to provide a low resistance path for static discharge and to dissipate the effects of a lightning strike.

7. Electricity may be produced by heat or temperature differences, as in thermocouples; by friction, as in static electricity; by light, as in photo-electric cells; by pressure, as in carbon microphones; by rotating mechanical forces in magnetic fields, as in generators; and by chemical action, as in batteries.

Definitions

8. As already stated, an electric current is simply a flow of electrons through a conducting element and it is measured in amperes (amps) by means of an ammeter. The lower the resistance to current flow, the greater the current flow and vice versa. Maximum current will flow when a short circuit exists (a direct connection between supply and return), and this causes an overload. No current will flow when an open circuit exists (when the circuit is broken, for example by opening a switch). The break in the circuit has created a condition of infinite resistance.

9. The basic units of electrical measurement are often inconvenient in that they are either too small or too large for practical circumstances and consequently prefixes are added. For example;

Prefix	Meaning	Examples
Micro	One millionth (10^{-6})	Microvolt, microamp
Milli	One thousandth (10^{-3})	Millivolt, milliamp
Kilo	One thousand (10^{3})	Kilowatt, kilovolt
Mega	One million (10^{6})	Megohm, megawatt

10. Potential difference (PD) is the way in which the magnitude of the surplus of electrons at one point in an electrical circuit, when compared with the deficiency of electrons at another point is expressed. Potential difference is often referred to as electrical pressure, and is measured in volts using a voltmeter.

11. Electromotive force (emf) is the force that causes electrons to flow, and is effectively the same as potential difference. The unit of measurement of emf is the volt. One volt is defined as the emf required to cause current to flow at the rate of one ampere through a resistance of one ohm.

12. Resistance is the reluctance of any material, electrical component or electrical machine to permit the flow of electricity. Resistance is measured in ohms using an ohmmeter. The usual consequence of a circuit offering resistance to current flow is the generation of heat.

13. The resistance of a conductor is directly proportional to temperature and to conductor length and is inversely proportional to the cross-sectional area of the conductor. As the temperature of a conductor rises the atoms gain energy and they become 'excited'. Any electron flow in the circuit will now experience difficulty in moving through the conductor. That is, increase in temperature causes an increase in conductor resistance. The change of resistance of the conductor as the temperature changes expressed as a fraction of its original resistance is called the temperature co-efficient of resistance referred to the original temperature. If the resistance of a conductor increases with increase in temperature it has a positive temperature co-efficient and a negative co-efficient if the resistance decreases with increase in temperature. Positive temperature co-efficient (PTC) and negative temperature co-efficient (NTC) resistors are used in aircraft systems for measurement of temperature. An example of this is a thermistor which is a thermally sensitive resistor with either a positive or negative temperature co-efficient.

14. For electrical current to flow there must be a path, or circuit, for it to follow and there must be a potential difference (voltage/pressure) to cause it to flow. The amount of current flow will depend upon the resistance of the circuit. The unit of measurement of current flow is the ampere, usually abbreviated to amp. One ampere is defined as a rate of current flow of one coulomb per second, the coulomb being the unit of measurement of electrical quantity. Current, or rate of flow of electricity, is symbolised by the letter I.

Ohm's Law

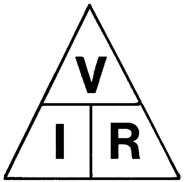
15. Ohm's Law gives the relationship between emf (V), current flow (I) and resistance (R). It states that the current in an electrical circuit is directly proportional to the voltage (emf) and inversely proportional to the resistance.

16. In other words, an emf of 1 volt will cause a current of 1 amp to flow through a circuit having a total resistance of 1 ohm. Equally, an emf of 25,000 volts will cause a current of 10 amps to flow through a circuit having a total resistance of 2,500 ohms.

The equation for Ohm's Law is: $I = \frac{V}{R}$

This is usefully given diagrammatically as shown at Figure 7-2.

FIGURE 7-2
Ohms Law



17. This simplifies transposition of the Ohm's Law equation viz:

$$V = I \times R$$

$$I = \frac{V}{R}$$

$$R = \frac{V}{I}$$

18. Power is defined as the rate of doing work or, to put it another way, the rate of dissipation of energy. Since voltage is the measure of electromotive force and amperage is the measure of rate of current flow it follows that electrical power is the product of the two. The unit of measurement of power is the watt and is defined as the power expended when one volt moves a quantity of one coulomb per second through a conductor.

$$1 \text{ Watt} = 1 \text{ Volt x } 1 \text{ Amp}$$

19. By combining the power equation above with the Ohm's Law equation it is possible to derive alternative power equations.

$$\text{Power (W)} = V \times I$$

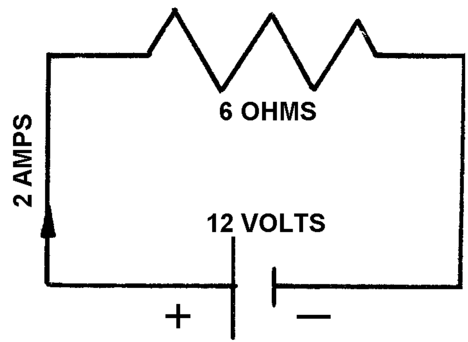
and

$$\text{Since } V = I \times R \text{ it follows that } W = (I \times R) \times I = I^2 R$$

$$\text{Since } I = \frac{V}{R} \text{ it follows that } W = V \times \left(\frac{V}{R}\right) = \frac{V^2}{R}$$

20. Electrical power is the rate at which work is done when electrical energy is expended. Electrical work is a measure of the amount of electrical energy expended and is measured in units called Joules. 1 Joule represents the work done by 1 watt of power in 1 second

FIGURE 7-3
Electrical Power = Watts



21. At Figure 7-3 a current of 2 amps is flowing through a resistance of 6 ohms, thanks to an emf produced by the battery of 12 volts (V = I x R). The power consumed in the circuit is therefore:

$$V \times I = 12 \times 2 = 24 \text{ Watts}$$

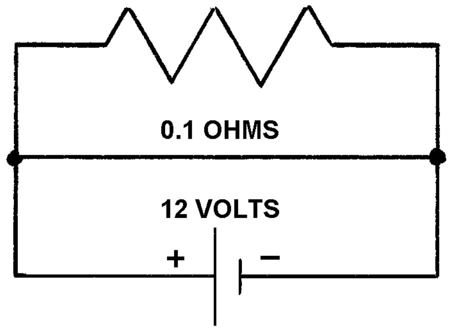
or

$$I^2R = 4 \times 6 = 24 \text{ Watts}$$

or

$$\frac{V^2}{R} = \frac{144}{6} = 24 \text{ Watts}$$

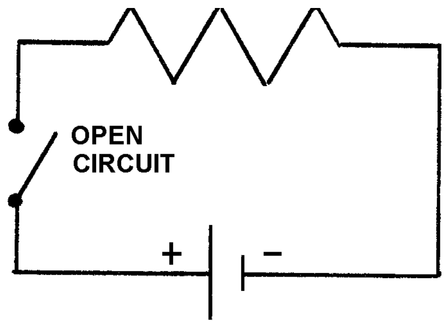
FIGURE 7-4
Short Circuit



22. At Figure 7-4 the battery has been short-circuited across its terminals by a conductor of low resistance (0.1 ohms). The current flowing through the short circuit will be:

$$I = \frac{V}{R} = \frac{12}{0.1} = 120 \text{ amps (an overload)}$$

FIGURE 7-5
Open Circuit



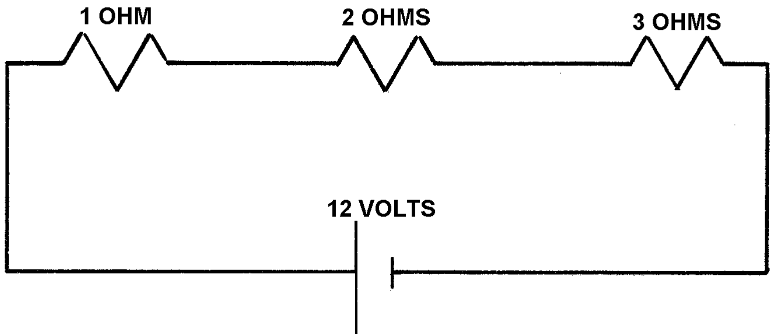
23. At Figure 7-5 the circuit has been open-circuited by a switch, introducing an infinite resistance. Current flow will now be:

$$I = \frac{V}{R} = \frac{12}{\infty} = 0 \text{ amps}$$

Series Loads

24. Up to this point the simple circuit diagrams have shown only one load, or resistance, for simplicity. Practical electrical circuits usually contain several separate loads, such as a number of light bulbs, for example. One method of connecting a number of loads is to place them in series with one another, as shown at Figure 7-6.

FIGURE 7-6
Series Loads



25. When loads are connected in series, since the current must flow through each load or resistance sequentially the total resistance (R_T) of the circuit is the sum of all the resistances:

$$R_T = R_1 + R_2 + R_3$$

26. Applying Ohm's Law to the circuit:

$$I = \frac{V}{R} = \frac{12}{6} = 2 \text{ amps}$$

27. Taking this further, the voltage drop across the circuit as a whole must equal the sum of the voltage drops across each resistance. Viz:

$$R_1(1\text{ohm})V = IR = 2 \times 1 = 2 \text{ volts}$$

$$R_2(2ohm)V = IR = 2 \times 2 = 4 \text{ volts}$$

$$R_3(3ohm)V = IR = 2 \times 3 = 6 \text{ volts}$$

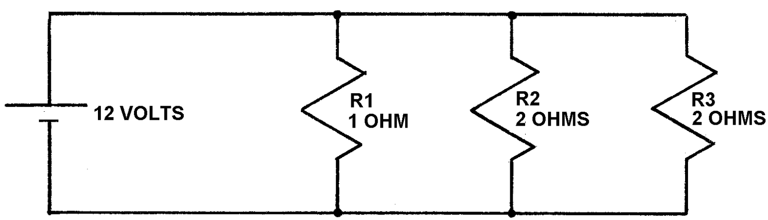
Total 12 volts

28. Circuits in which the loads are connected in series have few practical applications since it is impossible to switch off supply to one load without affecting the entire circuit. Similarly, if one load fails causing an open circuit, the entire circuit fails. Consequently such circuits are rarely used in aircraft systems.

Parallel Loads

29. Figure 7-7 shows a more conventional electrical circuit in which the loads are connected in parallel with each other and with the source of emf. In such a circuit each load may be individually disconnected without disrupting the flow of electricity to the remaining loads.

FIGURE 7-7
Parallel Loads



30. In order to determine the equivalent total resistance of this circuit it is necessary to use the formula:

$$\frac{1}{R\ total} = \frac{1}{R_1} + \frac{1}{R_2} + \frac{1}{R_3} \text{ and so on}$$

31. In a parallel circuit the total resistance is always less than the smallest individual resistance in the circuit.

32. In this case the total resistance is:

$$\frac{1}{R\ total} = \frac{1}{1} + \frac{1}{2} + \frac{1}{2}$$

$$\frac{1}{R\ total} = \frac{1}{1} + 0.5 + 0.5 = 2$$

$$R\ total = \frac{1}{2} = 0.5 \text{ ohm}$$

33. The total current flow in a parallel circuit is found using Ohm's Law:

$$I_T = \frac{V}{R_T} = \frac{12}{0.5} = 24 \text{ amp}$$

34. Ohm's law may be used to find the current flowing through each load in the normal way, and the current drawn from the battery will be the sum of the currents flowing in each load.

$$\text{In } R_1; I = \frac{V}{R_1} = \frac{12}{1} = 12 \text{ amps}$$

$$\text{In } R_2 \text{ and } R_3; I = \frac{12}{2} = 6 \text{ amps in each}$$

$$I_T = 12 + 6 + 6 = 24 \text{ amp}$$

35. The voltage drop across each load, or resistor in a parallel circuit is always the same as the supply emf (in this case 12 volts). This too can be demonstrated using Ohm's Law.

$$R_1(1\text{ohm})V = IR = 12 \times 1 = 12 \text{ volts}$$

$$R_2 \text{ and } R_3(2\text{ohms})V = IR = 6 \times 2 = 12 \text{ volts across each}$$

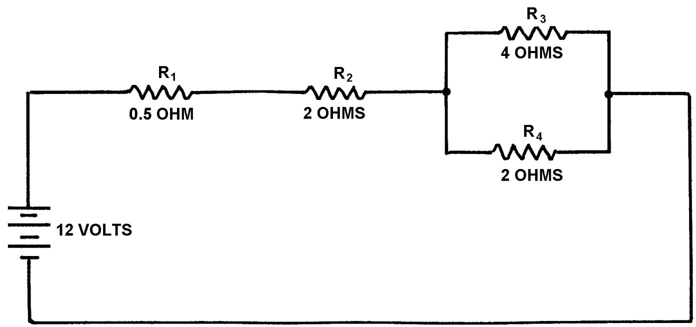
36. When only two resistances are connected in parallel an alternative formula, derived from the general formula for the total resistance of a parallel circuit, may be used. This is given as:

$$R_T = \frac{R_1 \times R_2}{R_1 + R_2}$$

Series/Parallel Loads

37. A series/parallel circuit is one in which some of the loads are connected in series with the source of supply and some in parallel with it. An example of such a circuit is shown at Figure 7-8.

FIGURE 7-8
Series / Parallel Loads



38. The total resistance (R_T) of this circuit may be found by resolving it into the equivalent of three resistances in series and adding them together. To do this, it is first necessary to find the total resistance of R_3 and R_4, the two paralleled loads, and add the result to the sum of the series resistances.

$$R_{(3,4)} = \frac{R_3 \times R_4}{R_3 + R_4} = \frac{4 \times 2}{4 + 2} = \frac{8}{6} = 1.33 \text{ ohms}$$

The total resistance of the circuit (R_T) is given by:

$$R_1 + R_2 + R_{(3,4)} = 0.5 + 2 + 1.33 = 3.83 \text{ ohms}$$

Circuit Protection

39. It is essential that any electrical circuit should be protected against current flows that are in excess of the maximum rated value of the circuit in question. Apart from damage to the circuit components, a short circuit to the airframe (for example) would cause a very high current flow which would generate a great deal of heat and pose a serious fire hazard. The standard safety devices take the form of fuses and circuit breakers.

Fuses

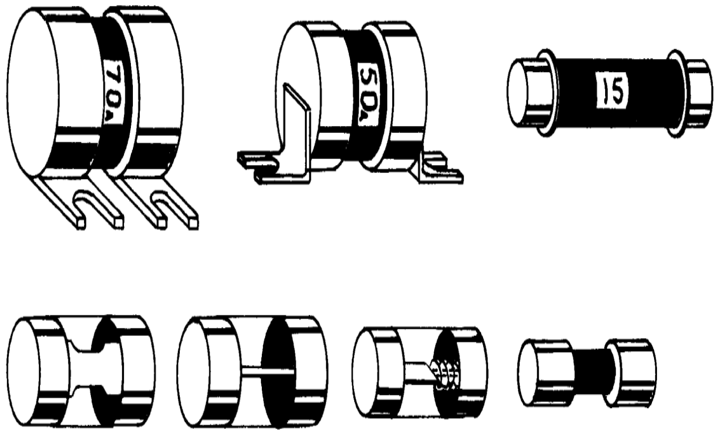
40. Fuses are thermal devices whose primary function is to protect the distribution cables of a circuit against excess current flow due to short-circuit or overload. The fuse is placed in series with the load (component) it protects. If a current that exceeds the rated value of the fuse flows through it, the fuse wire overheats and melts, resulting in an open circuit. The fuse wire is normally made of a zinc alloy, which has the desired low melting point.

41. All fuses are rated in amperes (amps). In general fuses are selected on the basis of the lowest rating which will ensure reliable operation of the system, given the known thermal characteristics of the cables, but which will not be prone to 'nuisance failure'.

42. In emergency circuits, failure of which may affect safe operation of the aircraft, fuses are of the highest rating possible consistent with cable protection. When replacing a blown fuse it is important that the new fuse be of the correct rating. Under no circumstances should a fuse of a higher value be used, since the rating has already been calculated at a value slightly in excess of the normal maximum load of the circuit, to accommodate the surge loads of initial switching on.

43. Aircraft fuses are of the cartridge variety, consisting of a ceramic tube with the fusible element (wire) passing through it and connected at either end to conducting caps. The caps fit into the inlet and outlet terminals of the fuse holder. Examples of typical aircraft cartridge type fuses are shown at Figure 7-9.

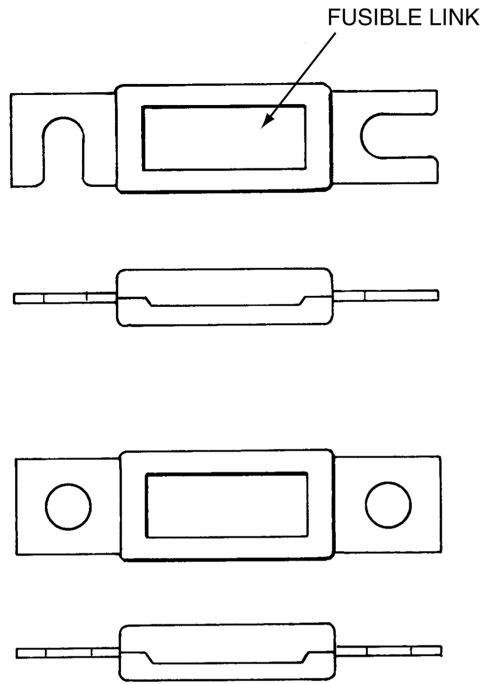
FIGURE 7-9
Cartridge Fuses



44. Heavy Duty (High Rupturing) fuses are often installed at power distribution points. These are of the cartridge type, but contain a number of fusible elements that rupture successively under overload conditions. The elements are packed in a medium that absorbs the explosive effects of rupture. These fuses are fixed in position by means of mounting lugs and bolts, as opposed to the spring-clip attachment of lighter-duty fuses. In some aircraft the fuse holders incorporate an indicator lamp which illuminates when the fuse ruptures.

45. Current Limiting Fuses are also used to protect high power, heavy-duty, circuits. They have a high melting point and will thus carry a considerable overload current before rupturing. This ensures that power to the whole circuit is not lost in the event of a surge, or transient, overload. The fusible element is typically a strip of tinned copper, "waisted" in its centre to the dimension required for the fusing area. The ends of the strip are formed into attachment lugs. The central, waisted, portion of the fuse is enclosed in a ceramic housing with an inspection window. Examples are shown at Figure 7-10.

FIGURE 7-10
Current Limiting
Fuses



46. After a fuse has first blown it should be replaced once only. Repeated replacement of the fuse could eventually result in overheating of the circuit.

47. When replacing fuses, ideally the power to the circuit in question should be switched off. The person replacing the fuse does not then have to worry about touching the live terminals of the fuse holder.

48. If a fuse of the correct rating is not available, then a fuse of a lower rating may be tried, but of course it is likely to blow again.

49. The number of spare fuses to be carried in an aircraft is stipulated by Joint Aviation Requirements.

50. Spare fuses for all electrical circuits, the fuses of which can be replaced in flight, consisting of 50% of the number of each rating.

Circuit Breakers

51. A circuit breaker or thermal trip is designed to isolate a circuit by means of a mechanical trip that opens a switch whenever a surge of current, or overload, occurs. The advantage of a circuit breaker over a fuse is that a circuit breaker can be reset once the overload situation has been remedied. Circuit breakers make use of bimetallic strips, which bend by an increasing amount as the temperature of the strip increases. At the temperature matched to the rated current flow for that particular circuit the bimetallic strip bends sufficiently to break the circuit.

52. The linkage between the bimetal element and the trip mechanism can be adjusted at manufacture to achieve very close tolerance 'trip-time' characteristics. Thus, the circuit breaker can be matched not just to current, but to a specific maximum time for a given current. The ratings for circuit breakers are established in much the same way as for fuses.

53. There are two types of circuit breaker, the trip-free and the non trip-free. In the trip-free system depressing the reset button will not remake the circuit, nor will it reset the circuit breaker, until the overload condition has been cleared. It has an internal safety device that ensures that no harm will be caused if the reset button is held in.

54. With a non trip-free circuit breaker it is possible to remake the circuit by holding the button in. Once the button is released the circuit will be broken again, since the circuit breaker cannot be reset until the overload has been cleared. Some circuit breakers are equipped with manual trip buttons so that they can be also used as a manually operated switching device.

55. In the past, non trip-free circuit breakers were installed in some essential service circuits to permit emergency manual reconnection of supply under overload conditions, despite the fire hazard involved. Nowadays this practice is not permitted by the CAA, who require that it must not be possible for circuit breaker contacts to be re-made under overload conditions. Figure 7-11 shows a schematic diagram of a non trip-free thermal circuit breaker operation.

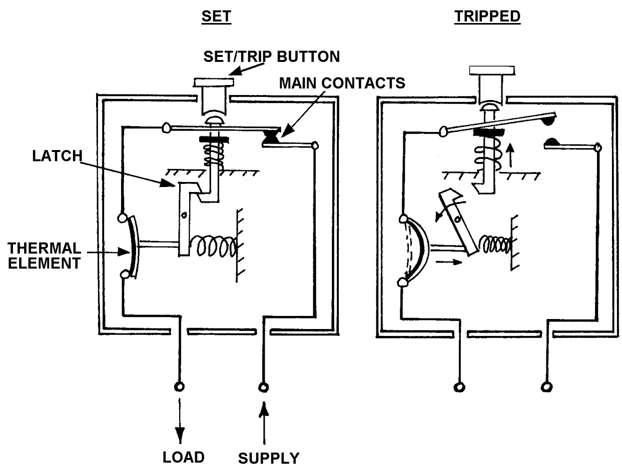


FIGURE 7-11
Thermal Circuit Breaker

Switches

56. A basic switch consists of two contacting surfaces which can be connected together by a link or isolated by the reverse action (open the link). The connecting link is commonly referred to as a pole and there are various combinations i.e. single pole, single throw or double pole, double throw. There are a number of different type switches used in aircraft electrical systems namely push, toggle, thermal, pressure, mercury, time, rotary, micro and proximity switches. Most of these switches are explained in the following text apart from the following :

(a) Toggle Switch. A tumbler type switch which is regarded as a general purpose switch. A guard cap can be incorporated which is a device to prevent accidental operation of the switch. The guard has to be physically lifted before movement of the switch is possible.

(b) Thermal Switch. The priciple of operation is based on the effects of differences of expansion between dissimilar metals typically INVAR and steel. The different heat sensitivity can be used where automatic temperature control is required, typically the operation of control valves in thermal de-icing systems.

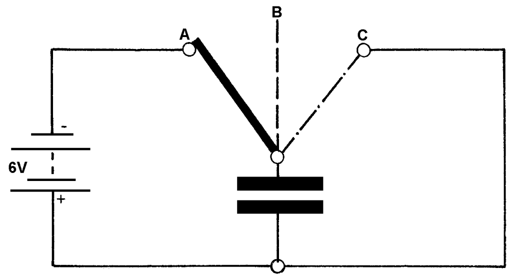
(c) Time Switch. The principle of operation varies, but in general it is based on an electric motor driving a cam assembly which in turn contacts microswitches or operates spring driven mechanisms. This will produce a programmed sequence of operations at limited intervals.

The Capacitor

57. Capacitors are electrical storage devices which, in a DC circuit, behave in some ways like a battery. They will accept and store an electrical charge if voltage is applied in one direction. A fully charged capacitor will discharge, in the opposite direction to the original charging supply, when the charging supply ceases and a suitable circuit is provided.

58. A capacitor comprises two plates of conducting material, separated by a non-conducting material called a dielectric. Figure 7-12 shows a capacitor in parallel with a 6v DC source. If the 3-position switch is placed in position A, the potential difference across the capacitor is the same as that across the battery (6v).

FIGURE 7-12
Capacitor Operation



59. This is because the plate connected to the negative source has an excess of electrons whilst the plate connected to the positive source has a deficiency of electrons. In other words, one plate is negatively charged and the other is positively charged. The result of this is to put the dielectric under 'strain' establishing an electric field between the two plates. Comparison can be made with a magnetic field which has lines of force with definite direction and strength. However the electrical field is achieved as a result of opposing positive and negative polarities, whereas a magnetic field is achieved naturally from a permanent magnet or by passing a current through a conductor.

60. If the switch is now moved to position B, the capacitor will store its electrical charge, there being no circuit through which it can discharge. The ability of a capacitor to store an electric charge is called capacitance and is measured in units known as farads. The capacitance of a given capacitor is determined by three factors, the area of the capacitor plates, the thickness of the dielectric and the material of which the dielectric consists. The dielectric qualities of a material are measured against the dielectric property of air, which is given a constant value of unity (1).

61. A material that has five times the electric charge storing ability of air in a capacitor is said to have a dielectric constant of 5. A material commonly used is mica, which has a dielectric constant of 5.5.

62. When the switch in the circuit at Figure 7-12 is moved to position C, the capacitor will discharge its stored charge through the circuit that is made. Capacitors are discussed in depth in the chapter on AC Electrics.

63. When cacitors are connected in series the total capacitance is given by the formula:

$$\frac{1}{C_{TOTAL}} = \frac{1}{C_1} + \frac{1}{C_2} + \frac{1}{C_3} + \frac{1}{C_4} \text{ and so on}$$

When capacitors are connected in parallel the total capacitance is given by the formula:

$$C_{TOTAL} = C_1 + C_2 + C_3 + C_4 \text{ and so on}$$

Batteries

64. Chemical action produces electricity in an electric cell. A single cell, or more commonly a number of cells connected together, is known as a battery. All battery cells produce direct current, because they are of constant electrical polarity.

65. A cell that cannot be recharged once it has become discharged is known as a primary cell. Conversely, a cell that is rechargeable is known as a secondary cell.

66. The principle of any electric cell is simple. The cell consists of two plates made of different metals and placed in a solution termed the electrolyte. An electrolyte is a solution of water and a chemical compound that will conduct electricity. It is capable of conducting current because it contains atoms having a positive or negative charge, known as ions.

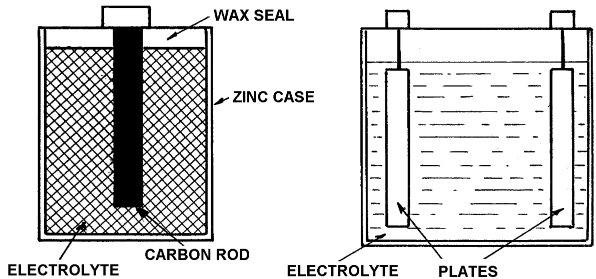
67. The chemical action of the electrolyte acting on the dissimilar plates causes an electron flow from one plate to the other. Consequently, one plate becomes positively charged (a deficiency of electrons) and the other negatively charged (a surplus of electrons). There is thus a potential difference (voltage) between the two plates.

68. Connecting the two plates via a conductor will allow electrons to flow from the negatively charged plate to the positive – an electrical current flow. This will eventually balance the charge between the plates, so that there is no potential difference and current flow ceases – the cell is said to be discharged and a reversal of the chemical action will be required to restore the differential (or electrical pressure).

Primary Cells

69. In the case of the conventional primary dry cell the positive core is made of carbon, the negative case is made of zinc, and the electrolyte is ammonium chloride in paste form, hence the name 'dry' cell. Dry cells are mainly used in flashlights and similar low voltage portable devices. Diagrams of a simple primary and secondary cell are shown at Figure 7-13.

FIGURE 7-13
Primary & Secondary Cell



Secondary Cells

70. The principal function of the battery or batteries in an aircraft electrical system is to provide electrical power when the primary source of electrical power, the engine-driven generators, is not available. A basic electrical supply system for a single engine aircraft is shown at Figure 7-14.

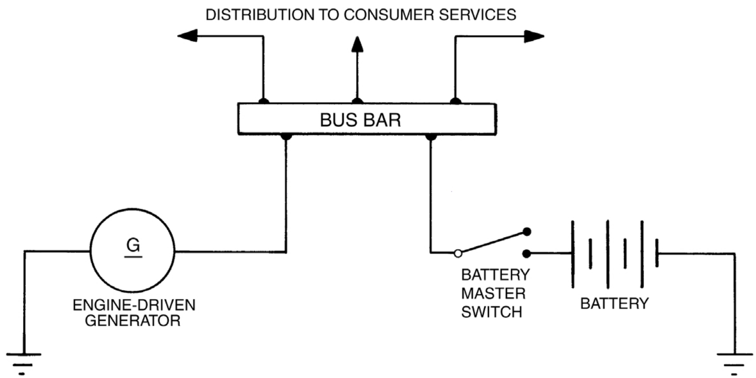


FIGURE 7-14
Basic Aircraft Electrical Supply System

71. Secondary cells are clearly necessary when a battery is to be used as the emergency source of electrical power, as is the case in all aircraft electrical systems. It must be possible to recharge the battery after use, from the aircraft's primary electrical source (the generators) so that it is available when subsequently required. Two types of secondary cell batteries are currently in common use in aircraft, the lead-acid battery and the Nickel-Cadmium battery.

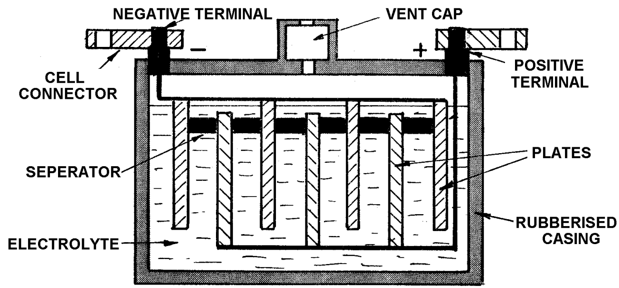
Lead-Acid Batteries

72. Secondary cells are usually grouped together to form a battery of cells, known as a storage battery. In the case of the lead-acid battery this consists of positive plates made of lead peroxide and negative plates made of lead. Each pair of positive and negative plates, comprising a cell, is connected in series with the next and the whole immersed in an electrolyte of 30% sulphuric acid and 70% water.

73. As the battery is discharged the chemical action changes both plates to lead sulphate, and the strength of the electrolyte is reduced. As a consequence the specific gravity of the electrolyte also reduces. As the battery accepts a charge the plates revert to their original compositions, and the acid of the electrolyte strengthens. This results in a higher electrolyte specific gravity.

74. In order to optimise the performance of a lead-acid cell, the plates of dissimilar metals described previously are sandwiched alternately, as shown at Figure 7-15, to present the greatest possible surface area for a given volume of electrolyte. Separators, made of insulating material, are fitted between the positive and negative plates to prevent internal short-circuiting.

FIGURE 7-15
Secondary Cell (Lead Acid) Construction



75. A lead-acid cell will produce a maximum of 2.1 volts when in a fully charged state. It is therefore necessary to connect 6 such cells together in series to produce a battery that has a nominal charge of 12 volts, since cell voltage falls to approximately 2 volts when connected to a substantial load. Aircraft batteries of the lead acid type are usually rated at 12v (6 cells in series) or 24v (12 cells in series).

76. Batteries are rated according to their voltage and capacity. Battery capacity is measured in terms of ampere-hours at a five-hour discharge rate; that is to say the battery was discharged to zero voltage in five hours to determine its capacity. Battery capacity is checked every 3 months, if it falls below 80% of its rated capacity the battery is removed from aircraft service.

77. A 12 volt battery with a capacity rating of 30 ampere-hours (Ah) is capable of supplying 30 amps for 1 hour, 3 amps for 10 hours or any multiple thereof. An alternative capacity rating is often applied to aircraft storage batteries, known as the 5-minute discharge rate. This is based upon the maximum current a battery will supply over a 5-minute period at an initial temperature of 26.7°C and a final average cell voltage of 1.2 volts. This provides a useful indication of a lead-acid battery's engine starting performance.

78. The health of a fully charged battery is checked before it is installed by placing a rated, or known, load across the battery terminals in parallel with a voltmeter. The voltmeter should show the rated voltage of the battery, and should continue to do so for 15 seconds. If it fails to do so this is a good indication that the internal condition of the battery has deteriorated and it should not be placed in service.

79. During charging some of the water in the electrolyte solution tends to break up into its constituent elements, hydrogen and oxygen, which are given off in gaseous form from vents in the top of the battery casing. Hence the proportion of water in the electrolyte solution diminishes and the specific gravity of the solution increases. Hence a battery requires topping up with water from time to time.

80. It is possible to determine the state of charge of a lead-acid cell by using a hydrometer to check the specific gravity (SG) of the electrolyte. When fully charged the SG should be between 1.25 and 1.30, depending upon the age and condition of the cell. When fully discharged the SG is likely to have fallen to approximately 1.17. 1.275 to 1.3 indicates a high state of charge; 1.24 to 1.275 indicates a medium charge state; 1.2 to 1.24 indicates a low charge state.

81. A hydrometer check is not usually practical on an installed battery, because of the risk of acid spillage and resultant airframe corrosion. An alternative way of checking the state of charge of an installed battery is by checking the battery open-circuit voltage (OCV) and closed-circuit voltage (CCV), in other words checking the battery voltage off-load and on load.

82. Off-load voltage (OCV) is virtually unaffected by battery state of charge until the battery is almost fully discharged, but CCV will fall significantly as state of charge diminishes.

83. Batteries are recharged by connecting them to a source of direct current of greater emf than the battery itself. The rate at which the battery accepts the charge must be kept reasonably low, otherwise overheating and subsequent buckling of the plates will result. There are a number of precautions to be observed when charging lead-acid batteries, other than during the trickle charge received from the aircraft generators, and these are summarised below:

(a) Charging must be carried out in a well-ventilated area, to disperse vented gas.

(b) The battery charger should be switched off before disconnecting the charging leads; to avoid sparks at the battery terminals.

(c) Ensure the battery vents are clean and working correctly.

(d) Remove the battery from the aircraft before charging as the electrolyte tends to vaporise during charging and it is highly corrosive.

(e) When removing the battery from the aircraft disconnect the negative lead first; when replacing it reconnect the negative lead last. This helps avoid accidental short-circuits between the airframe and the positive terminal of the battery.

84. As a lead-acid battery is being charged it gives off an explosive mixture of oxygen and hydrogen. It is for this reason that the battery compartment in an aircraft is vented to atmosphere. Lead-acid batteries should not be left in a discharged state for any significant period. If they are, the plates will become coated with lead sulphate and the battery will not subsequently accept a charge.

85. Battery fluid spillage will cause problems, both to the aircraft structure and to human tissue. Bicarbonate of soda (baking soda) is an effective neutralising agent for dealing with deposits in lead-acid battery compartments. Acid burns to the skin should be First Aid treated with a copious flow of water, followed by treatment with a dilute solution of bicarbonate of soda.

Nickel-Cadmium Batteries

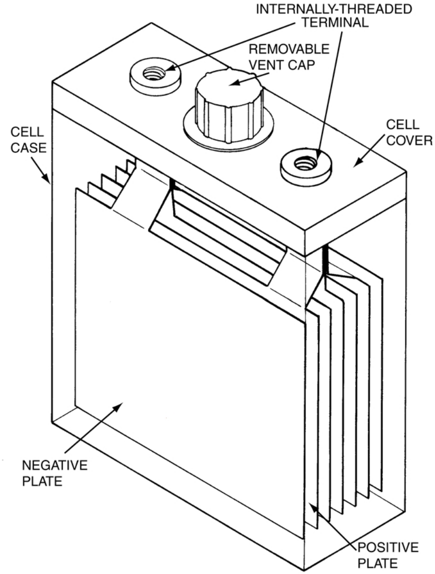
86. The lead-acid battery is by no means the only type of secondary cell in use. Another common type of cell in aircraft use is the Nickel-Cadmium (NiCad) battery. In this the positive plates are made of nickel hydroxide and the negative plates of cadmium hydroxide. The electrolyte is a solution of 70% distilled water and 30% potassium hydroxide. During charging the negative plates give up oxygen and become cadmium, whilst the positive plates pick up oxygen to form nickel oxides. During discharge the process is reversed.

87. NiCad batteries are lighter and more robust than lead-acid cells, they have a longer life, they are easier to store and they do not give off gases whilst charging. They also have a greater power-to-weight ratio. However, they are more expensive and they are only capable of producing 1.2 to 1.25 volts per cell. Consequently, a nickel-cadmium battery with a rated voltage of 24 volts may have 19 cells or 20 cells connected in series, depending upon the maximum total voltage required.

88. The specific gravity of the electrolyte is no indication of the state of charge of this type of battery, nor can the state of charge be determined by a voltage check against rated load, since voltage remains substantially constant over most of the discharge period. The only check that a NiCad battery is fully charged is the value of battery voltage when 'on charge'. As the state of charge reaches full charge the battery voltage rises to a maximum level, typically about 28 volts in a nominal 24-volt battery. As soon as the charging current is removed the battery voltage falls to about 25 volts.

89. Construction of NiCad batteries is fundamentally similar to that of Lead-Acid batteries, in that the cells are made up of interleaved alternate positive and negative plates immersed in the electrolyte and joined at their upper ends to the positive and negative terminals. A continuous woven nylon separator insulates the plates from each other. The cells comprising the battery are assembled within a container made of fibreglass or steel coated with epoxy resin. A diagram showing the construction of a typical nickel-cadmium cell is at Figure 7-16.

FIGURE 7-16



90. A Nickel-Cadmium battery may indicate overcharging by the formation of white crystals on the top of the battery. This is due to expelled electrolyte vapour reacting with the carbon dioxide in the atmosphere to produce potassium carbonate.

91. The capacity of a nickel-cadmium battery is much greater than that of a lead-acid battery of similar size and weight and thus it will deliver far more power and for a longer period. The battery voltage remains essentially constant over almost the whole of the discharge cycle, falling markedly only as the battery becomes fully discharged. This characteristic makes the NiCad battery particularly suitable for gas turbine engine starting, where a long start cycle requires protracted battery discharge before the engine-driven generators can supply power to recharge the battery.

92. The capacity of a NiCad battery is a direct function of the total plate area within the cells and may be up to 80-ampere hours (Ah) in a typical 24-volt battery. The Ah rating is always determined at a 5-hour discharge rate unless otherwise specified.

Thermal Runaway

93. Batteries will perform to their rated capacities so long as temperature conditions and charging rates are kept within the specified limits. If either is exceeded a condition known as thermal runaway may occur, which causes boiling of the electrolyte, violent gassing and eventual melting of the plates and battery casing.

94. Thermal runaway, or vicious cycling, is a condition to which Nickel-Cadmium batteries are particularly susceptible at high charging rates. During overcharging, oxygen is formed at the positive plates of the battery. If this oxygen reaches the negative plates it will re-combine with the cadmium and generate heat as a result. If this process is allowed to continue the battery may be seriously damaged, or even explode.

95. The condition is avoided by keeping charge rates within safe limits and by monitoring battery temperature. Some aircraft NiCad batteries incorporate a temperature sensor that activates an overheat-warning indicator, or a temperature gauge, in the cockpit.

96. If the temperature rises above a prescribed level the battery must be disconnected from the bus bar by opening the battery master switch.

Connecting Batteries

97. Two or more batteries of the same rating may be connected in series or in parallel as shown at Figure 7-17 and Figure 7-18. In series the voltage is the sum of all the battery voltages, but the ampere-hour capacity remains the same as for a single battery. When connected in parallel the voltage remains the same as for a single battery, but the ampere-hour capacity is the sum of the capacities of all the batteries.

FIGURE 7-17
Batteries Connected in Series

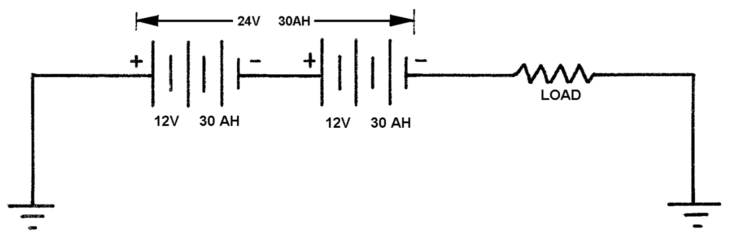


FIGURE 7-18
Batteries Connected in Parallel

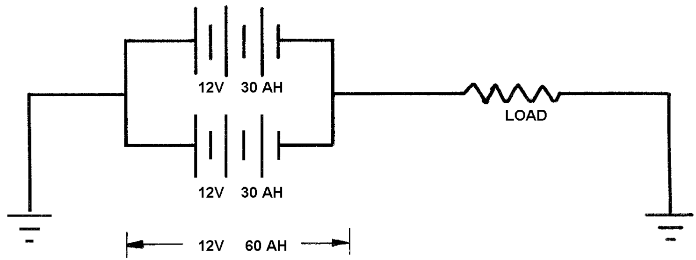


FIGURE 7-19
Comparison of Lead Acid and Nicad Batteries

	Lead Acid	Nicad
Voltage	2.1 volts per cell	1.2 volts per cell
Load Behaviour	Close circuit voltage (CCV) falls gradually from full charge to end of discharge cycle	CCV remains constant during entire discharge cycle
Thermal Runway	Not normally affected	This can present a big problem if battery is overcharged internal cell resistance is low and a high charge current can cause overheating
Storage Life	Best stored for short periods	Can be stored for long periods of time in a charged or discharged state

98. An advantage of the nicad battery is that it contains a greater power-to-weight ratio than the lead acid and it also has a higher capacity. It can accept high charge rates and can discharge at equally high rates without the voltage drop associated with lead-acid batteries. It also has a lower susceptability to very low temperatures. A disadvantage of the nicad battery is that it develops a memory which must be periodically erased.

99. Either lead-acid or nicad batteries are used in aircraft electrical systems. They provide uses ranging from main engine or APU starting, emergency lighting back-up supply to some navigation equipment (INS) through to an emergency DC supply in the event that all power generating sources fail.

Magnetism and Electricity

100. Magnetism and electricity are inseparable. When an electrical current flows through a conductor a magnetic field is created around the current-carrying conductor. The greater the current flow through the conductor, the greater the strength of the magnetic field surrounding it. This is the principle upon which electro-magnets work.

101. When an electrical conductor is placed within a magnetic field, providing that there is relative movement between the two such that the conductor cuts across the lines of magnetic force, electrical energy is induced in the conductor. This is known as electro-magnetic induction and is the principle upon which a generator works.

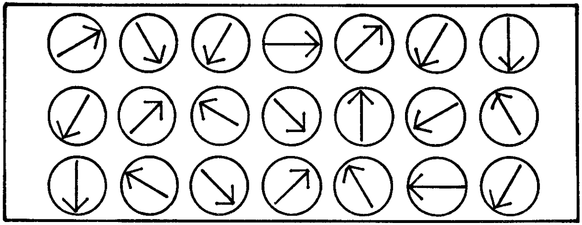
102. Before considering these principles it is necessary to discuss the basic principles of magnetism.

Magnetism

103. The name is derived from magnetite, an iron-oxide material that from early times was found to have the property of attracting iron and similar materials. The early explorers used this "lodestone" as a primitive compass since, when suspended freely, one end always points in a Northerly direction.

104. The composition of soft iron is such that groups of its atoms or molecules appear to combine to produce small permanent magnets, each known as a magnetic domain. Each domain has a north (N) pole and a south (S) pole. In a demagnetised piece of soft iron these domains will be randomly aligned as at Figure 7-20.

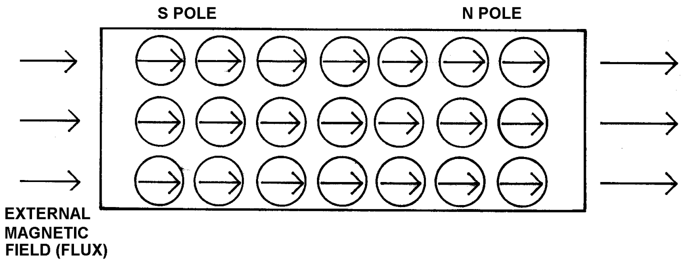
FIGURE 7-20
Domains - De Magnetised Soft Iron



105. When an external magnetic field is applied to a piece of soft iron the magnetic domains begin to align themselves with the polarity of the external field. Repeated application of the external magnetic field will result in more and more of the domains becoming aligned until, ultimately, they are all aligned in the same direction.

106. A bar of soft iron can be magnetised by stroking it along its length in one direction with one pole of a permanent magnet. Repeated application of an electro-magnetic field will have the same effect. At this point the soft iron bar is producing its maximum magnetic field strength and it is said to be magnetically saturated. This situation is illustrated at Figure 7-21.

FIGURE 7-21
Soft Iron - Magnetically Saturated

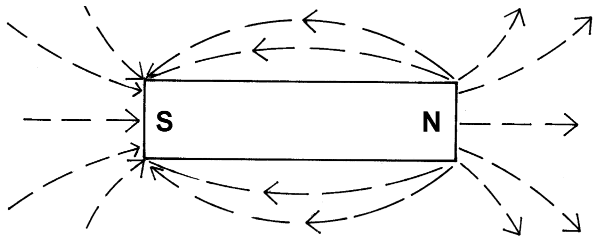


107. However, once the source of magnetism is removed most of the molecules of the soft iron will revert to their random polarisation and the material is no longer magnetised, or at best only a few remain aligned and the material retains a small amount of residual magnetism. This is because soft iron has high permeability, the ability of a material to become magnetised. It becomes magnetised easily because there is little internal friction between its molecules, but for the same reason it equally easily loses its magnetism.

108. Hard iron and some steel alloys have low permeability. The internal friction of their molecules makes them difficult to magnetise, but once the molecules have become aligned, or polarised, the same internal friction keeps them aligned. When such a material is removed from the magnetising source it retains its magnetism to become a permanent magnet.

109. The magnetic field, or flux, pattern of a permanent magnet is shown at Figure 7-22.

FIGURE 7-22
Permanent Magnet - Flux Pattern



110. When unlike or like magnetic poles are placed adjacent to each other flux patterns are produced as shown at Figure 7-23. Magnetic force, or flux, is considered as travelling from north to south in invisible lines. Whilst this is not literally the case it provides a useful reference by which calculations can be made and effects considered. Ferromagnetic materials that can be used for permanent magnets are:

(a) Hard steel

(b) Nickel

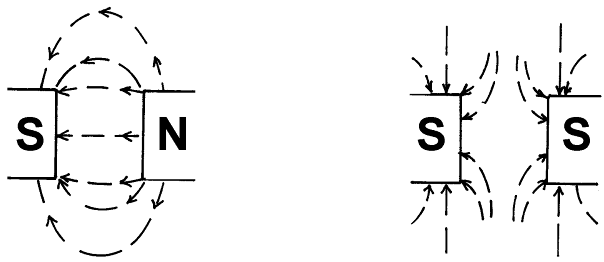
(c) Cobalt

(d) Alnico (alloy of iron, nickel and cobalt)

(e) Remalloy

(f) Permandur

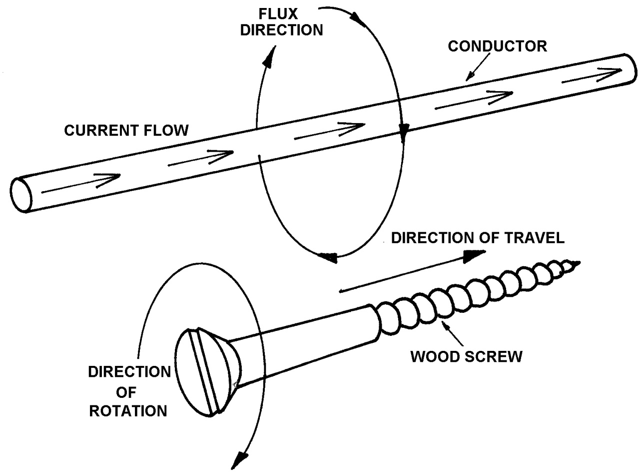
FIGURE 7-23
Polar Flux Patterns



Magnetic Fields Surrounding Current Carrying Cores

111. When an electric current flows through a conductor a magnetic field is produced around that conductor due to the movement of electrons through it. The direction of the magnetic flux is dependent upon the direction of current flow and can be determined by application of the 'screw rule'. This is shown at Figure 7-24.

FIGURE 7-24
Scew Rule

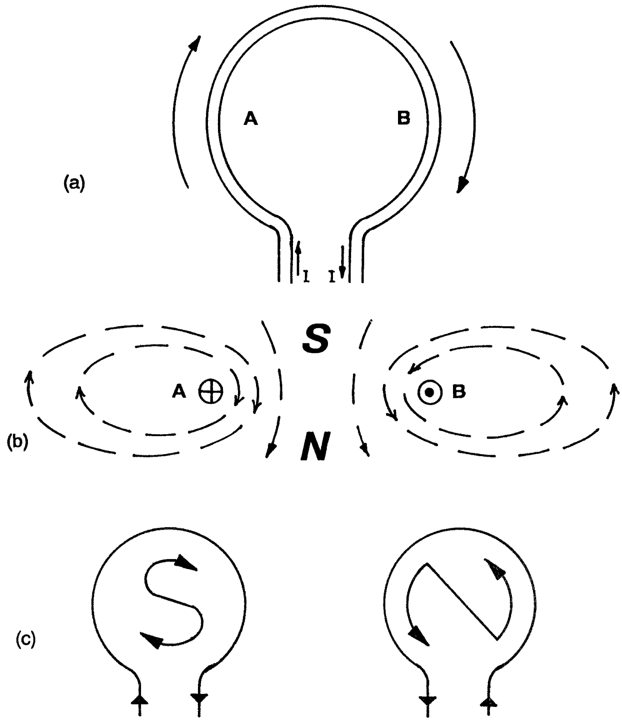


112. If you visualise a simple wood screw pointing in the direction of current flow the direction of the magnetic flux is given by the direction in which the screw must be rotated in order to propel the screw in the direction of the current flow.

113. When the current-carrying conductor is formed into a loop, the combination of two circular flux patterns produces a field having a north and south polarity, as illustrated at Figure 7-25.

FIGURE 7-25

Loop Flux Patterns



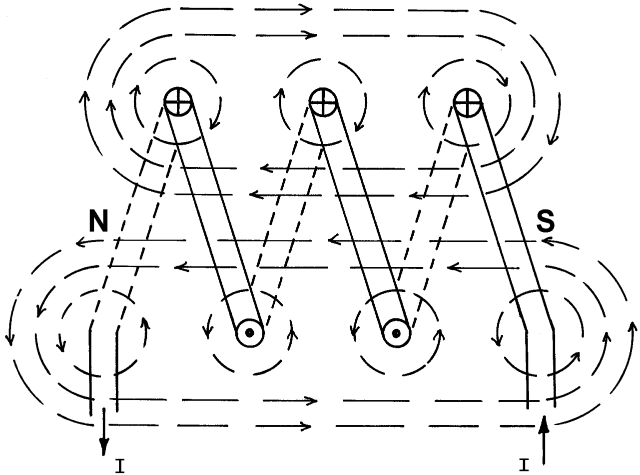
114. Figure 7-25 shows a plan view of the current-carrying loop and the direction of current flow. Figure 7-25 shows a section through the loop at A and B. With current flowing 'into' A and 'out of' B the two circular fields combine to produce a polarised field as shown. The strength of this magnetic field will depend upon the strength of current flow.

115. The 'dot' and 'cross' convention used in Figure 7-25 is another way of indicating the direction of current flow and the associated flux direction. The cross represents the tail, or flight of an arrow entering the paper, the dot the point of the arrow coming out of the paper. The arrow indicates conventional current direction (+ to -) and flux flow is always clockwise around the cross and anti-clockwise around the dot. Flux flow is always out of the magnetic field, from north to south.

116. The magnetic polarity at either end of a current-carrying loop can be determined by 'looking' at the end of the loop and using the method shown at Figure 7-25.

117. The effect can be concentrated, to produce a stronger magnetic field, by using a number of loops in the form of a coil of wire. The magnetic field produced by a coil of wire is shown at Figure 7-26.

FIGURE 7-26
Magnetic Field from Current Carrying Coil



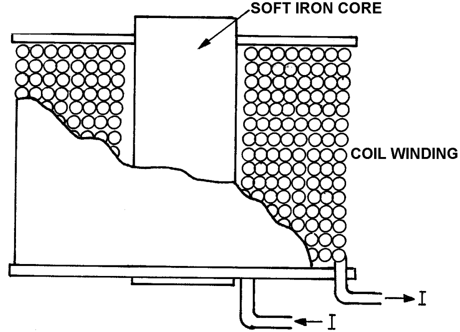
118. The magnetic field intensity, or strength, produced by a conducting coil is directly proportional to the current strength and the number of 'turns' in the coil and inversely proportional to the length of the magnetic circuit. If it is imagined that the coil is wound around a cylindrical core, the length of the magnetic circuit is, effectively, the length of the core.

Electro-Magnets

119. If a soft iron rod, or core, is placed within the current-carrying coil shown at Figure 7-27 the magnetic flux will traverse the iron core and its domains will align with the flux direction so that it becomes a magnet with polarity the same as the electro-magnetic field. Soft iron is used because of its high permeability, or ability to become magnetised or demagnetised. That is to say the magnetic domains will readily align with the external (coil-produced) field, but will take up random alignment when the external field is removed.

120. Since the external field requires current flow to produce it, the electro-magnet can be switched on or off by switching the coil current on or off. Furthermore, the strength of the electro-magnet can be increased (up to saturation level in the soft iron core) by increasing the current flow in the coil. Thus, the number of turns in the coil and the permeability of the core determine the strength of an electro-magnet. An example of an electro-magnet is shown at Figure 7-27.

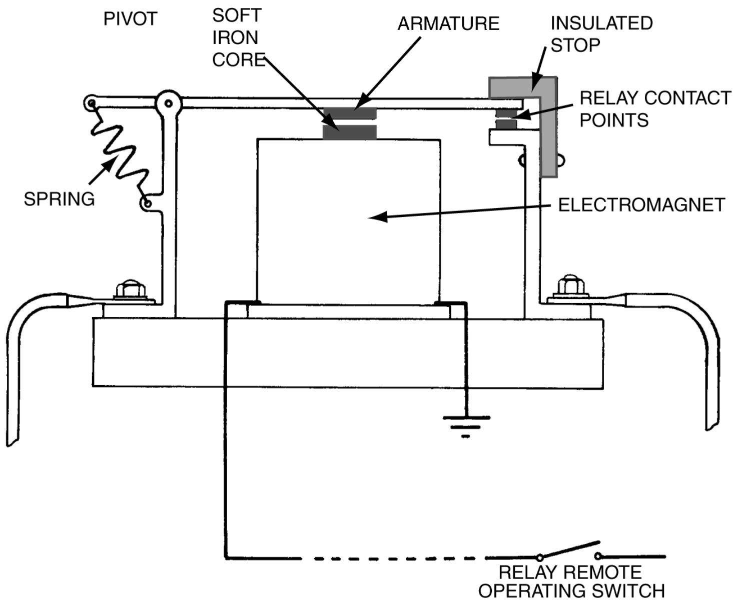
FIGURE 7-27
Electro Magnet



Relays

121. The flux field of the magnetism induced in the core of an electro-magnet may be used to create mechanical movement, which subsequently can be made to operate a piece of equipment, such as a switch. An electro-magnet having a fixed core and a magnetically operated mechanical linkage is called a relay and is often used for the remote operation of low-current switching devices. The principle is illustrated at Figure 7-28.

FIGURE 7-28
Relay



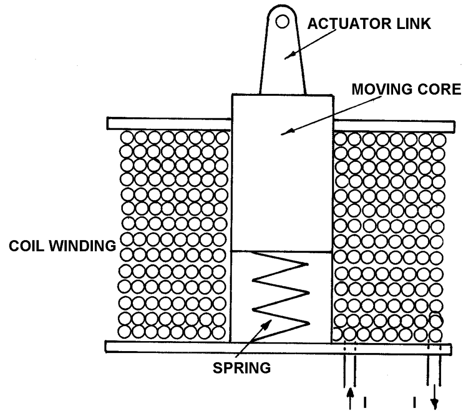
122. Attached to the mechanical linkage of the relay is an armature made from a material that responds to a magnetic field and which is therefore attracted by the field of the electro-magnet. The movement of the armature is used to open or close a set, or in some cases several sets, of contact points. In the example at Figure 7-28 the action of the electro-magnet closes the relay contact points to complete an electric circuit. The coil circuit is insulated from the contact circuit by means of insulated arms or insulated stops. The fuctional requirement of a relay will determine its type. A normally open relay will not activate a circuit until its coil is energised. A normally closed relay will de-activate a circuit when its coil is energised. A relay can employ a changeover contact which will function to activate one circuit and de-activate another when its coil is energised.

Solenoids

123. In some electro-magnets the soft iron core is movable, in which case the device is known as a solenoid. When the coil of the electro-magnet is energised the magnetic field produced attracts the soft iron core and draws it into the coil against the action of a spring. When current to the coil of the electro-magnet is switched off the electro-magnetic field collapses and the spring forces the core out of the coil. The linear motion of the core may be used to operate a variety of mechanical devices such as electrical contact points, valves and circuit breakers.

124. Usually the core of a solenoid consists of two parts, a non-magnetic sleeve fixed within the coil windings and a soft iron core that slides within the sleeve. A solenoid is illustrated at Figure 7-29.

FIGURE 7-29
Solenoid



125. The major advantage of solenoids is their suitability for remote operation. They can be sited almost anywhere within the aircraft and controlled from a remote position using low power switches or control units and circuitry.

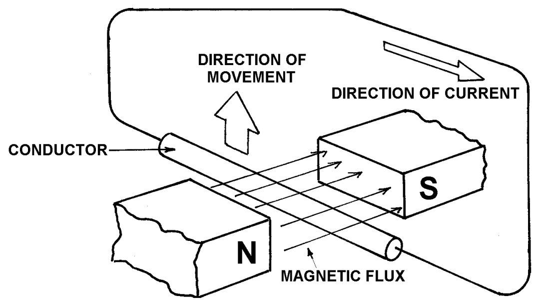
Electro-Magnetic Induction

126. Induction is the name given to the transference of electrical energy without the aid of conductors. When this transfer of energy is achieved by means of a magnetic field it is known as electro-magnetic induction.

127. Electro-magnetic induction occurs when there is relative movement between a conductor and a magnetic field such that the conductor is cutting across the lines of magnetic flux. If the conductor moves parallel to the lines of flux, there is no electro-magnetic induction. The relative movement may be achieved either by a moving conductor in a stationary field or a stationary conductor in a moving field. The field flux may be made to move by mechanically moving a magnet, as in some generators, or by varying the strength of the current in an electro-magnet, as in a transformer. The subject of transformers is discussed in the Power Distribution section of AC Electrics.

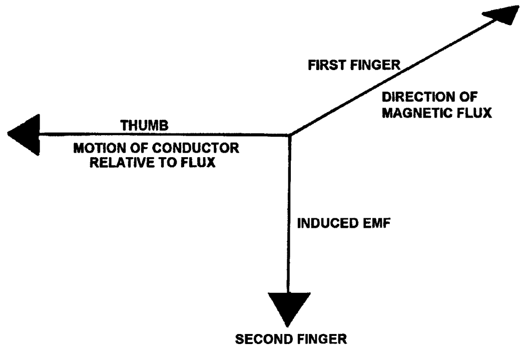
128. The principle of generator action, in which electro-magnetic induction produces power, is illustrated at Figure 7-30.

FIGURE 7-30
Generator Action



129. As the conductor is moved across the magnetic flux field an electromotive force (emf), or voltage, is induced in it. The voltage will cause current to flow, and thus power to be produced, if the conductor forms a complete electrical circuit, as at Figure 7-30. The direction in which the induced emf acts is determined by Fleming's right hand rule. This is illustrated at Figure 7-31.

FIGURE 7-31
Flemings' Right Hand Rule



The Laws of Electro-Magnetic Induction

The important laws concerning electro-magnetic induction are those laid down by Faraday and Lenz.

Faraday's Laws state that:

1st Law. An induced emf is established in a circuit whenever the magnetic field linked with the circuit changes. (in intensity or polarity).

2nd Law. The magnitude of the induced emf is proportional to the rate of change of the magnetic flux linked with the circuit.

Lenz's Law states that:

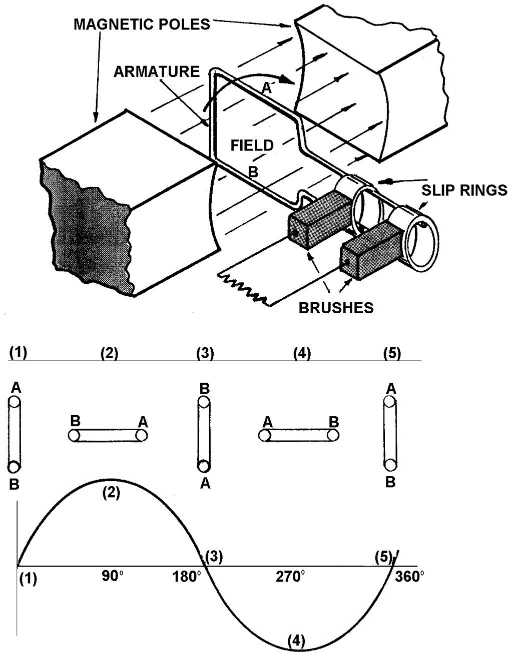
The induced emf acts to circulate a current in a direction that opposes the change in flux that causes the emf.

Generators

130. It was shown in section 2.127 that movement of a conductor within a magnetic field results in the induction of an emf in the conductor. It follows that, for continuous generation of emf in the conductor, there must be continuous relative movement between conductor and magnetic field. This is achieved by rotating a conducting loop within the flux field between the poles of a magnet.

131. Figure 7-32 shows a simple generator consisting of a conducting loop, or armature, placed in the field between the poles of a permanent magnet and rotated by an external mechanical drive (in an aircraft this drive is from the engine). As the armature loop rotates, the sides of the loop cut through the flux field and this relative movement between conducting loop and field induces an emf in the loop.

FIGURE 7-32
Simple Generator



132. Attached to each end of the loop is a conducting slip ring, which is in contact with a stationary brush made of carbon. The two brushes are connected to an electrical circuit in which is placed a load. The emf induced in the armature loop causes current to flow through the complete circuit formed by the armature and load circuit.

133. The progressive rotation of the armature in the stationary magnetic field is shown in Figure 7-32.

134. At point 1 the sides of the armature loop are moving parallel to the flux field. Hence, there is no relative movement between conductor and field and therefore no induced emf. This is also indicated on the output curve at Figure 7-32.

135. At point 2 the loop has rotated through 90° and the sides of the loop are moving at right angles to the field; relative movement is at its greatest and induced emf is maximum. This is indicated on the output curve.

136. At point 3 the situation is the same as at point 1, except that the armature has rotated through 180°.

137. At point 4 the situation is the same as at point 2, except that the armature has rotated through 270° in the stationary field, so the sides of the armature are moving in the opposite direction relative to the field. Induced emf is at a maximum, but is now of opposite polarity. Since, conventionally, current flows from positive to negative if the electrical polarity is reversed the current flow will be reversed. This is indicated by the reversal of the output curve at Figure 7-32.

138. At point 5 the armature has completed a full revolution and the situation is the same as at point 1. From the foregoing it will be seen that the voltage output of this simple generator reverses in polarity and fluctuates in magnitude from zero to a maximum every half cycle, or half revolution of the armature. Any current flowing in a circuit connected to the terminals of the brushes will reverse in direction and fluctuate in magnitude every half cycle. Such current is known as alternating current (AC). All rotary generators produce AC in their armature loops, or windings.

139. The device illustrated above is a simple AC generator, or alternator. Most modern aircraft use alternators as their primary source of electrical power, but clearly the output of the alternator cannot be used where DC is required, such as for battery charging. Consequently, the output of the alternator must be converted to DC, by means of a rectifier to meet such requirements. Virtually all modern, light piston-engine aircraft are equipped with engine-driven alternators whose output is rectified and distributed to all the major electrical components as DC. Larger commercial aircraft are equipped with AC Generators (big alternators) whose output is distributed as AC and only a few services use rectified AC (i.e. DC).

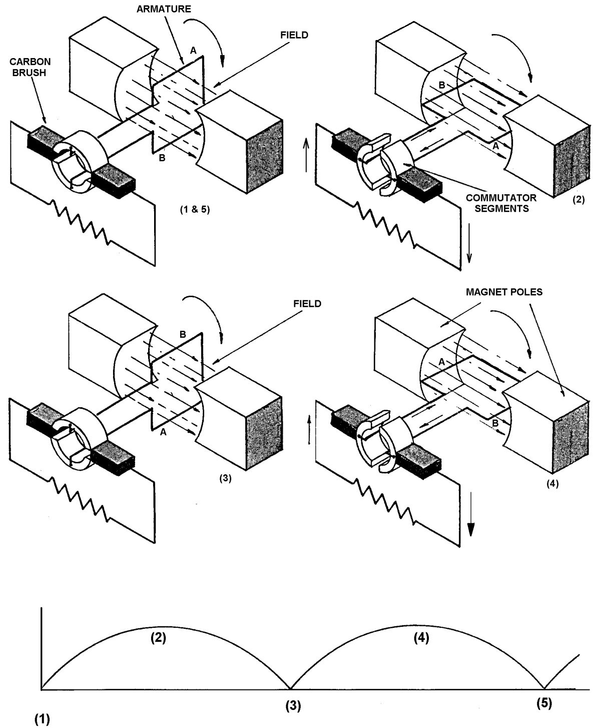
DC Generator

140. An alternative to rectification of the output of an alternator is to produce direct current at the outset, by means of a DC generator. Because DC generators are heavier than alternators, and generally require more maintenance, their use in modern aircraft is limited almost entirely to a combined engine starter motor and generator (starter generator) found in some small turbine powered aircraft

141. The construction of a simple DC generator is very similar to that of the simple AC generator described previously.

142. Figure 7-33 shows a simple DC generator from which a pulsating DC can be obtained by replacing the slip rings of the AC generator with a 'two-part slip ring', or split ring, known as a commutator. Each end of the armature loop is attached to one half of the commutator split ring and the two halves are insulated from each other, usually by a strip of mica.

FIGURE 7-33
Simple DC Generator



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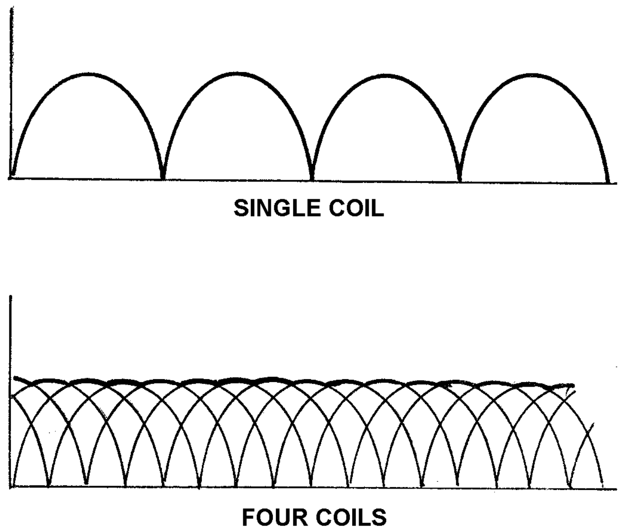
143. The stationary brushes are placed opposite each other so that, as the commutator rotates, the brushes pass from one commutator segment to the next as the current flow in the armature has ceased flowing in one direction and is about to start flowing in the opposite direction. This ensures that the potential (emf difference) between the commutator segments is minimal at this point of changeover, or switching. The effect of the commutator is to ensure that the electrical polarity of the brushes remains constant and so the direction of current flow in the load circuit remains constant.

144. As can be seen from the output curve at Figure 7-33 the DC produced by such a generator is pulsating and this is unsatisfactory for many current-consuming devices, which require constant voltage supply. This can be very nearly achieved by increasing the number of armature loops and the number of magnetic poles, producing a voltage and current flow which has only a slight pulsation, known as commutator ripple.

145. Figure 7-34 illustrates the change from pulsating DC produced by a single armature loop to the ripple DC achieved by four armature loops. It should be appreciated that, in the latter, the commutator would consist of eight segments instead of two.

FIGURE 7-34
DC from Four Armature Loops



146. Clearly the output of any generator using only one rotating loop of wire would be minute. Consequently many thousands of loops are wound onto a soft iron core to form the armature coil, or winding. The core serves additionally as a means of concentrating the magnetic field into the desired area.

Field Excitation

147. A fundamental requirement of any electrical distribution system is that the voltage delivered to the bus bar should be maintained constant under all conditions of electrical load. The aircraft generator, whether AC or DC, is driven by the aircraft engine and therefore its speed of rotation is variable, especially in the case of the piston engine aircraft.

148. Referring back to Faraday's second law it is apparent that the emf induced in the armature windings of a generator will vary directly with the speed of rotation of the armature. However, it is clearly not a practical proposition to use generator variable speed control as a means of controlling the emf induced and therefore the output voltage.

149. The magnitude of the emf induced in a conductor by electro-magnetic induction is dependent upon the following factors:

(a) The rate at which the conductor is cut by the lines of magnetic flux (in this case speed of armature rotation).

(b) The length of the conductor (determined by the number of turns in the armature winding).

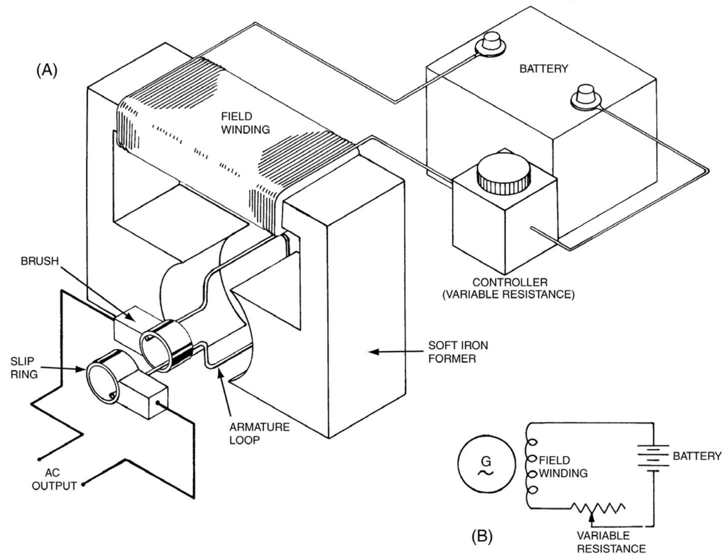
(c) The flux density (the strength of the magnetic field).

150. We have already seen that it is not practical to use the first factor as a means of generator output voltage control, and the second is fixed during manufacture. This leaves only one option remaining, variation of the strength of the magnetic field within which the armature rotates.

151. In the simple generators illustrated at Figure 7-32 and Figure 7-33 the generator field is produced by a permanent magnet and is therefore clearly not variable. However, as was shown in the previous chapter, passing a current through a conductor and varying the current strength can produce a variable strength magnetic field. If the conductor is wound around a soft iron core the magnetic field will be concentrated in the core and the device will be an electro-magnet, the flux density of which can be varied from zero to saturation level. This is known as field excitation.

152. By this means the emf induced in the generator armature, and therefore the generator output voltage, can be controlled regardless of generator speed or electrical load by varying the current supplied to the core winding of the electro-magnet. This core winding is known as the generator field winding and the current supplied to it as the generator field current. Figure 7-35 illustrates the basic concept.

FIGURE 7-35
Field Excitation

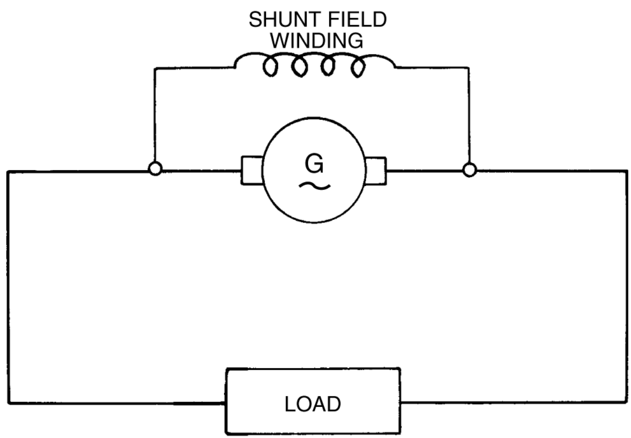


153. In the diagram at Figure 7-35a the field current is supplied from a battery and the strength of the field current is controlled by means of a variable resistance. Decreasing the resistance at the controller allows more current to flow to the field coil, which increases the flux density in the core and thus the strength of the magnetic field in which the armature loop is turning. Consequently, greater emf is induced in the armature and the generator output voltage rises. Increasing the resistance at the controller has the opposite effect.

154. Figure 7-35b shows the field excitation circuit in schematic form. When the generator field current is supplied from an external source of direct current, as in this case, this is known as external, or separate excitation. Field current must be DC in order to maintain constant polarity of the magnetic field so the source of field current for an externally excited generator is normally the aircraft battery.

155. In DC generators the current to the field coil is usually taken from the generator output, and this is known as self-excitation. This arrangement is shown schematically at Figure 7-36. From the output (positive) terminal of the generator current is supplied to the field coil through a controlling variable resistance to the field winding.

FIGURE 7-36



156. It will be noted that the field winding is in parallel with the generator armature, this is known as a shunt-wound generator and is the usual configuration. The current flow through the field winding is quite separate from that flowing from the generator to the distribution system loads and is therefore relatively simple to control, in order to maintain constant output voltage.

157. In the purely diagrammatic representation at Figure 7-35 a single electro-magnet is shown, but in practice there are a number of electro-magnets spaced evenly around the armature.

158. The soft iron of the electro-magnets retains a small amount of magnetism, known as residual magnetism, even when there is no field current. This residual magnetism is sufficient to induce an emf in the armature of the generator when it first starts to rotate, which initiates a current flow from the generator. This current flow supplies the field coils, increasing the magnetic field of the electromagnets, increasing induced emf in the armature, and so on.

159. Residual magnetism may be lost, or its polarisation reversed, due to excess heat, shock or reversal of field current flow. The residual magnetism can be restored by briefly passing a current through the field. This is known as field flashing, or flashing the field.

160. In circumstances where the generator speed can be maintained constant and load variations are relatively small it possible to configure the DC generator such that the field windings are in series with the armature (and therefore the load). Also, a type of generator in which some of the field windings are in series, and some in parallel with the load, known as a compound-wound machine is sometimes used. In summary it can be said that generator voltage depends on the number of turns in the armature, the strength of the field, the rpm of the armature and the supplied load.

Alternators

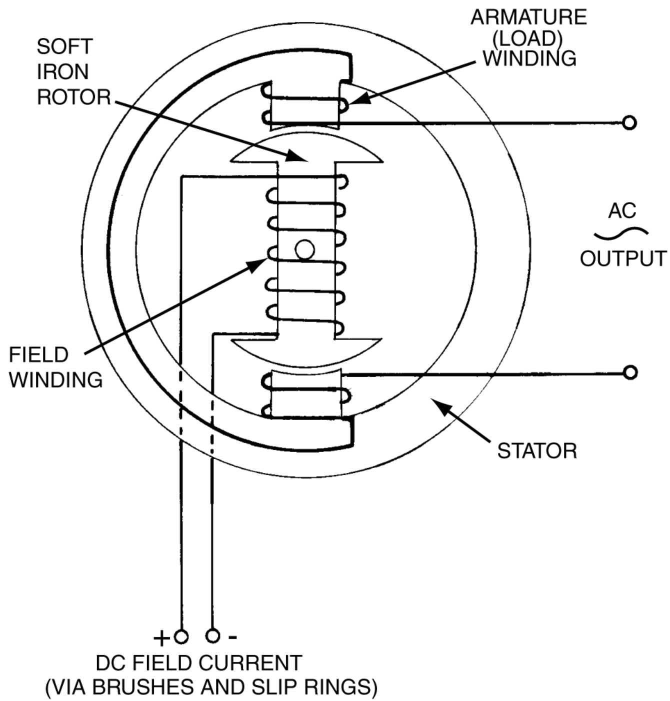
161. Many light single and twin-engined aircraft use direct current power distribution supplied from alternating current generators (alternators), the alternator AC output being rectified to DC at source. Alternators are lighter than DC generators and do not suffer from the problems of arcing (and the consequent radio interference) produced by commutation.

162. Because the alternator does not require a rotating commutator there is no need for the armature to be mounted on the rotor and the large load current to be transferred to the output terminals through slip rings and brushes. Instead, the armature winding is in the stationary casing of the machine and the generator field windings and their electro-magnets are on the rotor. Consequently, only the relatively small field current need be passed through brushes and slip rings to the rotating field windings.

163. Thus, in aircraft alternators, the rotating magnetic field cuts through the stationary conductors of the armature winding, inducing emf. The armature winding is connected to the output terminals of the alternator, from which the load current is supplied to the distribution bus bars through a rectification system that converts the AC output to DC.

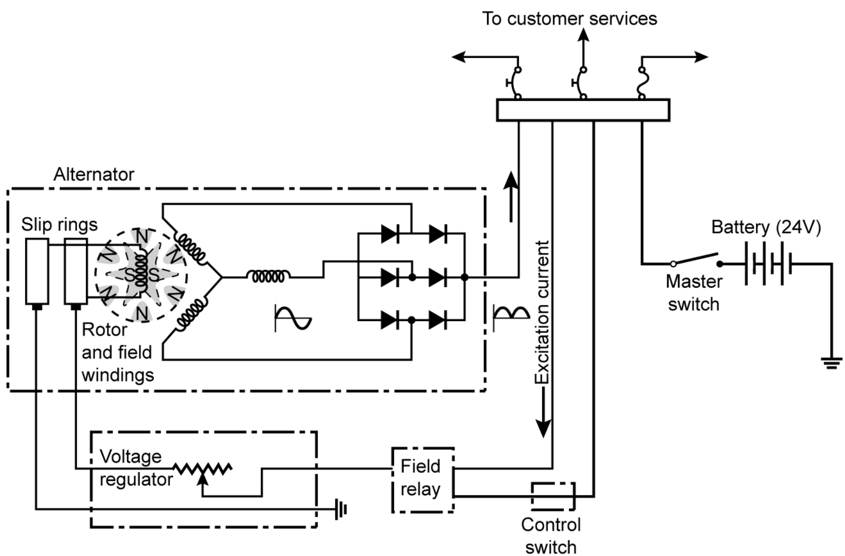
164. Figure 7-37 shows a simple alternator arrangement. The machine illustrated would produce single-phase alternating current. Single-phase and three-phase AC is explained in paragraphs 306-321.

FIGURE 7-37
Simple Alternator



165. The alternator output voltage is automatically controlled by regulating the current supplied to the field windings in the rotor. This is done by means of a variable resistance in series with the field windings called a voltage regulator. Figure 7-38 shows a schematic alternator circuit for a light aircraft 28-volt DC distribution system.

FIGURE 7-38
Alternator Circuit



166. The field current for the separately excited alternator is supplied from the aircraft battery.

167. A rectifier is a static semiconductor device that permits current flow in one direction only and thereby converts bi-directional AC into unidirectional DC.

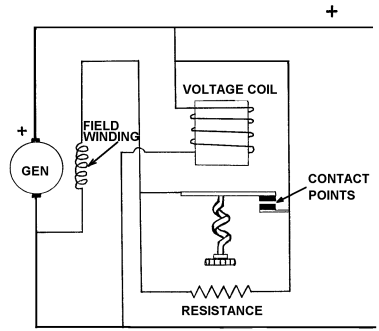
Control and Protection

168. The control and protection devices needed for an alternator supplying DC to the electrical distribution system of a light aircraft are a voltage regulator and a current limiter. In the majority of systems the current limiter is simply a circuit breaker that automatically opens if load becomes excessive.

169. Voltage regulators work on the principle of sensing the alternator output voltage and adjusting field current to maintain voltage at a constant value. Typically the voltage regulator is a transistorised unit that allows a set current to flow to the alternator field coil when alternator output voltage falls below a set value (say 27.5 volts). When output voltage rises above a set value (say 28.5 volts) it cuts off the current supply to the field coil. This cycle is repeated about 2000 times per second, maintaining alternator output voltage at about 28 volts.

170. An older version of the same principle is the vibrating contact voltage regulator. In light aircraft, using comparatively low output generators, voltage regulation is often by means of a vibrating contact regulator. The regulator contains a voltage coil, which regulates generator voltage and current. A vibrating contact regulator system is illustrated at Figure 7-39.

FIGURE 7-39
Vibrating Contact Regulator



171. The regulator contains a fixed resistance that is switched in or out of the field circuit by an electro-magnetically-controlled spring-loaded switch (the vibrating contact). The contact points are held closed by the spring until the current flow through the voltage coil creates sufficient electromagnetism to attract the ferritic contact armature and open the points against the force of the adjustable spring.

172. When generator voltage is low the current flow through the voltage coil is insufficient to open the contact points. Field current to the shunt field winding flows through the points, creating a strong generator magnetic field and causing output voltage to rise.

173. When output voltage reaches a predetermined level, current flow through the voltage coil becomes sufficient to create an electromagnet of sufficient strength to open the contact points. Field current can now only reach the shunt field winding via a fixed resistance. The low resultant field current reduces output voltage, and electromagnetic strength of the voltage coil, until the spring closes the contact points once more.

174. This process is repeated at between 50 and 200 cycles per second, maintaining an essentially steady voltage. The generator output 'control' voltage is adjusted by adjusting the tension of the spring. Vibrating contact regulators are only suitable for use with generators where the field current is low (less than 8 amps). Above this level the vibrating contact points would become overheated and possibly fuse together.

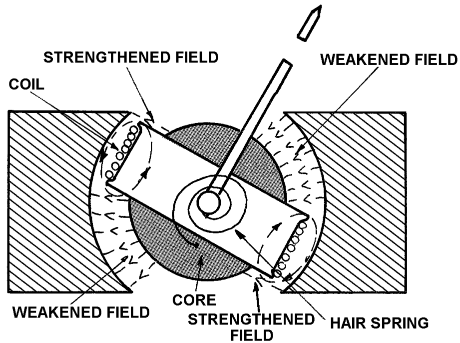
Monitoring Devices

175. There are basically only two monitoring instruments required for an alternator-supplied DC distribution system. These instruments are an ammeter to show current flow and a voltmeter to indicate proper functioning of the voltage regulator. Of these two instruments, the ammeter is the more important and most single engine aircraft are not fitted with voltmeters, only with a warning indication if voltage strays outside preset limits.

176. Ammeters and voltmeters are remarkably similar in construction. An ammeter is connected in series with each generator to show the output current (load). A voltmeter is connected in parallel with circuit loads to show circuit voltage. Both instruments are usually of the permanent magnet 'moving coil' type, shown at Figure 7-40.

177. Voltmeters usually have the required resistance built into the instrument itself. Increasing the measuring range of the voltmeter can be achieved by the use of additional resistances connected in series with the instrument. The resistances are called multipliers. Increasing the measuring range of an ammeter is achieved by the addition of a resistor connected in parallel with the instrument. This is called a shunt resistance.

FIGURE 7-40
Moving Coil Instrument

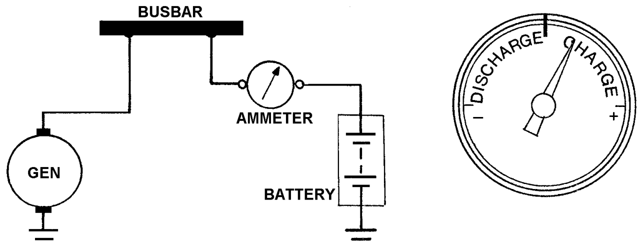


178. A coil is wound around a former mounted upon a soft iron core. The core is carried on a spindle supported by a bearing, so that it is free to rotate between the poles of a permanent magnet. Rotation of the core is limited, however, by a hairspring, one end of which is attached to the core spindle and the other to the casing of the instrument.

179. Current flow through the coil sets up a magnetic field, which interacts with the permanent field and causes the coil and core to rotate towards the weakened field, against the hairspring. The greater the current flow, the greater the extent of rotation. Attached to the rotating core is a pointer that moves against a fixed scale, calibrated in amps or volts as required.

180. Reversal of current flow through the coil reverses the direction of the rotary movement and this is employed in the 'centre-zero' arc scale type of ammeter used in conjunction with batteries to show charge or discharge, see Figure 7-41. Reversible moving coil indicators employ two hairsprings with opposite winding, to restrict rotation of the core in either direction.

FIGURE 7-41
Centre-Zero Ammeter



181. A generator ammeter, or loadmeter, is of the 'left-zero' type, illustrated at Figure 7-42, which shows an increased deflection in a clockwise direction with increased current flow through the coil.

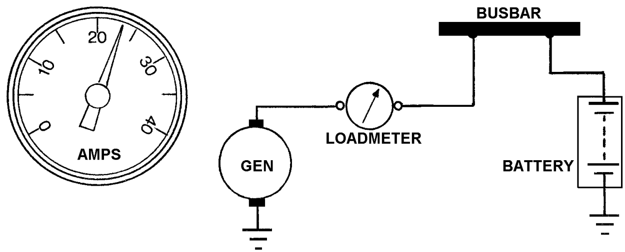


FIGURE 7-42
Left-Zero Ammeter

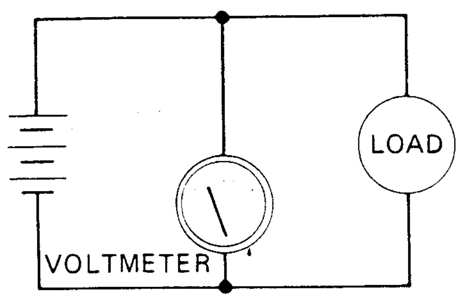


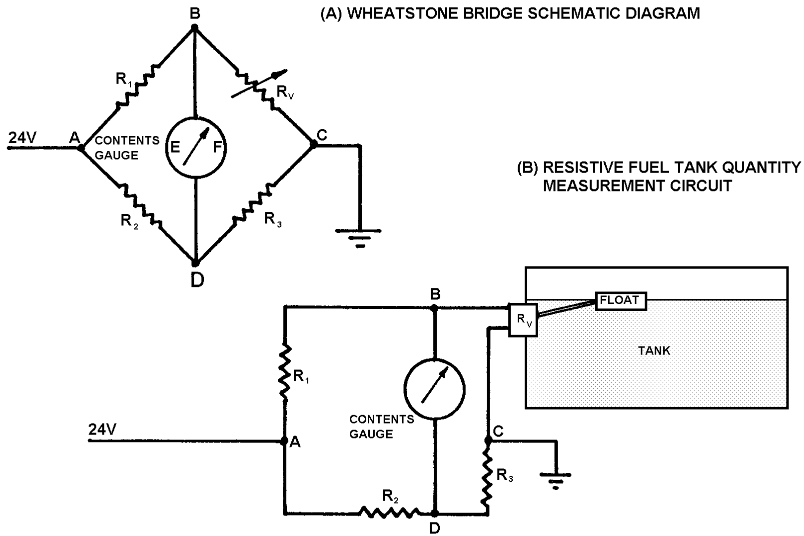
FIGURE 7-43
Voltmeter Circuit

182. Ammeters are connected in series with the load and are designed to offer a very low resistance. Voltmeters are connected in parallel with the load and are designed to offer a very high resistance, so that the majority of the current flows through the load and not through the voltmeter. Current flow is proportional to the voltage in a circuit of fixed resistance, so the scale of the instrument can be calibrated in volts for a range of current flows. See Figure 7-43.

Wheatstone Bridge

183. A typical bridge circuit is illustrated at Figure 7-44. The bridge circuit is in balance if the four resistances R_1, R_2, R_3 and R_V are equal i.e. 100 ohms. Now consider the variable resistance R_V changes to a value of 200 ohms. It can be seen that one side of the bridge has a total resistance of 200 ohms $R_2 + R_3$ and the other side R_1+R_V a total resistance of 300 ohms. Applying ohms law (V = I x R) with the supply voltage of 24V the current flow through R_2 and R_3 is .12 amps (24 = I x 200) and the current flow through R_1 and R_V is .08 amps (24 = I x 300). Using these values we can calculate the voltage drop R_2/R_3 and R_1+R_V Points B and D. Voltage drop across R_1 is 8V (V = 0.8 x 100). Therefore potential at B = 24 - 8 = 16V. Voltage drop across R_2 = 12V (V = .12 x 100). Therefore potential at D = 24 - 12 = 12V. If an instrument is placed between points B and D, current will flow from Point B (16V) to Point D (12V). Measurement of this current can be represented on a suitably calibrated scale i.e. fuel contents gauge.

FIGURE 7-44



Annunciators

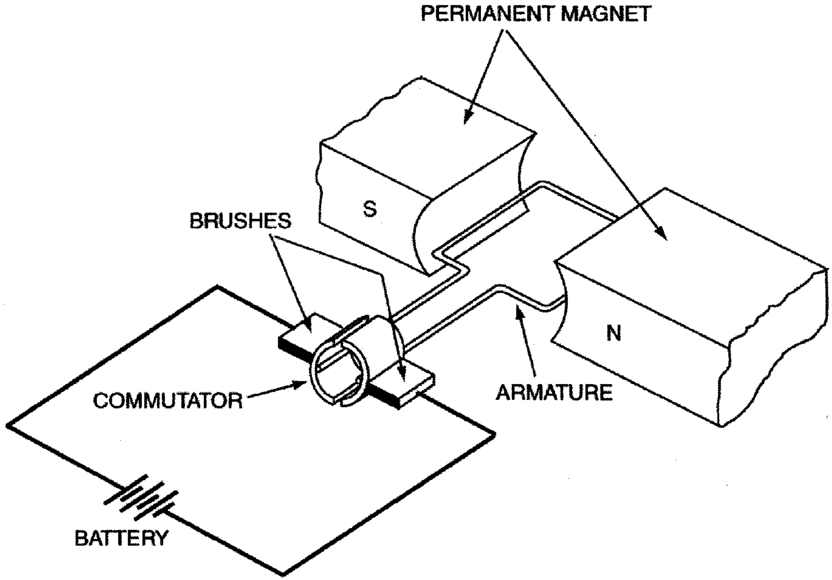
184. Display of normal, abnormal, malfunctioning or failure of various systems in an aircraft can be indicated by using various coloured lights for instance, warning lights are red, caution lights - amber and indicating lights - blue/green. Some aircraft incorporate magnetic indicators (MIs) as a form of annunciation. Some annunciations can be pre empted or accompanied by aural warnings (bells and chimes).

DC Motors

185. There are a number of instances where it is necessary for an electric motor to operate from the aircraft battery, rather than from the engine-driven alternator. An obvious example of this is the engine starter motor. Clearly such a motor must be capable of operating from a direct current supply. Compared with its AC equivalent, the DC motor is capable of producing greater starting torque and so is preferable, even when AC power is available, in certain applications. Examples of these are wing flap motors and landing gear.

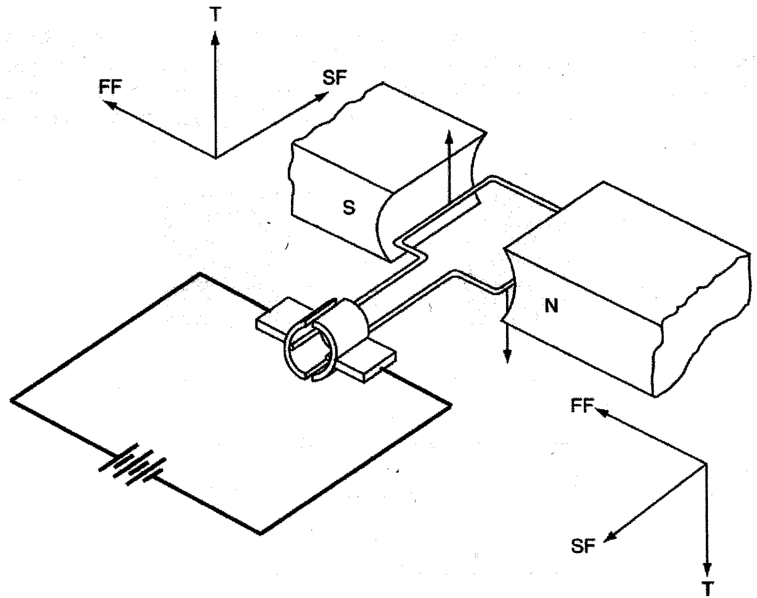
186. A DC motor comprises an armature mounted in bearings and that is free to rotate within a stationary magnetic field. The armature is a conducting loop supplied with direct current and the stationary magnetic field may be supplied by a permanent magnet or it may be electro-magnetic. Direct current is supplied to the rotating armature via stationary carbon brushes that are in contact with the two halves of a longitudinally split cylinder known as a commutator. Figure 7-45 illustrates a very simple DC motor in which the armature consists of a single conducting loop and the stationary field is supplied by a permanent magnet. In practical motors the armature comprises many conducting loops and the stationary field is controlled electro-magnetically.

FIGURE 7-45
Simple DC Motor



187. Electric motors operate according to Fleming's Left Hand Rule. To demonstrate this the thumb, first and second fingers of the left hand are arranged such that they are mutually at right angles to each other. With the first finger pointing in the direction of the magnetic field (N to S) and the second finger pointing in the direction of conventional flow within the conductor, the thumb will point in the direction the conductor will tend to move. This is illustrated at Figure 7-46.

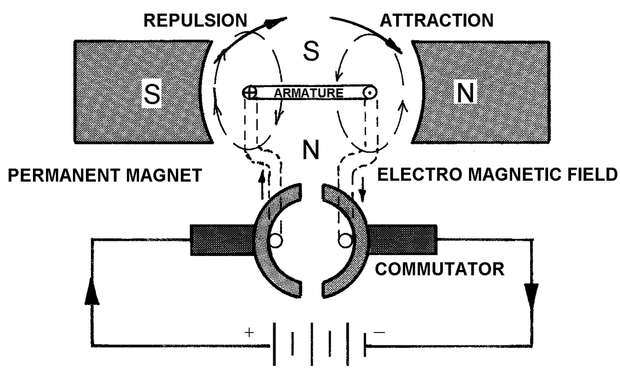
FIGURE 7-46
Fleming's Left Hand Rule



DC Motor Principle of Operation

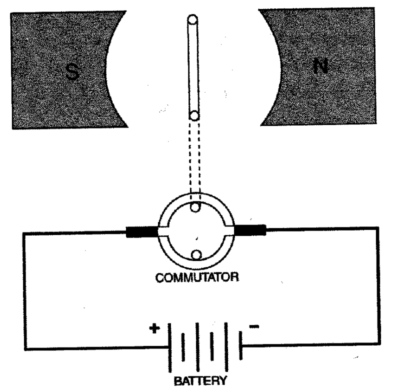
188. The movement of a current carrying conductor within a magnetic field is due to the interaction between the static field and the flux created around the conductor by the current flowing through it. Figure 7-47 shows the simple DC motor of Figure 7-45 with the direction of current flow in the armature indicated by a cross and a dot, and the consequent flux field produced. The cross indicates current flowing away from the viewer and the dot indicates current flowing toward the viewer. Since, in magnetism, like poles repel and unlike poles attract the armature in the diagram is compelled to rotate in a clockwise direction.

FIGURE 7-47
DC Motor Principle (1)



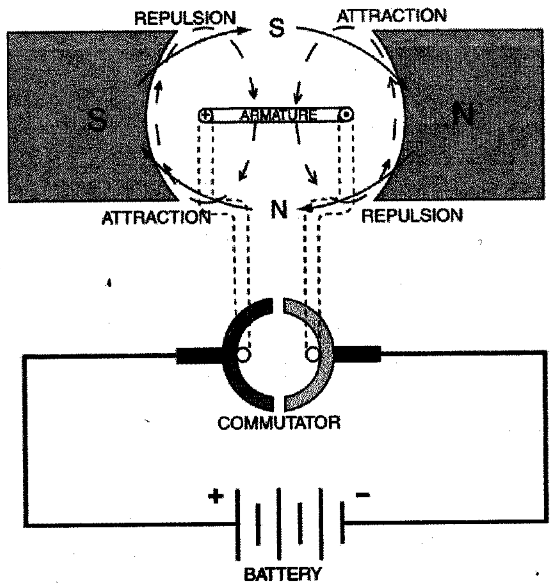
189. With refernce to Figure 7-47, current is being supplied through the left hand half of the commutator and is returning through the right hand half. This current flow through the armature loop, indicated by the cross and dot symbols, has produced an electro-magnetic field as described in the preceeding paragraph. The consequent repulsion and attraction produces a torque force to rotate the armature in a clockwise direction.

FIGURE 7-48
DC Motor Principle (2)



190. After a quarter of a revolution the armature and its attached commutator will have reached the position shown at Figure 7-48. Because of the gaps in the commutator there is now no current flow through the armature and therefore no electro-magnetic flux. However, inertia will keep the armature rotating.

FIGURE 7-49
DC Motor Principle (3)



191. It will be seen from Figure 7-49 that as the armature passes beyond the quarter turn position, current once again flows through the commutator halves in the same direction as before. The polarity of the electro-magnetic field is thus maintained constant, maintaining the direction of torque.

Back emf and Net emf

192. It has already been shown that when a conductor is moved relative to a magnetic field an emf is induced in the conductor. The loop of an armature is moving through the stationary field as the armature rotates and this inevitably induces an emf in the armature. This emf produces a current flow that opposes the applied current from the battery and therefore reduces the total armature current flow. The induced voltage is known as back emf.

193. The difference between the applied emf and the back emf is known as the net emf, and it is this that determines the torque produced in the armature shaft. In order to ensure that the net emf is sufficient the resistance of the armature windings is kept as low as possible.

194. The initial current flow through the armature, before it begins to rotate, is determined by the applied voltage and the armature resistance. If the resistance is low the current flow will be very high. As the motor gains speed the back emf increases and reduces the current flow through the armature. This explains why the current to a DC motor shows a surge on starting and then quickly falls to a much lower value. To avoid excess starting current, some DC motors have a resistance built in to the armature windings, which automatically cuts out as motor speed increases.

Types of DC Motors and their Characterstics

195. The stationary field of a DC motor is invariably produced electro-magnetically, so that varying the current flow through the field windings will vary the field strength. DC motors fall into three types, according to the arrangement of the field windings in respect to the armature windings. These types are series wound, shunt wound and compound wound, They are illustrated diagrammatically at Figure 7-50, Figure 7-51 and Figure 7-52.

Series Wound Motor

196. In the series wound motor the field coils are connected in series with the armature. Thus, the current flow through both is the same and the magnetic flux induced in both armature and field coils will be strong, producing high torque in the armature shaft. This is particularly true at starting, when the current flow is very high. Consequently a characteristic of the series wound motor is high starting torque. This is useful in circumstances where the motor will be required to start against a high load and where the running load is also high. The schematic wiring arrangement for a series wound motor is shown at Figure 7-50.

197. Examples of instances where series wound motors are used are engine starter motors, flap operating motors and landing gear operating motors.

198. Series wound motors should never never be allowed to operate without a mechanical load applied. This is because they are liable to overspeed, possibly to destruction. The reason for this is that, as the armature speed increases the induced back emf reduces the net emf and therefore the current flow through both armature and field windings. The reduced current flow through the field windings reduces the strength of the stationary magnetic field, preventing the back emf from reaching a value where the net emf is zero. Hence there is always a net emf to continue accelerating the unloaded motor.

FIGURE 7-50
Series Wound

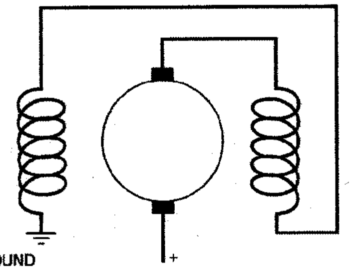
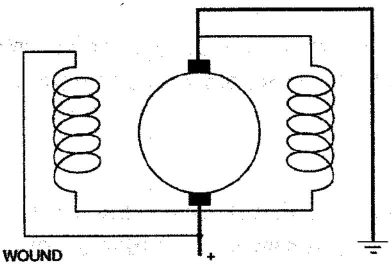


FIGURE 7-51
Shunt Wound



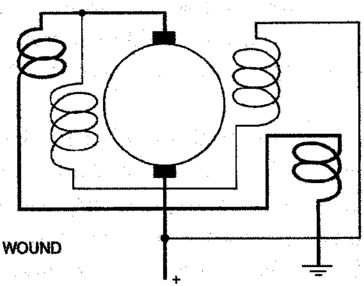


FIGURE 7-52
Compound Wound

Shunt Wound Motors

199. This schematic arrangement of the wiring for a shunt (or parallel) wound DC motor is shown an Figure 7-51. From this it will be seen that the field coils are connected in parallel with the armature windings. The resistance of the field coils is deliberately set to limit the field current to that required for normal operation of the motor, and is much higher than the armature resistance. This is usually achieved by using many thousands of turns of very fine wire for the field windings.

200. On start up the current flow through the armature is high, because of its low resistance. The current flow through the field coils is low however, because of their relatively high resistance, and field strength is correspondingly weak. Consequently, a characteristic of the shunt wound DC motor is low starting torque.

201. As the armature speed increases, increasing back emf will cause the armature current to decrease. Because the back emf does not affect the current flow through the field coils, since they are in parallel to the armature, the relative value of the field current to the armature current increases with increased motor speed. The result is that the torque increases until back emf almost equals applied emf and the motor settles at its normal operating speed. This speed is virtually constant within the design mechanical load range of the motor.

202. Shunt wound motors are used when starting torque is low and increases with motor speed. They are particularly useful where constant speed under varying load conditions is a requirement. Typical applications in aircraft are fuel pumps and fans.

Compound Wound Motors

203. Figure 7-52 shows the schematic arrangement of a compound wound DC motor. This has two sets of field windings, one connected in series with the armature and the other in parallel. The low resistance series windings are shown in heavy lining, the higher resistance shunt windings in lighter lining. The compund wound motor combines the characteristics of the series wound and the shunt wound motor. It is capable of high starting torque, but will not overspeed under light mechanical loading and will maintain a reasonably constant speed under varying conditions of load.

204. The compound wound motor is suited to applications where loads may vary from zero to maximum and where starting loads may be high. In aircraft they are often used to drive hydraulic pumps.

Reversible DC Motors

205. In certain aircraft applications, such as motor-operated flaps and landing gear, it is necessary for the motor to be reversible. Clearly this could be achieved by means of a switching arrangement that reversed the polarity of the DC supply to either the field or the armature (but not both). This would reverse the magnetic attraction and repulsion and thus reverse the direction of rotation of the armature. However, such switching would be complex and it is more usual to employ a spli-field motor where it is necessary for the motor to rotate in either direction.

206. In a split-field motor there are two sets of field windings, either wound in opposite directions on a common pole (or core) or on alternate poles around the inside of the motor casing. The motor is controlled by a single-pole double throw (SPDT) switch that can be placed in one of three positions. When the switch is placed in the mid-position the supply to the motor is broken. When thrown in one direction one set of field coils is energised and the motor rotates in (say) the clockwise direction. When thrown in the opposite direction the alternate field coils are energised, the field polarity is reversed and the motor rotates in the anti-clockwise direction. A split-field schematic circuit is shown at Figure 7-53.

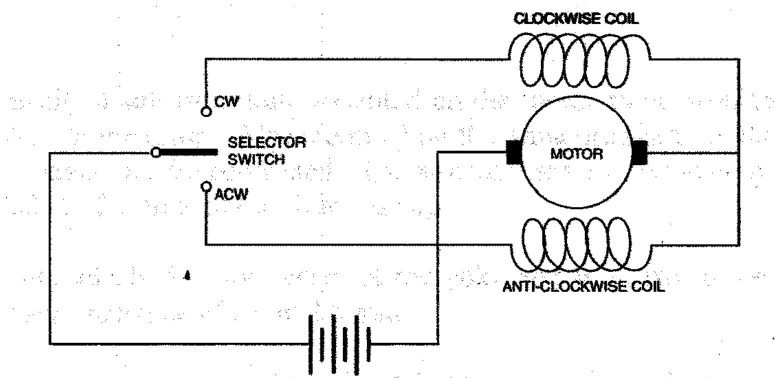


FIGURE 7-53
Schematic Circuit Diagram for a Split-Field Motor

207. Since the type of applications requiring a reversible motor also uses series wound motors, the field coils are in series with the armature and must therefore be wound with thick, low resistance wire. Reversible motors with a rating higher than 20 amps or so employ a remotely operated switching relay in the supply to the field coils.

208. Some low powered DC motors use a magnet instead of a field coil and in these the direction of rotation is reversed by reversing the polarity of the DC supply. Since only the armature supply need be reversed the switching is simple. Light aircraft flap systems often use permanent magnet reversible motors.

Aircraft DC Motors

209. In order to keep weight to a minimum it is important that aircraft electric motors have a high power-to-weight ratio. To achieve this they operate at considerably higher rotary speeds than in industrial applications and at higher armature currents. There is consequently a tendency for aircraft motors to operate at high temperatures and air-cooling is usually necessary to dissipate the heat generated. To keep centrifugal forces as low as possible armature diameters are deliberately kept small.

210. When DC motors are operated under emergency load conditions a cooling period is often necessary before they can be used again. Some motors are rated for intermittent duty only and these will overheat and the internal insulation may burn if they are operated continuously. Continuous duty motors are necessarily of a lower power-to-weight ratio. The type of duty for which a motor is rated is listed in the manufacturer's specifications and, possibly, on a rating plate attached to the motor casing.

DC Motor Construction

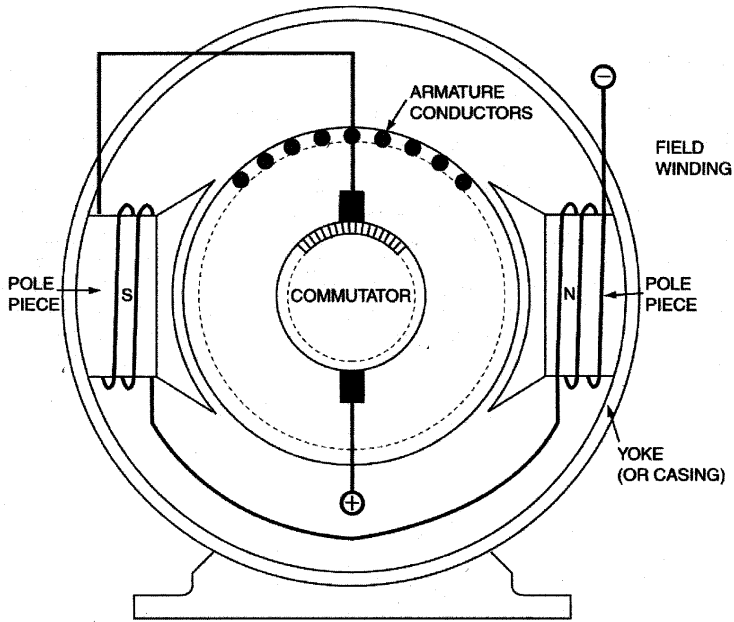
211. The major components of a typical aircraft DC motor are the armature, the yoke (or casing) and the field coils.

212. The armature is typically a soft iron drum mounted on the motor shaft, with the armature conductors set axially into the surface of the drum. Also mounted on the armature shaft are the commutator segments, to which the armature conductors are connected. The armature shaft is mounted in ball bearings at each end, the bearings being held in the ends of the motor casing.

213. The field windings are attached to the inside of the yoke and form two poles fitting closely around the armature with a running clearance of about 2.5mm.

214. A schematic drawing showing a section through a DC motor is at Figure 7-54.

FIGURE 7-54
Section Through a DC Motor (Schematic)



Starter Generator

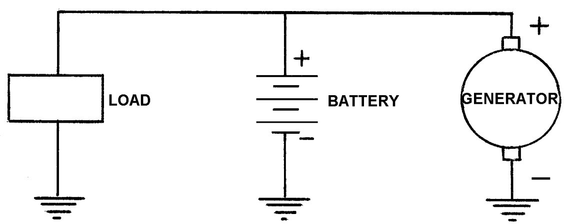
215. Some small turbine-powered aircraft use a generator that doubles as the engine starter motor. Typically the machine has two sets of field windings. During engine start, when the starter generator is acting as a motor, a low-resistance series field winding is used. This permits a high current flow through the field winding to give the high torque needed for turning the engine.

216. Once the engine is operating the series field winding is switched out and a shunt winding energised in its place, so that the machine becomes a shunt-wound DC generator supplying current to the aircraft's electrical system at 28 volts and up to 300 amps.

DC Power Distribution

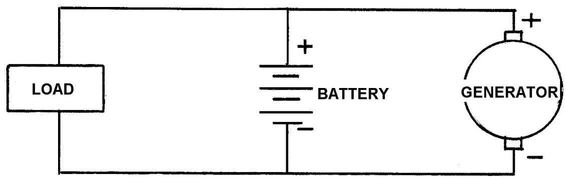
217. In any aircraft, electrical power will be distributed to the various items of electrical equipment by one of two methods.

FIGURE 7-55
Single Pole Distribution System



218. The single pole (unit pole) or 'earth return' system is used on aircraft of metal construction. In this system one wire (the single pole) connects the electrical power supply to the equipment, and the return path from the equipment to the power source is via the aircraft structure itself, as shown at Figure 7-55. The dipole or two-pole system is used on aircraft that are not constructed entirely of conductive materials. Here one wire connects the electrical power supply to the equipment, and the return path from the equipment to the power source is via a second, earth wire, as shown at Figure 7-56.

FIGURE 7-56
Two Pole (Dipole) Distribution System

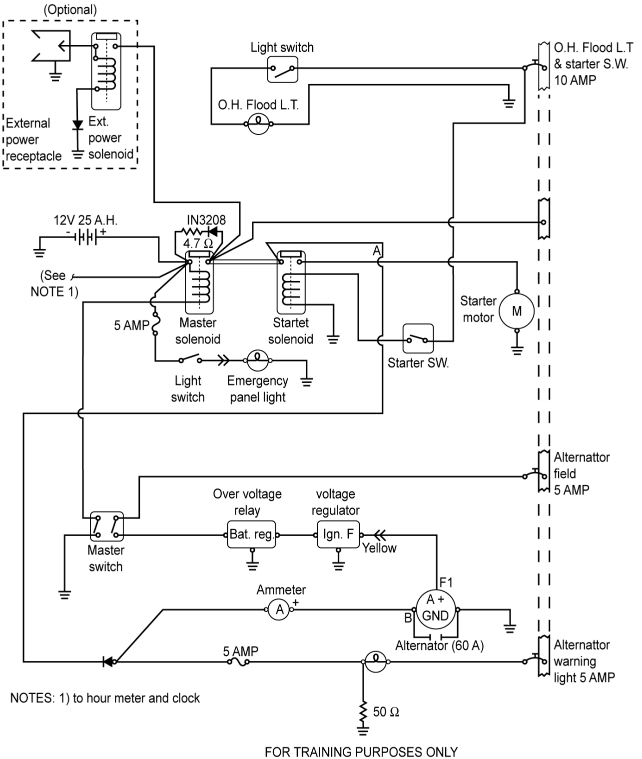


219. All aircraft electrical systems use at least one bus bar, which is simply a copper strip acting as a junction for the generator(s), battery and the various loads. However, distribution is complicated by the need to provide for abnormal conditions such as loss of a power source or faults in the distribution system.

220. In single-engine aircraft it is normal for the alternator and the battery to be connected to a single bus bar, from which the relatively few consumer services are drawn, through fuses or circuit breakers of suitable rating. A simple DC power distribution schematic is illustrated by Figure 7-57. The system consists of a battery circuit, an alternator circuit, an engine starter circuit, a bus bar with circuit breakers and lighting circuits. Not shown are the connection and protection for the radio circuits. High current carrying cables are connected between battery and master solenoid, starter

solenoid and starter motor and between alternator and main bus bar. All the normal loads are supplied by the alternator once on and running. Failure of the alternator output is indicated by a warning light. The loads are fed from the main bus and protected by individual circuit breakers (CB). Any wires that are not protected by CB's should be as short as practical and well insulated. Note that this schematic is a negative earth return system. When the alternator is running and connected to the bus, the ammeter will indicate the electrical load being supplied to the consumers. This system shows the option to incorporate an external DC power connection.

Electrics-DC



Typical single-engine aircraft power distribution systems

221. If the generator fails in flight on a single engined aircraft, the only remaining source of electrical power is the battery. It will be necessary to reduce electrical loading to a minimum and to land as soon as practical. If the generator and battery fail simultaneously a landing should be made as soon as possible. A battery should provide enough power for thirty minutes flying following failure of a generator.

222. Should the presence of fire be detected other than an engine it is most likely to be an electrical problem. The source could be identified by a process of isolating various circuits in turn by use of circuit breakers. On a single engined aircraft it may be necessary to turn off the generator or battery (or both). Once again a landing should be made as soon as possible.

223. Typical requirements of an aircraft electrical system are:

(a) Supply to all equipment must be maintained unless the total power demand exceeds available supply.

(b) System faults (earths, short-circuits, etc) should have the least possible effect on overall system functioning.

(c) A fault on one piece of equipment should not affect the power supply to the remainder of the system.

224. These requirements may be met by parallel operation of generators. The second and third requirements are achieved by arranging for faulty systems to be isolated from the distribution network by means of fuses and/or circuit breakers. The minimum disruption to services is ensured by providing each power source with its own bus bar, and prioritisation of consumer services into three categories; vital, essential and non-essential.

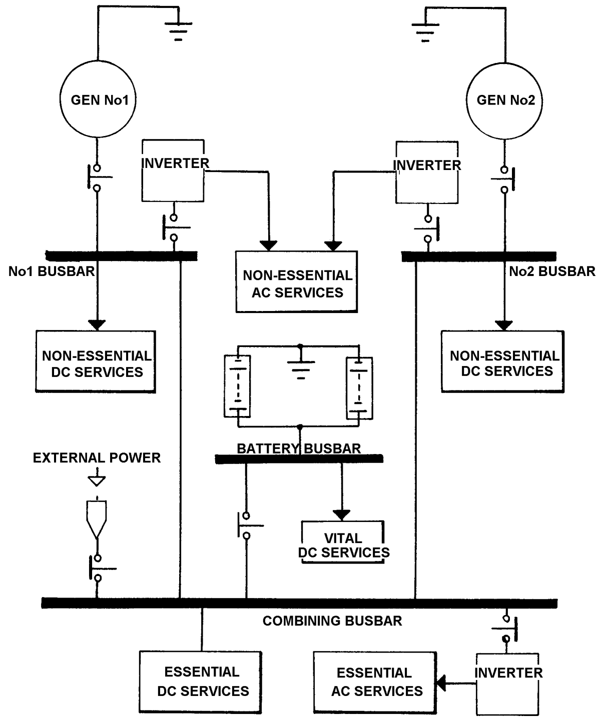
225. Vital consumers are those services required in an emergency when all main power sources, normally the engine-driven generators, are lost. These services, which include emergency lighting and fire detection/protection, are provided from a bus bar connected directly to the aircraft battery supply.

226. Essential consumers are those services necessary for safe flight in an emergency situation. They are connected to a bus bar that can always be supplied either from a generator or the aircraft batteries.

227. Non-Essential Consumers are those services that can be safely disconnected during an in-flight emergency, for purposes of load shedding. These are supplied from the generator bus bars.

228. Figure 7-58 shows a basic twin-engined aircraft bus bar DC distribution system designed to conform to the above categories.

FIGURE 7-58
DC Bus Bar
Distribution



229. Note that the system includes inverters. These convert DC to AC, which is necessary for many of the flight instruments, radios, and navigation systems (avionics in modern jargon). The system is designed for parallel operation of the generators through a combining bus bar. In the event of failure a generator can be isolated, by opening its circuit breaker, and the remaining generator can supply all services.

230. Vital services are taken from the Battery Bus Bar, where power (at 24v) is always available. Essential services are supplied from the Combining Bus Bar, fed by either generator and/or Batteries and therefore not subject to interruption. Non-Essential services are supplied from the Generator Bus Bars and could be subject to interruption, but this is unimportant and makes load shedding easy when necessary. The inverters would, typically, supply 115v AC at a frequency of 400 Hz.

231. Inverters may be either rotary or static. In either case they are used to convert direct current (DC) to alternating current (AC). In the case of the rotary inverter, DC is used to drive a DC motor at constant speed. This in turn drives an alternator (AC generator) to provide alternating current at constant frequency (usually 115 volt, 3 phase AC at 400 Hz).

232. Static inverters, as their name suggests, have no moving parts and achieve the same result electronically. They are much more common in modern aircraft. The circuitry of the static inverter contains such electronic components as diodes, transistors, capacitors and transformers, all of which are explained in later sections. These solid-state components form an oscillator circuit that converts DC input into a 400 Hz constant frequency AC output. Static inverters are usually designed to produce single phase AC.

233. System monitoring typically comprises a single ammeter with switching to select load monitoring of one alternator at a time. A circuit breaker or fuse of suitable rating protects each of the consumer services at their connections to the bus bars and the alternators are typically connected to their bus bars through current limiting fuses.

234. A ground power source can be connected to the bus bar distribution system thus allowing all electrical systems to be powered independently of aircraft battery or generating systems. The source can be either a motorised generating unit or a battery unit. An alternative at some airports is to connect up to a cable from a ground supply routed to the dispersal. It is important that the unit is of correct voltage and polarity and that maximum amperage loading is not exceeded. When connecting or withdrawing the connection it is advisable to switch the unit off first. An aircraft should not be left unattended if a ground power supply is connected and supplying the aircraft systems.

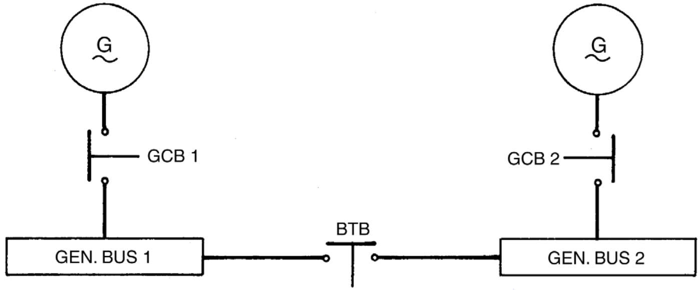
Elementary Switching Circuits

235. In Figure 7-57 it will be noted that the battery master switch, situated in the cockpit, operates a remotely located solenoid-operated switch. Similarly, the engine starter switch operates a remotely located relay. Relay- and solenoid-operated switching circuits are widely used in aircraft electrical distribution systems, not only for remote operation of switches in heavily loaded circuits, but also for sequential switching functions.

236. For example, in non-paralleled (split bus bar) distribution systems in twin-engined aircraft the generators are connected to their own bus bars in the normal way, but the generator buses must never be cross-connected with both generators operating, since the generators are not equipped to operate in parallel. However, should either generator fail it is important that supply to all essential services be maintained without interruption.

237. A simple split bus bar system is shown at Figure 7-59. A switch called a bus tie breaker (BTB) is situated between the two generator buses, which will cross-connect them when it is closed. However, the BTB can only be closed if one or both GCB's are open.

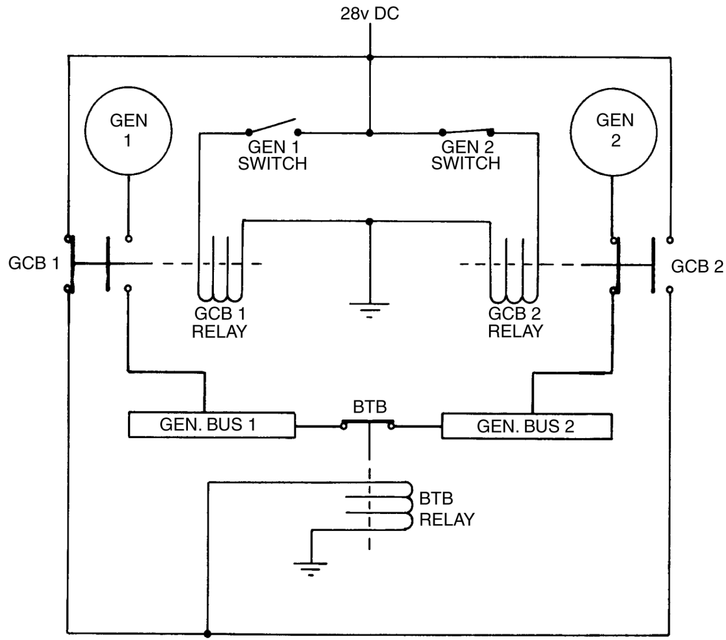
FIGURE 7-59
Split Bus Bar System



238. The bus tie breaker is held open when both generator circuit breakers (GCB) are closed, but it will automatically close if either generator circuit breaker is opened, keeping both bus bars energised and maintaining electrical services. A switching circuit to achieve this is illustrated at Figure 7-60.

FIGURE 7-60

Switching Circuit - Split Bus Bar System



239. The generator switches in the cockpit operate their respective GCB solenoids. When a generator switch is closed a 28-volt DC supply energises the appropriate GCB solenoid, which closes to connect the generator to its bus bar. The GCB solenoids operate double-pole switches, which also control a 28-volt DC supply to the BTB solenoid. If either generator switch is opened its associated GCB solenoid will be de-energised, disconnecting the generator from its bus bar and connecting the 28-volt DC supply to the BTB solenoid, closing the BTB and cross-connecting the generator bus bars.

240. If both generator switches are open, both GCB solenoids will be de-energised and both generators isolated from their buses. The BTB solenoid will be energised, cross-connecting the generator bus bars. This would be the situation with both engines shut down and the aircraft connected to an external power supply.

241. When both generators are operating and connected to their bus bars the BTB solenoid is isolated from its 28-volt supply and the generator bus bars are isolated from each other.

Electrical Consumers

242. Some typical electrical consumers (loads) and their uses are as follows:

(a) **Lighting.** Most aircraft will have some form of lighting. Lights can be powered from the aircraft main generating system (DC or AC) or from aircraft or separate batteries. Amongst these will be position lights (Nav lights), anticollision lights, landing lights, instrument panel lights, warning lights, cabin lights and emergency lights.

(b) **Heating.** Electrical heating circuits are used for de-icing/ anti-icing of airframe, propellor, engine, windscreen, pitot probe and stall and warning devices. Some air conditioning systems use electrical supply to supplement heating.

(c) **Magnetic Devices.** Electrical supply is used to operate various mechanisms using electro magnets i.e. solenoids, relays and switches. Some indication systems are magnetically operated i.e. MIs (dolls eye).

(d) **Avionic Systems.** Aviation electronics (avionics) encompass a variety of electronic systems. Avionic systems in aircraft can include, communication and navigation radio, autopilot, weather radar, inertial navigation and flight management systems.

(e) **Instruments.** Operation of modern aircraft would not be possible without the use of instruments. Large aircraft have instruments which are electrically or electronically operated. Instruments measure pressure, temperature, altitude, velocity and rates of flow. Navigation instruments and auto flight will have a commonality.

Bonding and Screening

243. An aircraft flying through the atmosphere will, to a greater or lesser extent, acquire electro-static charges in the metallic structure of the airframe. If different sections of the airframe acquire different electrical potentials then current will flow between them, and sparking (arcing) across small gaps in the structure is liable to occur. At best this will cause radio interference and at worst it could lead to fires. In order to prevent this the individual parts of the airframe are electrically bonded together, using woven copper wire strips to provide a low resistance path to discharge points on the structure.

244. In flight these points are the **static wick dischargers,** which are copper strips extending from points of static concentration such as trailing edges of primary flying control surfaces.

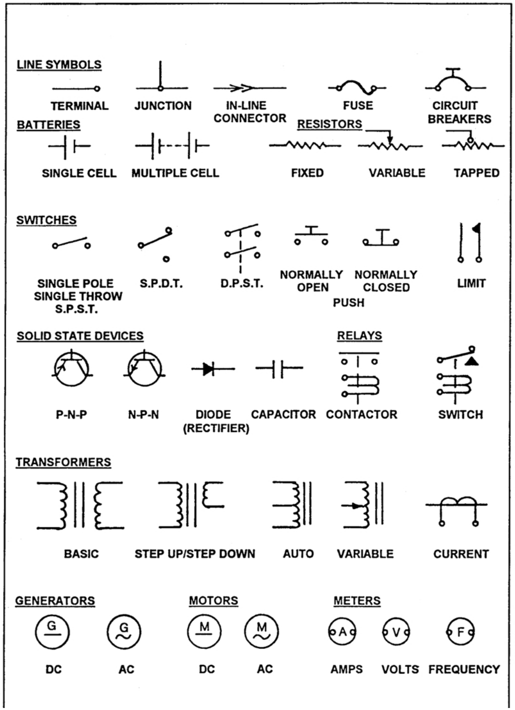
245. On the ground the aircraft is earthed to the tarmac through conducting tyres which have a high proportion of carbon in the tyre material and through which any accumulated static electricity is discharged. Since these tyres are relatively expensive they are only fitted to the smaller wheels, normally the nose wheels.

246. Friction due to the flow of liquid in a pipe can also produce electro-static charges. This accumulation of static is particularly marked during re-fuelling when the atmosphere is dry, especially in pressure re-fuelling when the rate of flow may be 1,000 gallons per minute or more. It is therefore essential that before pressure re-fuelling commences the bowser and aircraft are connected electrically (bonded) to each other, and that both are bonded to earth. In overwing re-fuelling a further requirement is that the re-fuelling nozzle should be bonded to the aircraft tank filler pipe, since the static electricity may be induced in the fuel flow as it leaves the nozzle.

247. Screening is incorporated on aircraft to prevent radio interference as a result of sparking in electrical components. The radio interference can be suppressed by fitting suppressors in the cable attached to any source of sparking, and by installing a metal sheath around the cables. Ignition systems, DC generator and motor commutators, and indeed any equipment making and breaking a circuit (especially at frequencies in excess of 10Hz), need to be either sheathed or suppressed.

248. Suppression is usually achieved by connecting a number of capacitors across the source of interference, so that they provide a low resistance path for the stray voltages induced by the fluctuating magnetic fields associated with interruptions of current flow.

FIGURE 7-61


LINE SYMBOLS
TERMINAL
JUNCTION
IN-LINE CONNECTOR
FUSE
CIRCUIT BREAKERS
BATTERIES
SINGLE CELL
MULTIPLE CELL
RESISTORS
FIXED
VARIABLE
TAPPED
SWITCHES
SINGLE POLE SINGLE THROW S.P.S.T.
S.P.D.T.
D.P.S.T.
NORMALLY OPEN
NORMALLY CLOSED
PUSH
LIMIT
SOLID STATE DEVICES
P-N-P
N-P-N
DIODE (RECTIFIER)
CAPACITOR
RELAYS
CONTACTOR
SWITCH
TRANSFORMERS
BASIC
STEP UP/STEP DOWN
AUTO
VARIABLE
CURRENT
GENERATORS
G
DC
G
AC
MOTORS
M
DC
M
AC
METERS
A
AMPS
V
VOLTS
F
FREQUENCY

Electrics-AC

Electrics-AC

Alternating Current (AC)

1. Virtually all large modern commercial aircraft use alternating current power distribution systems. The large power requirements of such aircraft would result in very large current flows if the low voltage (usually 28v) of DC distribution systems were used. Because it is very easy to change voltages with AC it can be generated and distributed at relatively high voltage (usually 115v AC) and reduced to lower voltage for conversion to DC where necessary. Avionic systems incorporating components such as transistors, transformers, diodes and capacitors require AC power for their operation. A high percentage of aircraft lighting is also AC powered. Even in light aircraft some alternating current is used, produced by inverting DC. Consequently, the study of alternating current is an essential requirement for pilots.

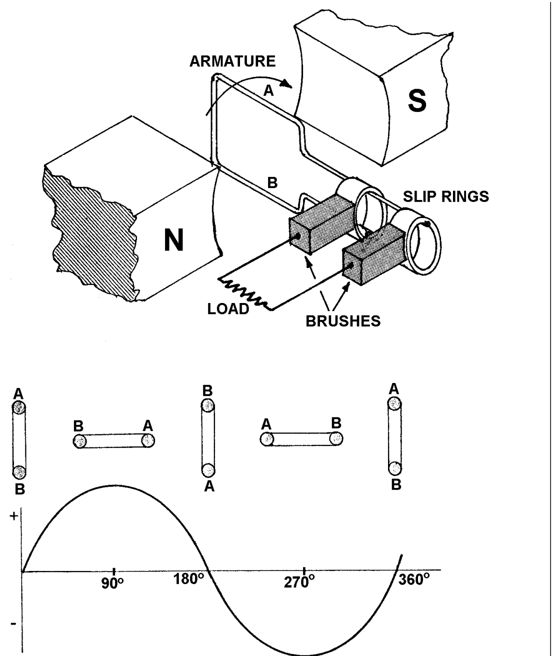
Alternating Current

2. Alternating current is defined as current flow that periodically changes in direction and continuously changes in magnitude. As has been shown in the chapter covering generation of electricity the emf produced in the armature of a generator changes in electrical polarity every half revolution of the armature.

3. Since current (conventionally) flows from positive to negative, this means that the current flow reverses in direction every half revolution, or half cycle. The magnitude of the voltage (emf) induced in the armature, and the resulting magnitude of current flow in a supplied circuit, varies as the sine of the degree of rotation of the armature, from 0° to 360°.

4. Thus, the magnitude of voltage and current, when shown graphically follows a sine curve, or sine wave. This is illustrated at Figure 8-1.

FIGURE 8-1
Production of AC Sine Wave



5. Each complete sine curve represents one revolution, or cycle, of the generator armature. One cycle of alternating current covers a period in which the voltage and/or current increases from zero to maximum value in one direction, falls back to zero and then repeats the process with opposite polarity and direction of current flow. The magnitude of the peak voltage/current rise is known as the amplitude of the cycle. The rate at which the cycles of AC are repeated, or cycles per unit time, is known as the frequency of the AC

Frequency

6. AC frequency is the number of cycles per second and is measured in units called hertz (Hz). In aircraft AC systems the standard frequency is 400 Hz, or 400 cycles per second. Since each cycle of AC is the result of one revolution of the generator armature (see Figure 8-1), it follows that the greater the speed of rotation, the higher the output AC frequency. For a ‘two-pole’ generator as shown (one North Pole, one South Pole) the armature would clearly need to rotate at 400 revolutions per second to produce a frequency of 400 Hz.

7. Machine rotary speed is conventionally measured in revolutions per minute (rpm) and, since there are 60 seconds per minute the rpm of the armature would need to be:

In other words:

$$400 \times 60 = 24,000 \text{ rpm}$$

Therefore, by transposition:

$$F = \frac{N}{60} \text{ for a 2-pole generator}$$

8. Most practical AC generators use more than two magnetic poles, since the more the 'pairs' of poles (North and South) the lower the rotary speed of the armature necessary for a given output frequency. If the generator shown at Figure 8-1 had two north poles and two south poles, spaced evenly around the armature in the sequence N, S, N, S, the rotary speed required for an output frequency of 400 Hz would be halved. Consequently, the formula for calculating AC frequency must include the number of pairs of poles (P) and is given as:

$$F = \frac{N \times P}{60}$$

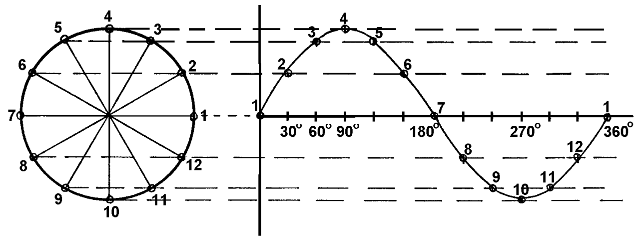
9. In some cases the number of poles is given. Since this is obviously twice the number of pole pairs the formula then becomes:

$$F = \frac{N \times P}{60 \times 2} \text{ or } F = \frac{NP}{120}$$

Sine Wave Format

10. The geometric production of a sine curve is illustrated at Figure 8-1 and from this it can be seen how it is that the output of an AC generator, produced by a rotating armature, varies sinusoidally with time.

FIGURE 8-2
Geometric Production of Sine Curve



11. As has been shown already, most practical AC generators are constructed with a rotating magnetic field (the rotor) and a stationary armature winding in the casing (stator).

12. The advantages of alternators over DC generators as primary supply sources are that alternators (AC generators) are lighter and have a simpler brush assembly, requiring less maintenance. Also, their output voltage remains more constant over the operating speed range.

13. The main disadvantage lies in the fact that, unless a complex additional exciter mechanism is included in the AC generator construction, it is reliant upon separate excitation from the aircraft battery.

14. Alternating current has some significant advantages over direct current when it comes to distribution. It will be explained in the text covering transformers how its voltage can be readily transformed either up or down. Thus, for a given power, load current can be decreased if distribution voltage is increased and power losses in transmission significantly reduced as a result. Conducting cable diameter can therefore be reduced, saving in weight.

15. The induced voltages caused by AC can lead to stray currents, radio interference and extra generator loads, but these disadvantages are small compared to the weight saving advantages, especially in large aircraft with high power requirements.

RMS Equivalent

16. In order to determine the power available from an electrical supply it is necessary to know the current and voltage of the supply. With DC this is straightforward since both are easily measured. With AC, however, the value of current is continuously changing sinusoidally and it becomes necessary to determine a mean effective current.

17. It has been found that the square root of the mean value of the square of the current gives a value that has the same heating effect as an equivalent direct current. This value is 0.707 ($1/\sqrt{2}$) of the peak value of alternating current. Because of the method by which it is determined it is known as the root mean square (rms) equivalent value. The mean voltage can be determined in the same way.

18. The current and voltage quoted for an AC system is the rms equivalent value. Given that:

$$\text{RMS Equiv. Value} = \frac{\text{Peak Value} \times 1}{\sqrt{2}}$$

$$(\text{Peak Value} = \text{RMS Equiv. Value}) = \sqrt{2(1.414)}$$

19. Thus, for an AC circuit with an RMS voltage of 115v, peak voltage = 115 x 1.414 = 163v.

Opposition to Current Flow

20. Only the resistance of the circuit and its components opposes the current flow in a DC circuit. In an AC circuit, however, current flow may be opposed not only by resistance, but also by inductance and capacitance.

Inductance

21. Whenever there is a flow of DC or AC current a magnetic field is set up around the conductor. If the conductor is wound into a coil the electromagnetic effect can be used to magnetise (DC), or de-magnetise (AC) magnetic materials. This is the principle that is applied to solenoids, relays and so on. These devices use DC. By varying the strength of DC it is possible to control the strength of a magnetic field. This same principle is used to control the output of a generator or the torque of a DC motor.

22. Lenz's Law states that induced emf acts to circulate a current in a direction that opposes the change of flux which causes the emf. In a conductor carrying alternating current, the current is continuously varying in magnitude, which produces a magnetic flux continuously varying in strength. In effect, a moving magnetic field is produced; the rate of movement of which depends upon the frequency of the alternating current.

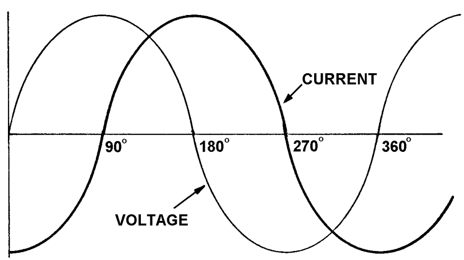
23. This moving field induces an emf in the current-carrying circuit and that emf circulates a current in a direction opposing the change of flux - in other words, opposing the normal current flow. This is known as self-inductance and a circuit is said to have the property of self-inductance when a change in the current in that circuit causes an emf to be induced in the circuit. All AC circuits are self-inductive. The symbol for self-inductance is L. The unit of self-inductance is the Henry (H) which is defined as follows:

24. A circuit has a self-inductance of one Henry if an emf of 1 volt is induced in the circuit by a rate of change of current of one ampere per second.

25. Because this emf produces a current flow in opposition to the original, or main, current flow it is known as back emf. A straight conductor has inductance, but this is usually negligible. As was shown in the chapter covering electro-magnets, when a conductor is formed into a coil a much stronger magnetic field is produced. Coils of wire in AC circuits have an inductance that produces a significant back emf, opposing the applied current.

26. According to Lenz's Law, induced emf is greatest when current change is greatest. With AC, rate of current change is greatest when it is passing through zero (changing direction) and least when at its peak value. Hence induced emf (voltage) is greatest when current is zero and least when current is maximum. This is illustrated at Figure 8-2, from which it will be seen that the effect of inductance in an AC circuit is to cause current to lag voltage.

FIGURE 8-3
Effect of Inductance



27. In a purely inductive circuit, current would lag voltage by 90° as shown and this is known as phase quadrature. Pure inductive circuits are non-existent, since there is always resistance as well, and the phase lag between current and voltage is less than 90°.

Inductive Reactance

28. The effect of inductance in an AC circuit is known as inductive reactance and is given the symbol X_L. It is the actual opposition to current flow created by inductors, whereas inductance is the ability of an inductor to oppose changes in current flow. Because it impedes current flow inductive reactance is measured in ohms. It is directly proportional to the inductance of the circuit and the frequency of the alternating current. The formula for inductive reactance is:

$$X_L = 2\pi FL \text{ ohms}$$

where: F = Frequency (Hz) L = Inductance (H)

29. In DC circuits the effect of inductive reactance is only noticeable when the current is switched on or off. In AC circuits inductive reactance can be a considerable problem, especially if frequency is allowed to vary. The variation in opposition to current flow, due to variation in frequency, causes variations in voltage to which avionic equipment is highly sensitive. Frequencies above, and especially below, design frequency can lead to overheating in inductive components, because of the increase in current flow caused.

The equation for Ohm's Law states that:

$$I = \frac{V}{R}$$

30. If we substitute X_L for R (both measured in ohms, remember) it is apparent that a reduction in X_L at constant voltage will cause an increase in I (current). An increase in current has a heating effect upon any conductor.

31. Since $X_L = 2\pi F.L$, it follows that a decrease in frequency will cause an increase in current flow, provided that voltage remains constant (which, of course, the voltage regulator ensures it does).

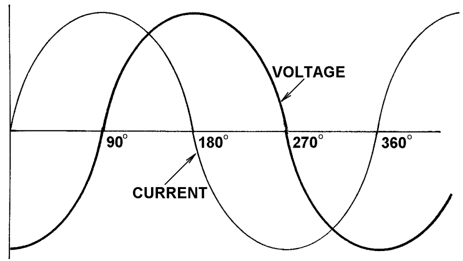
32. Clearly, since distribution voltage is maintained at a constant value, a decrease in AC frequency will cause an increase in current flow. This can cause overheating in inductive devices such as transformers, motors and generators.

Capacitance

33. The basic principle of the capacitor was described in the section dealing with the Basic Principles of DC Electricity.

34. When a capacitor is placed in an AC circuit it has the opposite effect to that of an inductor, in that it causes the current to lead the voltage, as shown at Figure 8-3.

FIGURE 8-4
Effect of Capacitance



35. As voltage rises the capacitor becomes charged and the dielectric stress set up in the capacitor opposes current flow, so current flow decreases. By the time voltage has reached its peak value the capacitor is fully charged and current flow is zero. As supply voltage begins to fall current flows out of the capacitor in the opposite direction because the capacitor stored voltage is higher than the supply voltage.

36. By the time supply voltage has fallen to zero, current flow is maximum because there is now no opposition to current flow. If there were no resistance in the circuit, current would lead voltage by 90° (phase quadrature). Of course, all circuits are resistive to some extent so this theoretical condition is never reached.

37. To help the student remember the fact that current leads voltage in a capacitive circuit and lags voltage in an inductive circuit, the mnemonic CIVIL is helpful, where:

(a) C = Capacitive

(b) I = Current

(c) V = Voltage

(d) L = Inductive

38. The unit of capacitance is the farad (F). Since this is an extremely large unit capacitors usually have values expressed in microfarads, nanofarads or picofarads. One farad is defined as the capacitance present when one volt will store one coulomb of electrical energy in the capacitor.

Capacitive Reactance

39. The effect of capacitance in an AC circuit is called capacitive reactance. Because it opposes current flow in a circuit it is measured in ohms. The symbol for capacitive reactance is X_C, which represents the actual opposition to current flow caused by capacitors in an AC circuit. The formula for capacitive reactance is:

$$X_C = \frac{1}{2\pi FC} \text{ ohm}$$

where: F = Frequency (Hz). C = Capacitance (Farads)

As with inductive reactance, a variation in frequency of the AC supply will affect the current flow in the circuit.

From Ohm's Law:

$$I = \frac{V}{R}$$

therefore, in a capacitive circuit:

$$I = \frac{V}{X_C} = V \times 2\pi FC$$

40. From this it will be seen that an increase in frequency, at constant voltage and capacitance, will cause an increase in current flow.

Resonance

41. A resonant circuit is one in which the capacitive reactance X_C is equal to the inductive reactance X_L.

Impedance

42. In an AC circuit the combined effect of inductive reactance, capacitive reactance and resistance is known as impedance and has the symbol Z. However, impedance is not the sum of the three. In the first place, inductive reactance and capacitive reactance have opposite effects, as has been shown. Total reactance is the algebraic sum of the two:

$$X_T = X_L + (-X_C)$$

43. Secondly, resistance does not cause a phase difference between current and voltage so its effect will be 90° ahead of inductance and 90° behind capacitance. Consequently, impedance must be the vector sum of resistance and reactance. A vector sum is illustrated at Figure 8-4.

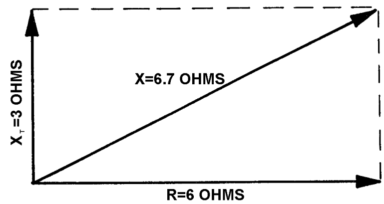


FIGURE 8-5
Vector of Impedance

Power in AC Circuits

44. Reactance and reactive load are the terms applied to AC circuits that include inductive and/or capacitive components, causing current flow to be out of phase with voltage (where there is a back emf, or reactance, opposing current flow). All circuits will also contain some purely resistive components and these, too, oppose the flow of current. The total opposition to current flow, caused by inductance/capacitance and resistance, is known as impedance.

45. The work done by the generator in overcoming resistance represents useful or real power, since resistance is due to the load of the circuit components. The work done in overcoming reactance is wasted effort and is known as reactive power. The total work done (power generated) will have to be that necessary to overcome both the real (effective or active) system load and the reactive (idle or wattless) load. This is known as the apparent power. The ratio of effective power to apparent power gives the power factor (PF) of the circuit being supplied.

46. As we have already seen, the amount of inductive and capacitive reactance in a circuit will determine the extent by which circuit voltage lags or leads (is out of phase with) current. If voltage and current are in phase with each other, power available for useful work will be maximum, since power (watts) = volts (V) x amps (I).

47. The more voltage and current are out-of-phase the greater the reactive load and the less the useful power available. Thus, it is the phase relationship of circuit voltage and current that determines the circuit power factor. The cosine of the phase angle (ϕ) between voltage and current is equal to the power factor:

$$PF = \cos\phi = \frac{\text{Real Power}}{\text{Apparent Power}}$$

48. Knowledge of the power factor is necessary when determining the power rating of the generator(s) needed to meet the real and reactive loads of the system to be supplied. In order to distinguish apparent power, which is the total power consumed by the entire AC circuit, from real or reactive power, apparent power is quoted in units of kilovolt amperes (kVA). AC generators are rated in kVA rather than kilowatts (kW), which is the unit of real power. The reactive load, sometimes displayed in larger aircraft, is measured in kilovolt-amperes reactive (kVAR).

49. Real Power is the power resulting purely from the current flow due to the resistance of the AC circuit, and is given by the formula:

$$\text{Power (Watts)} = I^2R$$

50. It is the power consumed usefully in the circuit in the form of work or heat and is measured in Watts (W) or kilowatts (kW).

51. Reactive Power is the power resulting from the current flow due to the total reactance (inductive reactance – capacitive reactance) of the AC circuit and is given by the formula:

$$\text{Power (Kilovolt Amps Reactive)} = I^2(X_L - X_C)$$

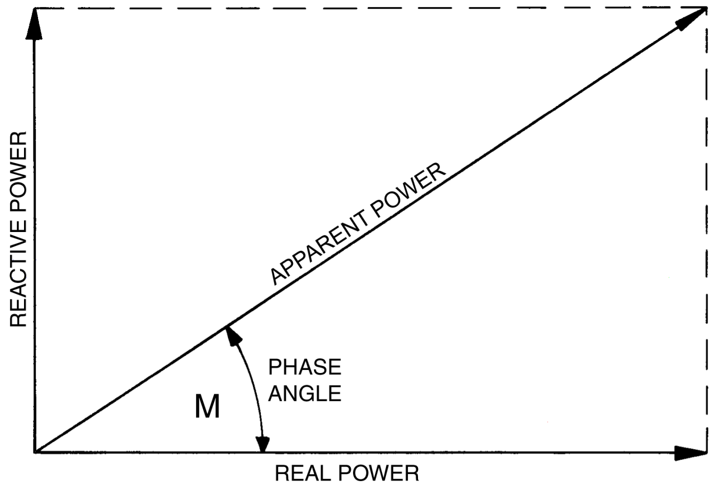
52. It is measured in volt amps reactive (VAR) or kilovolt amps reactive (kVAR).

53. Apparent Power is the power consumed by an AC circuit that contains reactance as well as resistance. It is the power resulting from the current flow due to the impedance of the circuit.

54. It is measured in volt amps (VA) or kilovolt amps (kVA).

55. When real and reactive power are plotted respectively as the horizontal and vertical components of a vector diagram, as at Figure 8-5 below, apparent power is the vector sum of the two.

FIGURE 8-6
AC Power Vector



56. If the phase angle (ϕ) is known, the Power Factor can be found, since:

$$PF = Cos\phi$$

57. And from this Real Power can be found if Apparent Power is known, or vice versa, because:

$$PF = \frac{\text{Real Power}}{\text{Apparent Power}}$$

58. Reactive power can be found either by using Pythagoras' Theorem, or by triangulation.

EXAMPLE 8-1

EXAMPLE

The phase angle (ϕ) between current and voltage in an AC system is 30°. When the apparent power consumed by the system is 540 kVA, find the real and reactive power.

SOLUTION

Using Pythagoras:

$$PF = \cos 30° = 0.866 \qquad PF = \frac{\text{Real Power}}{\text{Apparent power}} \therefore \text{Real Power} = \text{Apparent Power} \times PF$$

$$= 540\text{kva} \times 0.866$$

$$= 467.6 \text{ kw}$$

$$(\text{Reactive Power})^2 = (\text{Apparent Power})^2 - (\text{Real Power})^2$$

$$\therefore (\text{Reactive Power})^2 = 540^2 - 467.6^2 = 72,950.24$$

$$\therefore \text{Reactive Power} = \sqrt{72,950.24} = 270 \text{ kVAR}$$

Using triangulation:

$$\text{Reactive Power} = \text{Apparent Power} \times \text{Sin}\phi$$

$$= 540 \times 0.5$$

$$= 270 \text{ kVAR}$$

The observant student will have noticed that the sum of real and reactive power is greater than apparent power.

$$467.6 + 270 = 737.6$$

This is because, when considering the effect of resistance in an AC circuit (real power) the voltage and current are in phase, but when considering the effect of reactance (reactive power), the voltage and current are out of phase. Hence the total effect must take the phase difference into account and the total, or apparent power must be the vector sum of the two.

AC circuits are rarely purely resistive, but since inductive and capacitive effects are opposite they can be used to compensate for each other's reactance. A discharging capacitor can compensate for the current lag caused by an inductor, or an inductor can overcome the current lead caused by capacitor discharge. When an exact balance between the two is achieved a state of resonance is said to exist. Under these circumstances current will be in phase with voltage and the circuit appears to be purely resistive.

Most aircraft AC equipment employs electro-magnetic coils, which are inductive. To cancel their reactance, capacitors are introduced. It is never possible to completely achieve a state of resonance and any inductive reactance remaining will result in additional current (load) being carried wastefully by the alternators and distribution cables

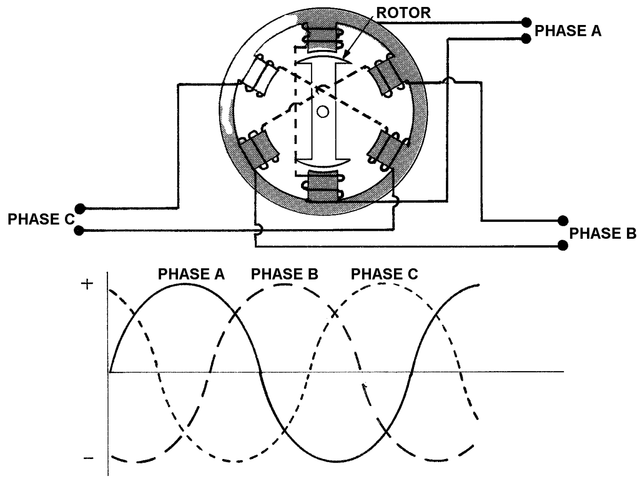
Single and Three Phase Supplies

59. AC electricity is used to power most of the lighting on an aircraft, plus low-powered AC motors. It is also converted to DC for battery charging, relay operation, powering DC motors and other DC loads. This type of equipment operates on a single wire supply and earth-return distribution system from the alternator and is known as single-phase operation.

60. However, heavy duty AC motors operate on a rotating magnetic field principle, which requires three AC supplies at equally spaced time intervals. Many avionic components also operate on similar principles, so it is normal for the three AC supplies, or phases, to be produced by a three-phase, or polyphase, alternator.

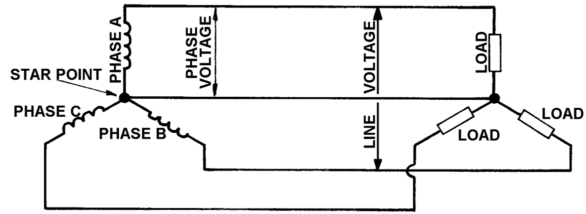
61. Three phase alternators have three equally spaced and independent armature windings in the stator casing, to give the required outputs at 120° intervals. Each winding operates like a single-phase alternator, and can be used to supply single-phase loads. Alternatively, the outputs can be used together for three phase (or two phase) loads. A three-phase system is shown diagrammatically at Figure 8-6.

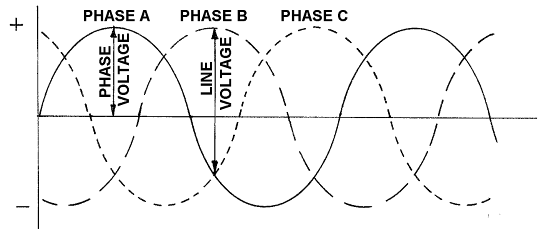
FIGURE 8-7
Three Phase System



62. In practice each of the three phases is not usually brought out to individual pairs of terminals as at Figure 8-6, since this would involve unnecessarily excessive use of cable. Instead, the three-phases are interconnected by means of either a star or delta connection as shown at Figure 8-7 and Figure 8-8.

FIGURE 8-8
3 Phase Star Connection





63. Figure 8-7 illustrates the star or wye connection, which is the norm for aircraft AC generators. One end of each phase winding is connected to a common junction known as the star, or neutral point. Also connected to this point is the common return, or neutral line from the single-phase AC loads. The other end of each phase winding is connected to a distribution, or live line.

64. The live lines supply the single-phase loads, so it will be seen from Figure 8-7 that the single-phase loads are connected in parallel with the generator phase windings. The voltage across these loads is known as the phase voltage and it is the voltage between live line and neutral. It is typically 115v (rms).

65. Between any pair of lines there are two generator phase windings connected, but these phase voltages are in opposition and out-of-phase, so line-to-line, or line voltage is equal to phase voltage x $\sqrt{3}$. Since $\sqrt{3} = 1.73$, if phase voltage were 115v, line voltage would be 115 x 1.73, or 200v. In a star connected or star wound generator distribution system, line and phase currents are equal, because the current in any phase load is clearly flowing through the live line connected to it.

66. The majority of the loads on the generator are single-phase loads, so they are connected between a line and neutral. In an aircraft system the designer attempts to ensure that, in normal operation, the total aircraft loads are approximately equally shared between the three-phases.

67. When the phase loads are not equal there will be a small current flow in the return line, equal to the phasor sum of the current flows in the live lines. For example, if phase A load was 100 amps, phase B load 120 amps and phase C load 110 amps, the phasor sum of these current flows would be only 15 amps. This is because the three phases are at 120° to each other so the currents flowing through the each of phase loads are in different directions at any instant of time.

68. If a current flow from line to neutral of 100 amps is assumed in phase A loads, the current flows in phase B and C loads will be less than full load (see Figure 8-7) and will be from neutral to line. The values and direction of these currents at this instant can be shown mathematically as follows:

$$\text{Phase A } 100\text{a} \times \text{Cos}0^\circ = +100\text{a}$$

$$\text{Phase B } 120\text{a} \times \text{Cos}120^\circ = -60\text{a}$$

$$\text{Phase C } 110\text{a} \times \text{Cos}240^\circ = -55\text{a}$$

69. Phasor Sum = 100 + (-60) + (-55) = -15amps or 15amps from the star point (at this instant). Students will appreciate that this is a trigonometrical solution and hence the reason for using the cosine of the angles. The problem could equally have been solved by geometry.

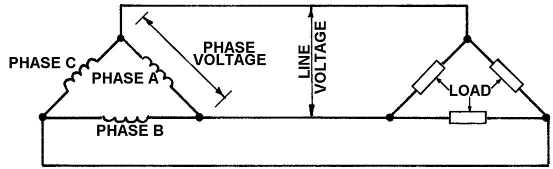
70. If the three individual phase loads were always equal, the current flow in the neutral return would always be zero and the return line could be dispensed with. In an aircraft distribution system however, the loads in the three-phases will vary as items of equipment are switched on and off, so a neutral return is necessary.

71. Should an internal break lead to open circuiting of one phase of a star wound three-phase generator the voltage across that phase will fall to zero and there will be no current flow in that phase. The phasor sum of the remaining currents will be higher than before, hence there will be an increase of current flow in the neutral return. This is liable to cause a voltage drop in the two remaining phases, the magnitude of which will depend upon the magnitude of the phase loads.

72. Short-circuiting between generator phases (line-to-line) or between a phase and neutral (line-to-earth) will cause very high current flow and the voltage will drop to near zero. The high current flow causes overheating of the stator windings.

73. As previously stated, if it can be assured the loads on the three-phases will always be equal, there is no need for a fourth, return, wire. In these circumstances an alternative form of three-phase connection called a delta, or mesh connection can be used.

FIGURE 8-9
3 Phase Delta Connection



74. The delta connection is illustrated at Figure 8-8, from which it can be seen that voltage across one phase will be the same as the voltage between the lines connected to the opposite ends of that phase. In other words, line voltage is equal to phase voltage. In such a connection, line current is equal to phase current x √3. This is because each line is connected to the ends of two phases so, given a balanced load situation the current flow in a line must be greater than the current flow in one phase (phase current).

AC Generators (Alternators)

75. There are several advantages of using an alternator (AC system) compared to a DC generator. The alternator has a much better power/weight ratio and can produce maximum circuit voltage requirement at low rpm. Because there is no requirement for commutation the problem of brush sparking is eliminated and in addition a wider range of current supply is available for consumer devices.

76. AC generators in aircraft conform generally to two basic types, depending upon the system to be served. These two types are frequency-wild and frequency-controlled generators. In either case the generator is normally a three-phase machine producing single-phase output voltage at 115v, and three-phase voltage at 200v.

77. As we have already seen when discussing alternators and DC generators, output voltage is controlled by a voltage regulator that controls the strength of the field excitation, by varying the field current.

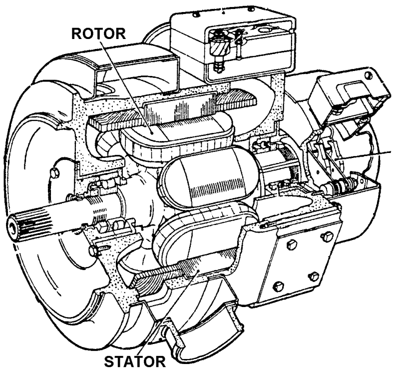
78. Simple resistive circuits, such as electrical de-icing systems are not frequency sensitive. These can be supplied with AC from a generator whose speed is uncontrolled and whose output frequency is therefore variable. The generators used to supply AC at variable frequency are known as frequency-wild generators.

79. Most AC equipment requires the frequency of the supply to be at a constant value. In order to achieve this the rotational speed of the generator must be maintained constant, regardless of the speed of the engine driving it. The generators supplying such a system are known as constant speed, or frequency-controlled generators.

80. Frequency-wild generators are usually only used in aircraft in which the primary distribution system is DC. The AC output from the frequency wild AC generator is only used to power specific resistive systems, such as engine and propeller de-icing. For distribution to the bus bars the output of the generator is transformed to a lower voltage, typically 28v, and rectified to DC.

81. Frequency-wild generators are separately excited, DC excitation current being fed through brushes and slip rings to the series-connected field windings on the 6-pole rotor. Generator output is typically three-phase, 200v, 22 kVA over a frequency range of 280 Hz to 400 Hz. The generator is cooled by ram air. A diagram of a typical frequency-wild generator is at Figure 8-9.

FIGURE 8-10
Frequency Wild Generator

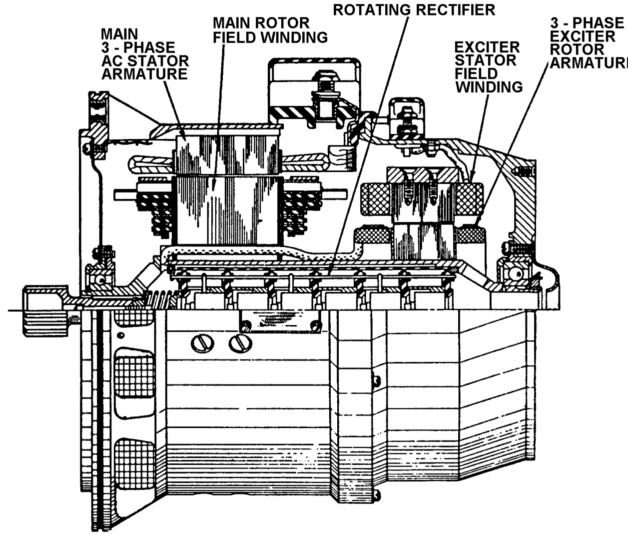


82. Frequency-controlled generators require complex constant speed drive units (CSDU's) in the transmission train between engine and generator. These are usually hydro-mechanical variable ratio drives that can be remotely disconnected in flight in the event of generator or transmission failure. They can usually only be re-connected on the ground with the associated engine shut down.

83. Alternators (AC generators) usually consist of a rotating field, or fields within a stationary armature. In order to avoid confusion the term rotor and stator are used instead of field and armature. This type of arrangement can be termed an internal pole machine. The magnetic field is created by an electro-magnet, which must be supplied with direct current for field excitation. Since the field winding is wound around a rotating core (the rotor) the field current, from an external source such as a DC bus bar, must be supplied to the field windings via brushes and slip rings.

84. Most large, frequency-controlled AC generators are high-power machines producing very large current flows at full load. To eliminate the losses, which inevitably occur with slip rings and brushes, they are brushless machines. The principle of the brushless AC generator is illustrated at Figure 8-10.

FIGURE 8-11
Brushless AC Generator



85. The generator consists of three major components. These are (1) an exciter-generator that produces the field current for the main generator, (2) a rotating rectifier assembly that converts the output of the exciter-generator to DC, and (3) the main generator, producing AC at constant (400 Hz) frequency for supply to the generator bus bar.

86. The rotor of the generator carries the exciter generator armature at one end and the main generator field at the other. Mounted on the rotor, between the two, is the rotating rectifier assembly supplied with AC from the exciter armature and converting this to DC and supplying it to the main generator field.

87. The exciter armature rotates in a magnetic field created by the exciter stator field windings. These are supplied with regulated DC from an external source. The main generator field rotates within the three-phase stator (armature) windings, which are connected to the generator output terminals.

88. The number of poles in such a generator is usually eight, and output frequency is 400 Hz.

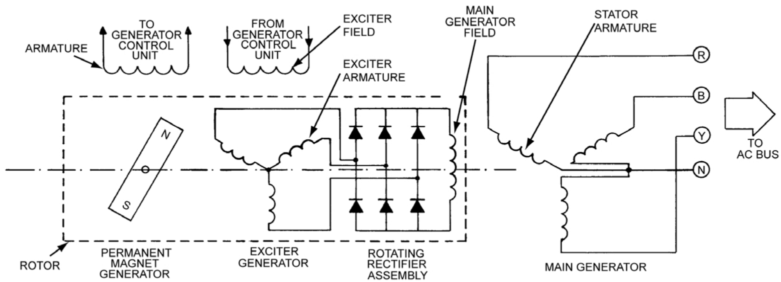
$$\text{Given that frequency (F)} = \frac{\text{rpm} \times \text{No. of poles}}{120}$$

$$\text{by transposition rpm} = \frac{\text{F} \times 120}{\text{No. of poles}} = \frac{400 \times 120}{8} = 6000\text{rpm}$$

89. The constant speed drive unit (CSDU) for such a generator would therefore have to maintain a constant drive speed of 6000 rpm. This is also known as synchronous speed.

90. An alternative type of brushless AC generator that is completely self-contained and requires no external DC supply for its exciter-generator is shown in diagrammatic form at Figure 8-11.

FIGURE 8-12
Self Contained Brushless AC Generator



91. This comprises three generators on the same shaft. The first is a permanent magnet generator (PMG), consisting of a rotating permanent magnet that induces alternating current in a stationary armature. This is rectified in the generator control unit (GCU) and the DC is supplied to the stationary field winding of the exciter-generator, where it induces alternating current in the rotating armature of the exciter-generator.

92. The exciter-generator output is fed to a rectifier mounted on the rotating shaft, where it is converted to DC for supply to the rotating field windings of the main generator. Here it induces alternating current at 400 Hz in the stationary armature of the main generator. The output voltage is regulated at 115v single-phase, 200v three-phase, by the GCU, which regulates the field current to the exciter-generator field winding.

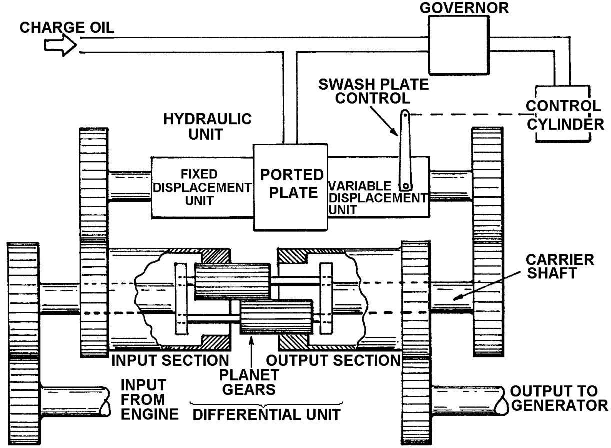
Constant Speed Drive System (CSDS)

93. Constant frequency AC is essential to parallel operation of AC generators, which is necessary in three- and four-engined aircraft. Even where parallel operation is not required, many AC components are frequency sensitive and must be supplied with constant frequency AC. Inductive components will overheat if frequency is too low and most avionic components will only function properly within a narrow frequency range.

94. The output frequency of an AC generator is totally dependent upon its speed of rotation; in order to maintain constant output frequency it is necessary to maintain constant speed of rotation of the generator. Since the engine (through the accessories gearbox) drives the generator, and engine speed is variable from idling to maximum rpm, it is necessary to introduce a constant speed drive system between gearbox and generator. The types of CSDUs range from a pneumatically operated combined air motor/starter type to the oil controlled system utilising either the wobble pump or variable speed arrangement.

95. Figure 8-12 shows a schematic block diagram of a hydro-mechanical constant speed drive. It is a variable ratio drive comprising three basic components, a fixed displacement hydraulic unit, a variable displacement hydraulic unit and a differential gear unit. Hydraulic oil for operation of the system is supplied by a CSDU charge pump with its own reservoir and oil cooler.

FIGURE 8-13
Hydro-Mechanical Constant Speed Drive



96. The variable and fixed hydraulic units are swash plate pumps/motors similar to that described in the Hydraulics section of the course. They are mechanically independent of each other, connected only by a stationary ported plate that permits a flow of oil from one to the other.

97. The differential gear unit consists of a through carrier shaft with a gear at each end, one driven by the engine and the other meshing with the variable displacement hydraulic unit drive. Thus the variable unit rotates at a speed proportional to engine speed.

98. Carried on the carrier shaft are two planet gears which are in mesh with each other and also mesh internally with the two separate sections of the differential unit. This permits the two sections to rotate at different speeds. Each section has an external gear. One is in mesh with the fixed displacement hydraulic unit drive, the other meshes with the output drive to the generator.

99. The variable displacement hydraulic unit has a variable angle swash plate controlled through a control cylinder by a governor sensitive to generator speed. Depending upon the swash plate angle the variable unit will either pump oil to, or accept oil from, the fixed unit.

100. When the input speed from the engine is the same as the required output speed for the generator, the governor, via the control cylinder, positions the variable displacement unit swash plate such that it is neither pumping oil to, nor accepting oil from, the fixed displacement unit. Consequently the fixed unit remains stationary and there is direct drive from the engine, through the differential unit carrier shaft and the variable hydraulic unit, to the generator drive.

101. When the input speed from the engine increases there is an initial tendency for the generator to overspeed. This is sensed by the governor, which adjusts the variable displacement unit swash plate such that the unit will accept oil from the fixed displacement unit. The fixed unit is forced to rotate in the same direction as the variable unit, so the input section of the differential unit rotates in the same direction as the carrier shaft. This reduces the speed of rotation of the planet gears and the output section of the differential unit, reducing generator drive speed relative to engine input speed, to maintain a constant speed. This is known as the underdrive condition.

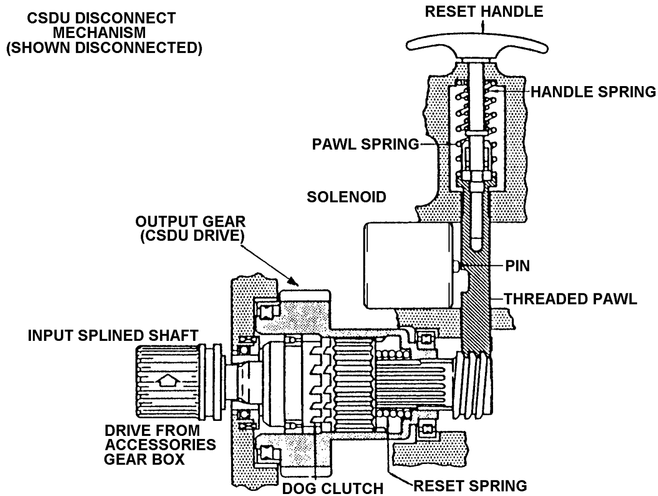
102. When engine speed decreases, the tendency of the generator to underspeed causes the governor to alter the variable unit swash plate angle such that the variable hydraulic unit pumps oil to the fixed hydraulic unit. This causes the fixed unit to rotate in the opposite direction to the variable unit and so the input section of the differential unit rotates in the opposite direction to the carrier shaft. This increases the speed of rotation of the planet gears and output section of the differential unit, increasing generator speed relative to engine speed, to maintain a constant required speed. This is known as the overdrive condition.

103. To provide monitoring of the health of the CSDU the flight deck has provision for indication of oil temperature, oil pressure as well as generator indications. The oil temperature on the **outlet** side of the CSDU or the oil temperature **rise** across the CSDU (sometimes both) are indicated on gauges on the flight deck. Low oil pressure is normally indicated by the illumination of an amber light. Reference to the generator frequency meter can also provide an indirect indication of a faulty CSDU. Some installations provide for indication of the CSDU rpm. The significance of the monitoring instruments is to provide the operator with a means of anticipating or trouble shooting CSDU problems.

Remote CSDU Disconnect

104. Failure of the CSDS to maintain constant generator speed will result in unacceptable fluctuations or variations of the AC supply frequency. Mechanical failure of the CSDU or generator could well cause expensive damage to the engine accessories gearbox and necessitate engine shut down. To prevent the damage that would be caused in either case, a facility is provided to disconnect the drive to the CSDU. Figure 8-13 shows the mechanical disconnect mechanism, which is remotely operated from the flight deck.

FIGURE 8-14
CSDU Mechanical Disconnect



105. In normal operation the threaded pawl is held out of engagement with the threaded CSDU input drive shaft by a solenoid-operated pin. Operation of the ‘disconnect’ switch on the flight deck withdraws the pin and the pawl spring forces the pawl into engagement with the threaded shaft. The rotating drive shaft therefore screws forward until the teeth of the dog clutch un-mesh, disconnecting the CSDU shaft from the accessories gearbox drive shaft, whereupon the CSDU ceases to rotate.

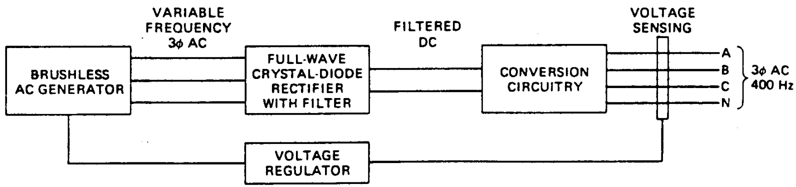
106. Disconnect action can be made provided the engine is running above a specified minimum rpm. The dog clutch can only be reset on the ground, since access to the engine is necessary. By pulling the reset handle the pawl is disengaged from the threaded shaft and a reset spring re-meshes the dog clutch teeth. The solenoid pin will now hold the pawl out of engagement with the CSDU shaft until such time as the remote disconnect switch is again operated.

107. When disconnect selection is made from the flight deck a captioned warning indication typically displays on the electrical control panel and, on EICAS equipped aircraft, a caption appears on the primary display. Also, the associated generator load or power meter will read zero and the generator circuit breaker indicator will display an OPEN indication.

Variable Speed Constant Frequency System (VSCF)

108. Variable speed constant frequency systems, referred to as VSCF systems, are replacing hydromechanical constant-speed drives in modern aircraft design AC systems. Using state-of-the-art electronic components, the VSCF systems improve reliability and allows for more flexibility of generator installation. The system employs a standard brushless AC generator which is driven directly from the engine gearbox and therefore its frequency output will vary with engine speed. The variable three phase output is fed to the full-wave rectifier within the VSCF converter, where it is then changed into DC and filtered. The direct current is fed to the inverter circuitry where it is formed into square wave outputs that are separated and summed to produce three-phase 400Hz AC. Because the VSCF control system can be mounted virtually anywhere on the aircraft and since there is no requirement for a CSD, it allows for a much more compact engine nacelle design. A schematic integrated VSCF system is shown at Figure 8-15. The voltage and frequency output of a VSCF is identical to the output of a CSDU or IDG.

FIGURE 8-15
Variable Speed Constant Frequency Systems



Integrated Drive Generator (IDG)

109. The integrated drive generator is a line replaceable unit (LRU) that combines the brushless generator and its constant speed drive unit in a single casing, which is mounted on the engine accessories gearbox by means of a quick attach/detach (QAD) attachment ring. The integrated drive assembly is completely self-contained, having its own lubrication and cooling system and typically rotates at a constant 12,000 rpm. The high rotary speed enables a reduction in generator size for a given power output. Modern IDG units are typically rated for continuous operation at 90 kVA.

110. The lubricating oil for the IDG is also the cooling medium for the generator. Oil pressure, temperature and (in some cases) quantity is monitored and activates warning annunciators on the flight deck. In some systems low pressure or high temperature will automatically activate the CSDU disconnect mechanism.

AC Power Distribution

111. There are two types of electrical distribution system in use in AC powered aircraft. Twin-engined aircraft use the Split Bus Bar System, in which the two engine-driven generators each supply their own AC bus bar, between which the total electrical load of the aircraft is approximately equally divided. The generators are constant speed, frequency-controlled machines, producing 400 Hz, 115/200 volts single-/three- phase. A third generator drive directly by the APU also produces 115/200 Volt 3 ϕ 400Hz supply.

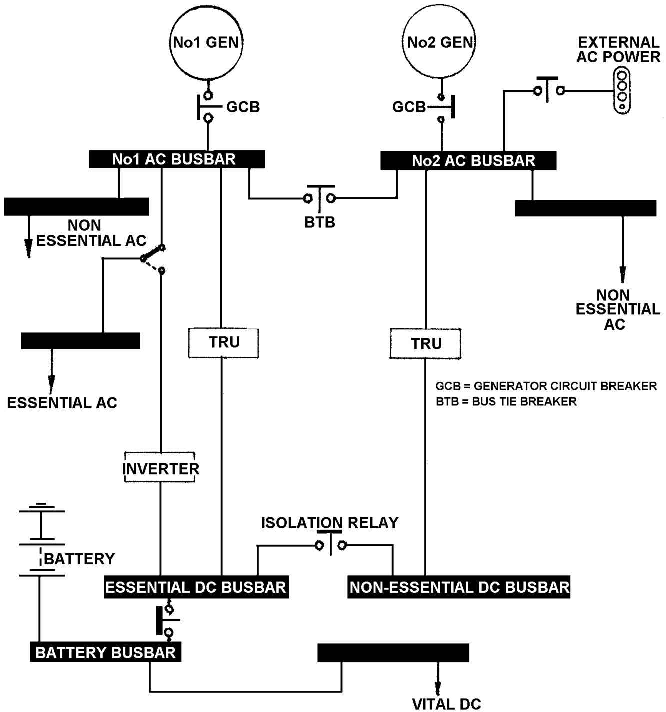
112. Any two generators are capable of supplying all the aircraft's power requirements and any single generator is capable of maintaining vital, essential, and most non-essential services. The generators cannot be cross connected, or paralleled; generator bus bars can be cross connected, or tied, in the event of generator failure.

113. Three- and four-engined AC powered aircraft use an electrical distribution system in which the engine-driven generators are normally operated in parallel. The advantage of such a system of distribution is that power supply is not interrupted, even momentarily, in the event of a generator failure.

Split Bus Bar Distribution

114. Many modern airliners use non-paralleled constant frequency AC as the primary power source, with DC services provided via transformer-rectifier units (TRU's). The generators provide 115v single-phase (200v three-phase), 400 Hz AC to their bus bars, from which non-essential AC consumers are supplied. The essential AC bus can be supplied either from one of the generator bus bars, or from a static inverter supplied with DC from the battery bus bar via the Essential DC bus bar. The generator bus bars are isolated from each other, only being connected by the bus tie breaker (BTB) in the event of failure of one of the generators. There are no load sharing circuits. A typical split bus bar system is shown at Figure 8-15.

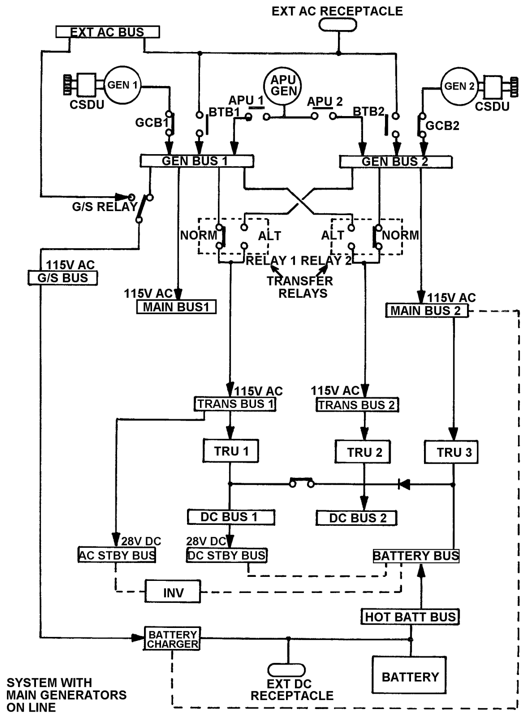
FIGURE 8-16
Split Bus Bar System



115. A practical example of the split bus bar system is that used in the Boeing 737-400 series. This is illustrated at Figure 8-17. It must be appreciated that other aircraft systems will differ considerably in detail, although the general concept remains similar. Please refer to Figure 8-16 when reading the following system description.

FIGURE 8-17

Boeing 737 - 400
Split Bus Bar
System



116. AC and DC power is normally supplied to the system from one of three sources. These are the engine-driven generators, an auxiliary power unit (APU) generator and (on the ground) an external power supply. An interlocking system between breakers ensures correct sequencing (to avoid paralleling) and a source of power selected always takes priority over an existing power source, automatically disconnecting the latter.

Standby power. The standby busses supply the essential AC and DC systems. The AC standby bus is normally supplied from the number one AC transfer bus and the DC standby bus from the number one DC bus. In the event of failure of all AC supplies, both standby busses are supplied from the aircraft battery via the battery bus, and the AC standby bus via a static inverter. A fully charged battery has sufficient capacity to provide power to essential flight instruments, communication and navigation equipment for a minimum of 30 minutes.

117. In flight, automatic switching is provided from normal power sources to alternate power sources when the Standby Power Switch is in the AUTO position. If DC Bus 1 or Transfer Bus 1 loses power, both Standby busses automatically switch to the Battery Bus. Automatic power transfer is an in-flight feature only. To prevent the battery discharging through the Standby busses when the aircraft is on the ground, the air/ground safety sensor inhibits automatic transfer. This can be by-passed by placing the Standby Power Switch in the BATTERY position.

118. Consider a typical sequence of events in normal operation:

External power (ground servicing only). If external power is connected and the ground power control switch (on the flight deck) is selected OFF, BTB 1 and BTB 2 will be open and the ground-servicing relay will be energised to connect the external AC bus to the ground servicing bus. The ground servicing bus will normally supply power to the cabin lighting (for the cleaners), the freight holds (for the loaders), the equipment/undercarriage bay lighting circuits (for the engineers) and the battery charger.

External power (aircraft systems). With the ground power control switch selected ON, both BTBs close to connect 3-phase AC external power to the whole system. DC power is provided by the TRU's and the GS Relay is energised to connect the number one generator bus to the GS bus.

APU generator. On the ground the APU generator can be connected to both generator bus bars, to power the whole system, by selecting the APU relays closed from the flight deck control panel. Under these circumstances, GCBs 1 and 2 will be automatically held open. BTBs 1 and 2 can only be closed once external power to the external AC receptacle has been switched off, otherwise they will also be automatically held open.

Engine driven generators. When the first engine driven generator (say number one) is available, switching its control switch ON trips BTB 1 open and closes GCB 1. The number one generator, via its bus bar, now supplies number one system. The number two system is still supplied by external power or APU generator via BTB 2 and the number two generator bus bar.

119. When the second engine driven generator is available, switching its control switch ON opens BTB 2 and closes GCB 2. Their respective generators now supply the two halves of the system and ground power can be switched OFF and disconnected.

Loss of an engine-driven generator. If either generator fails, its GCB will be tripped open by that generator's under-voltage protection system. A set of auxiliary contacts in the GCB closes to energise the associated transfer relay, causing it to move from NORMAL to ALTERNATE.

120. Imagine that the generator 2 has failed. GCB 2 trips open, disconnecting the generator from its bus bar. The number two transfer relay moves to the ALT position, connecting generator bus number one, via transfer relay two, to transfer bus number two. All services are maintained except the non-essential AC consumers supplied from the number two generator bus via Main Bus 2. If these are required, the APU can be started. When its number two system switch is set to ON, APU 2 will close (connecting the APU generator output to generator bus 2), and the number two transfer relay will return to NORMAL. The APU has now replaced the number two generator.

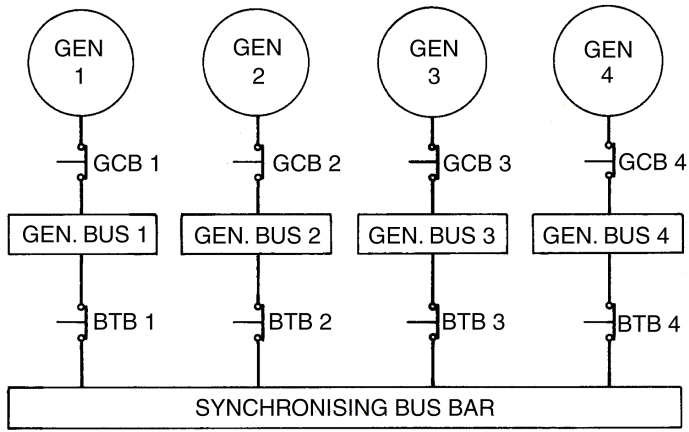
121. During flight the APU can only be connected to one of the generator bus bars, since the APU circuit breakers do not contain auxiliary contacts to activate the transfer relays.

122. In the event of failure of all generators the aircraft battery is the only source of power, supplying the Battery Bus (Essential DC), the Hot Battery Bus (Vital DC), the Switched Hot Battery Bus and the Standby busses.

Parallel AC Electrical Systems

123. In paralleled AC electrical distribution systems each generator is connected through a generator circuit breaker (GCB) to its own AC bus bar, from which a proportion of the electrical services is drawn. Each generator bus bar is connected through a bus tie breaker (BTB) to a single bus bar, known as the tie bus or synchronising bus. The basic concept is illustrated at Figure 8-17.

FIGURE 8-18
Paralleled AC Distribution System



124. It will be seen from Figure 8-18 that, with all generators operating and all GCBs and BTBs closed, the generators are connected in parallel. Provided that all the generators are producing AC at the same frequency and voltage, and that they are in phase with each other, they will equally share the total electrical load. If one generator should fail its GCB will open, disconnecting it from the system, and the remaining generators will share the loads on the four bus bars, as shown at Figure 8-18.

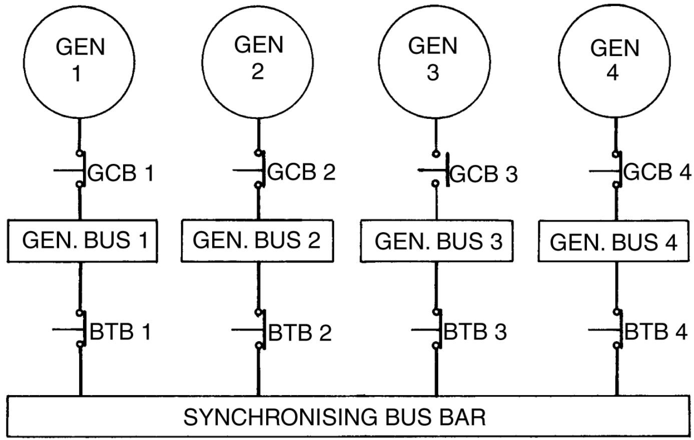
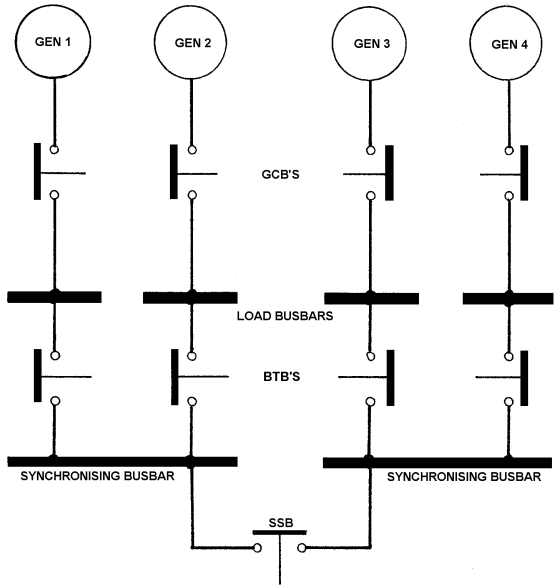


FIGURE 8-19
Paralleled AC Distribution System. No3 Failed

Split Parallel System

125. An alternative form of split bus bar system is one that permits either isolated or paralleled operation of generators. An example of this is shown at Figure 8-19.

FIGURE 8-20
Split Parallel Bus Bar System



126. Each generator is individually connected to its own load bus bar. Pairs of load bus bars may be paralleled by means of a synchronising, or combining, bus bar. On a four-engined aeroplane closing the split system breaker (SSB) will parallel all four generators. A split parallel system offers a number of alternatives; split bus bar operation of the generators, or parallel operation of pairs of generators, or fully-paralleled operation by closing the split system breaker. It is used in several large multi-engine aircraft, of which the Boeing 747 is an example.

127. In 3-phase AC distribution each generator phase is connected to its own single-phase bus bar so there are, in reality, three load bus bars for each generator from which single phase supplies are taken. 3-phase supplies, for larger induction motors for example, are taken by connecting to all three bus bars. For the sake of simplicity only one load, or generator, bus bar is usually shown. Non-essential AC is taken from the load bus bar. Essential AC is taken from a bus bar with an automatic alternate source of power, the synchronising bus bar.

128. With the generators operating, DC supplies are provided by transformer rectifier unit (TRUs) supplied with AC from the appropriate bus bars and supplying DC bus bars, which in turn provide non-essential DC. Essential DC is provided from DC bus bars connected to both TRU and battery supplies. Vital DC is from bus bars connected directly to the aircraft batteries.

Parallel Operation of AC Generators

129. If the voltage output of one AC generator is at its maximum, whilst another (connected in parallel with it) is at its minimum a heavy current will flow between them. The generator giving the higher voltage will be practically short-circuited by the low resistance armature windings of the low output generator. The low output generator will be motored against its drive shaft, which may well shear since it is usually designed as a weak point to prevent damage to the engine accessories gearbox.

130. Similarly, AC generators are synchronous machines. That is to say, their speed of rotation and AC frequency is interdependent. If one of two paralleled generators is at a lower output frequency than the other, it will try to speed up to the synchronous speed of the higher frequency (it can't of course, because it is coupled to the engine gearbox). In other words, the high frequency machine attempts to drive the lower frequency one. Thus, the 'driven' generator is doing nothing and the 'driving' generator takes the entire system load.

131. In order to prevent these imbalances and to ensure that system real and reactive load is shared equally between paralleled generators, it is necessary to ensure that all paralleled AC generators are at the same output frequency and voltage. The first is achieved by generator speed control, through the CSDU, and the second by control of generator field excitation current.

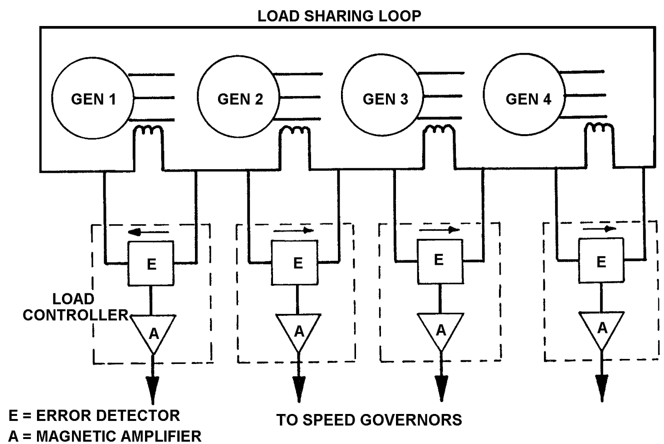
Real Load Sharing

132. For paralleled generators to equally share real load their rotational speeds must be identical. It is generator speed that determines output frequency, and paralleled generators must lock together with respect to frequency.

133. The higher frequency of a higher speeding generator will establish system frequency and this generator will take more than its share of the total system load, tending to motor the remaining paralleled generators.

134. Current transformers are used to sense the real load at phase C of the output from each generator. These transformers are then connected in series to form a load-sharing loop (see Figure 8-21). The output of each transformer is fed to a load controller, comprising an error detector in parallel with the current transformer, and a magnetic amplifier. The function of the error detector is to produce an electrical output signal to the CSDU speed governor in proportion to the magnitude and direction of current flow through the detector. The magnetic amplifier boosts the strength of the signal to improve control accuracy

FIGURE 8-21
Real Load Sharing Group



135. The current in phase C of each generator output induces a proportional voltage in each current transformer. Since the transformers are connected in series a current will flow in the load-sharing loop and this will be the average of the currents produced by each transformer.

136. When all the generator loads are equal, the current from each transformer will be the same and no current will flow through the error detectors.

137. At Figure 8-20, generator 1 is assumed to be running at a higher speed than the remaining generators and is consequently taking an increased load. This will induce an output voltage in its current transformer higher than that in the other three current transformers, producing an unbalanced situation in the load-sharing loop.

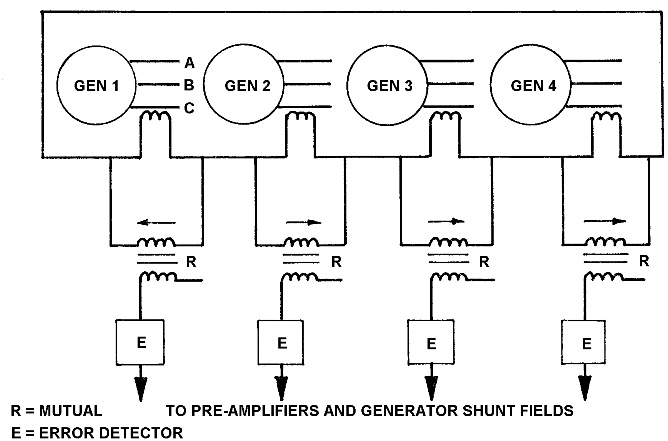
138. The higher voltage in the number 1 current transformer produces current flows in the load-sharing loop and error detectors as shown at Figure 8-20. This causes the number 1 generator error detector to send a decrease speed signal to its CSDU governor, whilst the other three error detectors will send increase speed signals to their respective CSDU governors.

139. The result is that generator 1 will slow down, reducing its output frequency and the other three will speed up, increasing their output frequencies. When each of the four output frequencies match, all four generators will be carrying exactly the same load current (load sharing). The induced voltage, and current flow, will be the same in each current transformer. The current flow in the load-sharing loop will be the same as the current flow in each transformer and so there will be no current flow through the error detectors, which will consequently send a ‘steady-state’ (constant speed) signal to their respective CSDU governors.

Reactive Load Sharing

140. For paralleled generators to equally share reactive load their output voltages must be equal, and these are dependent upon voltage regulators and field excitation current. Suppose the excitation current is higher in one generator than in the remainder, because its voltage regulator is set slightly above mean system value. This will produce a reactive component of current flowing in opposition to the reactive loads of the other generators. Consequently, its load is increased whilst the loads of the other generators are reduced, resulting in unbalanced reactive load sharing

FIGURE 8-22
Reactive Load Sharing Loop



141. Referring now to Figure 8-21, the current transformers which measure the current flowing in phase C of each generator are connected in series to form a second load sharing loop. The output from each transformer is fed to an error detector by way of a transforming device, called a mutual reactor, which produces a phase displacement of approximately 90°.

142. If a reactive load imbalance occurs, the current transformers detect this and voltages proportional to the differential currents are induced in the mutual reactors. These voltages will either lead or lag the respective generator currents by 90°.

143. When the voltage in a particular reactor leads the associated generator current a reactive load exists, indicating that it is taking more than its share of the total reactive load. If the reactor voltage lags generator current this indicates that the generator is taking less than its share of total reactive load.

144. At Figure 8-21, generator 1 is over-excited and taking more than its share of reactive load. The excess voltage induced in its current transformer causes a current flow in the load-sharing loop and through the mutual reactors as indicated by the arrows at Figure 8-21.

145. The result is that number 1 mutual reactor voltage leads current and its associated error detector signals the number 1 generator voltage regulator to reduce excitation current. In the remaining three mutual reactors, the load-sharing loop current flow is in the opposite direction, such that mutual reactor voltage lags current. The associated error detectors consequently signal their respective voltage regulators to increase excitation current.

146. When all generators are carrying the same load, all four current transformer induced voltages will be identical and there will be no current flow in the load-sharing loop and no adjusting signals to the voltage regulators - a steady-state situation.

Auxiliary Generators

147. APU driven generators can rarely be operated in parallel with engine driven generators. This is because their speed (and therefore frequency) is dependent upon the APU gas turbine speed, which is not controlled on the same basis as a CSDU. The Lockheed L-1011 Tristar is a notable exception.

148. Ram air turbine driven generators are fitted in many aircraft, especially those required to meet extra safety standards such as EROPS. A fan, or air turbine, can be extended into the air stream to drive a generator through gearing at about 12,000 rpm. A governor system maintains constant rpm by adjustment of fan blade angle. The generator is a 3-phase AC machine, similar to the main engine-driven generators and its output is regulated to normal system values of voltage and frequency.

Generator Control and Protection

149. As has been demonstrated, AC generation systems are provided with controlling circuits to maintain constant and correct output voltage (115v single phase) and frequency (400 Hz). In the event of a fault developing in the voltage regulators a condition of over- or under-voltage could occur, with potentially damaging results to system components and generators. Similarly, a fault in a CSDU, or its control system, could lead to under- or over-frequency, which may cause overheating due to excess current flow and faulty operation of motors. To guard against the potential hazards of such faults, various protection systems are usually provided with AC distribution systems.

Over-voltage Protection. Over-voltage in any electrical generation system is usually the result of a fault in the field excitation circuit, such as short-circuiting or earthing of the field windings or open-circuiting of the voltage sensing lines. In three phase constant frequency AC systems it is usual for the voltage in all three phases to be detected and fed to a solid-state circuit which will trip the generator circuit breaker (GCB), disconnecting the generator from its bus bar. The over-voltage relay is set to operate at a voltage greater than 130 ± 3 volts.

Under-voltage Protection. Under-voltage occurs whenever a generator is being shut down and could also be due to a field excitation circuit fault such as earthing of the voltage sensing circuit. The protection circuit is similar to that used for over-voltage protection in that an incorrect detected voltage trips a relay which opens the GCB. In both protection devices this also causes an annunciation of the condition on the flight deck.

150. The under-voltage protection circuit operates at a voltage less than 100 ± 3 volts. A time delay of 7 ± 2 seconds is included to ensure that the GCB relay is not tripped due to transient voltages. In addition, it allows the CSDU to slow down to an under-frequency condition during engine shut down, thereby inhibiting tripping of the GCB. Under-voltage protection units operate in conjunction with the reactive load sharing circuits of paralleled generators.

Under- and Over-Frequency Protection is provided by the real load sharing circuit of a paralleled AC generating system. Non-paralleled constant frequency AC systems have sensing, warning and (in some cases) automatic disconnect systems associated with the CSDU's or generators.

Differential Current Protection. This system protects against short-circuit of a feeder line or bus bar and the very high current flow which would result. In AC systems current transformers measures the current flow on the return side of the generator and on the downstream side of the bus bar. A short circuit would cause a significant difference between the two, which causes the differential protection detector to trip a relay, opening the generator circuit breaker.

Monitoring Equipment

Synchronising lights. Before two or more generators are connected to the same synchronising bus bar it is necessary for them to be at the same voltage and frequency and for the AC wave forms to be in phase with each other. In some aircraft a system of flashing lights is provided to indicate the phase comparison between generator outputs, before they are connected to the synchronising bus bar. These synchronising lights are connected into phases A and C of each generator between the generator output and the synchronising bus bar.

151. With reference to Figure 8-19, assume Gen. 1 and 2 are to be paralleled and that Gen. 1 is running and connected to its load bus bar, the synchronising bus bar and Gen. 2 load bus bar via its GCB and the two BTBs. Gen. 2 GCB is open. Gen. 2 is running and both generators have been adjusted as closely as possible to the "master" frequency. If the synchronising light selector switch is moved to the Gen. 2 position, Gen. 2 is connected to the synchronising bus bar via the synchronising lights whilst Gen. 1 is connected to it directly. Hence the lights are sensitive to phase difference between the two generators. The less the phase difference the longer the time interval between flashes of the synchronising lights. The frequency of the 'oncoming' Gen. 2 is now finely adjusted to obtain the greatest time interval between flashes and its GCB is closed while both lights are out, indicating that the two generators are in phase with each other.

AC voltmeters are usually of the moving coil type previously described in DC Components, but the instrument additionally contains a bridge rectifier to convert AC to DC, since it is essentially a DC device. Bridge rectifiers are described in the section dealing with Semiconductor devices.

AC ammeters contain a bridge rectifier, transformer and shunt. A shunt is essentially a low value resistance connected in parallel with the moving coil of the ammeter.

Frequency meters are a standard item of AC instrumentation. They too are moving coil instruments. A potential, dependent upon supply voltage and frequency, is supplied to both fixed and moving coils. Interaction between the magnetic fields produced determines the extent by which the moving coil is deflected against the control springs.

Power meters indicate total power being generated (kVA) and, in some cases, real power (kW) and reactive power (kVAR). Both real and reactive power displays may be combined in a Watt/VAR meter.

152. Functioning of the power meter is similar to that of the frequency meter. A current transformer senses generator phase B load and energises the field coil of the moving coil instrument, producing a field proportional to load. The instrument's reading will therefore be calibrated in kW. When the VAR reading is selected, the moving coil is connected across phases A and C. This causes the current in the moving coil to be 90° out-of-phase with field coil current. If the power factor is 1 there is no interaction between the two magnetic fields produced in the instrument, and the reading is zero (no reactive load). The lower the power factor, the greater the interaction between the two fields and the higher the reactive (kVAR) reading displayed by the instrument.

Failure warning lights are supplied from a DC bus bar to ensure that they will continue to operate when the AC service they are indicating has failed. They usually incorporate a ‘press-to-test’ function, activated by pressing in the bulb holder to complete a circuit through the bulb filament. Indicating lights fall into three categories:

(a) Warning, coloured red and indicating unsafe conditions (generator overheat, engine fire warning, low CSDU oil pressure).

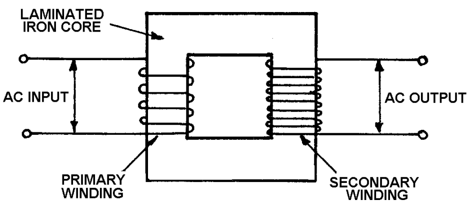
(b) Caution, coloured amber and indicating abnormal conditions (power not available to a bus bar).

(c) Advisory, coloured green or blue and indicating safe/operable (power available). White indicating lights or captions are for information only.

Transformers

153. Transformers are devices for converting AC at one voltage and frequency to AC at another voltage, but the same frequency, by means of electro-magnetic induction. A basic transformer consists of three major components, which are illustrated at Figure 8-23.

FIGURE 8-23
Basic Transformer



154. A laminated soft iron core provides a low reluctance circuit for an alternating magnetic flux field. This field is created by alternating current flow through a primary winding connected to a power source. The alternating magnetic field induces an emf in a secondary winding, connected to the output circuit. Thus, the transformer is a device for the transference of electrical energy by mutual induction.

155. There are two classes of transformer; voltage or power transformers, and current transformers. Various types of power transformers, and current transformers are discussed in greater detail later in this chapter.

Transformer Function

156. The principle of operation of the transformer is as follows. When an AC voltage is applied to the primary winding self-induction establishes a voltage that is almost equal to the supply voltage. The difference between the two voltages produces an excitation current sufficient to set up an alternating flux in the transformer core. This flux cuts across the secondary winding and, by mutual induction induces a voltage in the secondary winding.

157. When a load is connected to the output terminals of the secondary winding the secondary voltage causes current to flow through the load and the secondary winding. This current flow produces a magnetic flux that tends to oppose the primary flux, reducing it. This in turn reduces the self-induced voltage in the primary winding, allowing more primary current to flow and maintaining the core flux at an approximately constant value.

158. Thus, as secondary load current increases, the primary supply current also increases. Eddy currents induced in the iron core lead to heating and power loss, but this is minimised by constructing the core from soft iron laminations separated by a thin insulating coat of varnish.

159. The alternating field in the core of the transformer cuts through both primary and secondary windings. It will be remembered that the magnitude of the emf induced by electro-magnetism depends upon the number of turns of conductor ‘cut’ by the flux. Thus, if the secondary winding has twice as many turns as the primary winding, the voltage at the secondary terminals will be twice that of the voltage applied to the primary winding. The output to input ratio, known as the transformation ratio, is expressed in the equation:

$$\frac{Es}{Ep} = \frac{Ns}{Np}$$

Where:

(a) E_s = secondary voltage

(b) E_p = primary voltage

(c) N_s = number of turns in secondary winding

(d) N_p = number of turns in primary winding

160. When secondary voltage is greater than primary the transformer is said to be a 'step-up' type, the reverse function is known as a 'step-down' type. If the small power loss due to eddy currents, plus other minor losses, is ignored the power in is equal to power out. Thus, if the secondary output is 100v, 50a (5000w) the input will also be 5000w. If this transformer has a 5:1 step down function the primary voltage must be 500v and so the primary current must be 10a.

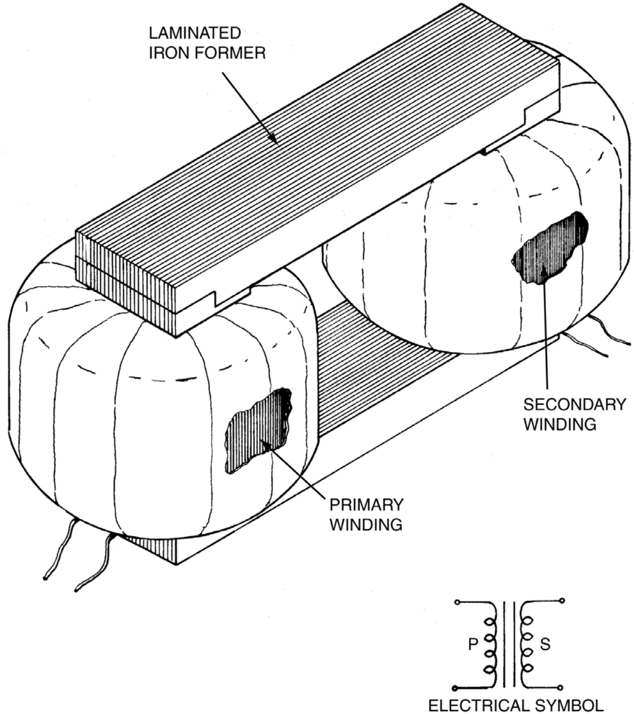
161. As well as step down applications in TRUs (discussed shortly), step up transformers give an invaluable cable weight saving by supplying higher voltage, and therefore lower current, to remote equipment. A local step-down transformer can then supply the equipment at the requisite voltage. Step-down transformers are used to supply fluorescent lights, for example.

162. Transformers are usually rated in volt-amperes or kilovolt amperes. Because the resistance of the primary winding is so low it may be considered a purely inductive unit. It has already been shown that in such a circuit if voltage remains constant and frequency is reduced the current flow will rise. The increased current brings the transformer core nearer to magnetic saturation, which reduces inductance and increases current flow further.

Types of Transformer

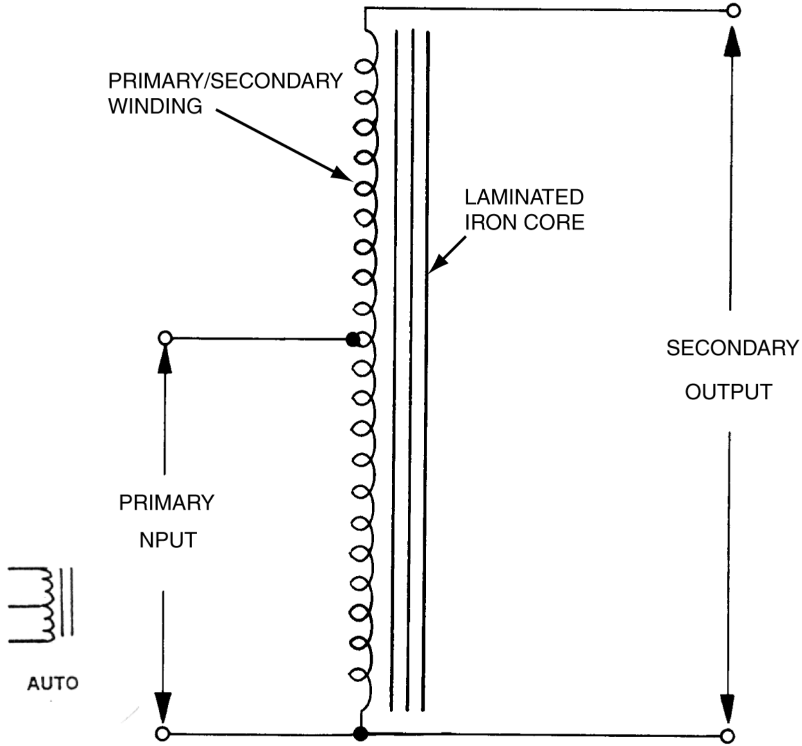
163. A drawing of a standard power transformer is shown at Figure 8-24, together with its electrical symbol. The two parallel lines in the symbol indicate that it is a cored transformer; in some publications the symbol is shown with three parallel lines, indicating the same thing.

FIGURE 8-24
Power Transformer



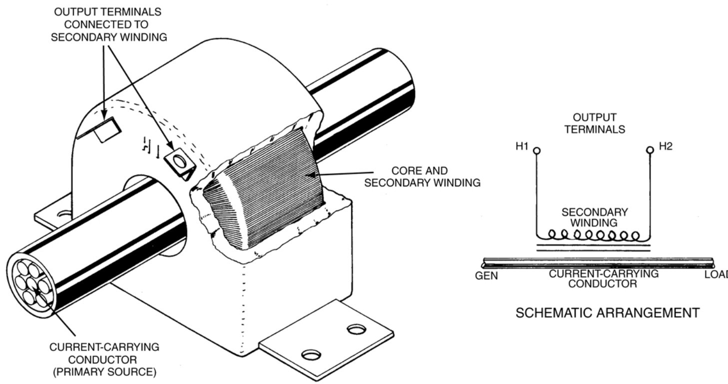
164. Where only a small voltage step, up or down, is required a simpler type of transformer called an auto-transformer can be used. The principle of operation is the same as with a conventional transformer, but now the primary and secondary windings are combined. Auto transformers are consequently smaller and lighter than dual winding transformers, however their use in aircraft systems is usually restricted to lighting supplies, since the lack of primary/secondary isolation makes them suitable for use in low current systems only. The circuit arrangement for an auto-transformer is shown at Figure 8-25, together with its electrical symbol.

FIGURE 8-25
Auto Transformer



165. Current transformers operate on the same basic principle as voltage transformers, but use a circuit's normal insulated conductor as the primary winding, in order to give a secondary output voltage proportional to the current that is flowing in the current-carrying primary conductor. The principle of the current transformer is illustrated at Figure 8-26.

FIGURE 8-26
Current Transformer

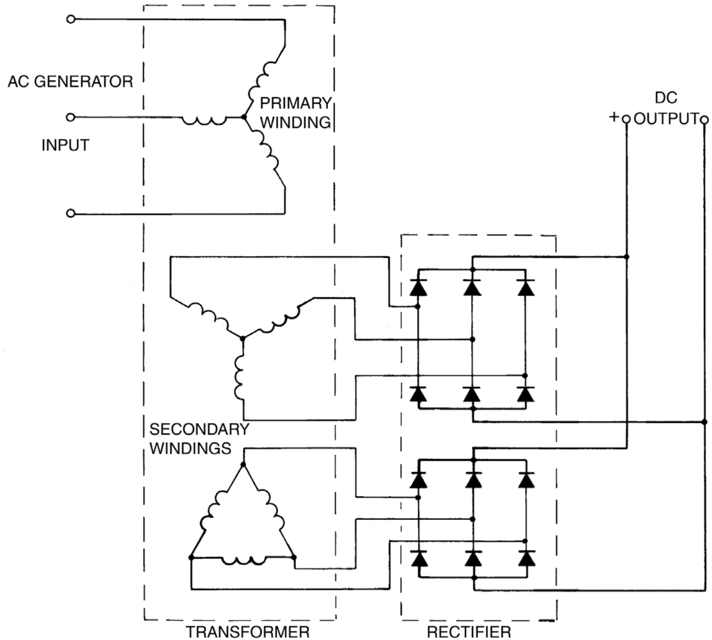


166. Current transformers are used to supply ammeters and power meters, to monitor current flows for fault detection systems and in many AC generator regulation and protection systems.

Transformer Rectifier Units (TRUs)

167. These are used in the conversion of AC to DC. An aircraft's regulated three phase power supply is stepped down by transformers and then rectified in combined TRUs, to supply the 28v DC bus bars for battery charging and other DC loads. A typical three phase transformer consists of a star wound primary winding and secondary windings in star and delta form. The reduced voltage three phase output from the secondary windings is fed to six diode bridge rectifiers, which convert the AC to DC. The circuit diagram for such an arrangement is shown at Figure 8-27.

FIGURE 8-27
Transformer - Rectifier Unit (T.R.U.)



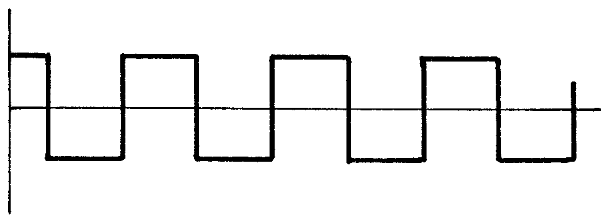
168. A description of bridge rectifiers and diodes is given in the section dealing with semiconductor devices, but for the moment it is sufficient to appreciate that a diode will only permit current flow in the direction indicate by the arrow in the diode symbol.

Inverters

169. Whereas rectifiers convert AC to DC, inverters are devices that convert DC to AC. On older aircraft, employing DC engine-driven generators as the primary electrical power source, constant speed, DC motor driven, AC generators (rotary inverters) were employed to supply the AC power needed by avionic equipment.

170. Later generations of smaller aircraft, whilst still using DC as the principal source of electrical power, employ static (solid state) inverters as an AC power source. 28v DC is supplied to a transistorised circuit that produces 400 Hz in square pulse form, and then reshapes this into the requisite sine wave form. A square wave pulse form is illustrated at Figure 8-28.

FIGURE 8-28
Square Wave Pulse Form



171. Static inverters have the advantage of having no moving parts, and therefore being far more reliable. With static inverters there is very little heat generated and therefore the need for a cooling air supply is removed. Finally, static inverters do not suffer from 'arcing' at low air pressures, as their rotary forebears did, and consequently there is no overriding necessity to locate them within the pressure hull.

172. Most large modern aircraft use AC generators (alternators) as the primary electrical power supply. Emergency AC power supply to essential AC consumers is provided by a battery fed static inverter when the normal 115v AC power sources have failed or are not available.

AC Motors

173. Where there is a requirement for high torque and/or variable speed control, then DC motors are generally superior to AC motors. There are, however, many aircraft applications in which these requirements are either absent or non-rigorous. In these circumstances AC motors prove to be suitable, or even preferable, since the most widely used types of AC motor are brushless and therefore require less maintenance. Also the need for heavy and expensive AC/DC conversion equipment is reduced.

Types of AC Motor

The universal AC/DC commutator motor is essentially the same as a DC motor in construction and is found in many household appliances. It is unsuitable for use with high frequency supplies and is therefore not normally found in aircraft, which usually employ 400Hz AC supplies.

The synchronous motor is so called because it rotates at a speed that is synchronous with the applied AC frequency. In small motors the rotor is a permanent magnet and the stator a soft iron shell with coils wound around it and supplied with AC. The permanent magnet rotor follows the rotating magnetic field set up in the stator by the sequential application of current to the stator coils.

174. In larger synchronous motors the rotor is an electromagnet excited by DC from an external source. There are one, two and three-phase versions of this type of motor. None of them are inherently self-starting although, once running, they maintain constant speed under varying load conditions. Unfortunately, should a synchronous motor be overloaded it will stall. The principal advantages of synchronous motors are their constant speed characteristic and the fact that they tend to correct the power factor because they behave in the same way as a capacitor as far as the AC supply is concerned.

175. Synchronous motors are commonly used as remote rotary speed indicators (tachometers). The aircraft engine drives a small three-phase alternator and its AC output is connected to a synchronous motor inside the rpm indicator. The frequency of the alternator output is directly proportional to engine speed and speed of rotation of the motor is synchronous with its supply frequency. The motor drives the indicating needle of the tachometer through a permanent magnet and spring-loaded drag cup arrangement. A flowmeter is similar in operation, except that a turbine, the speed of which is dependent upon fluid flow, drives the alternator.

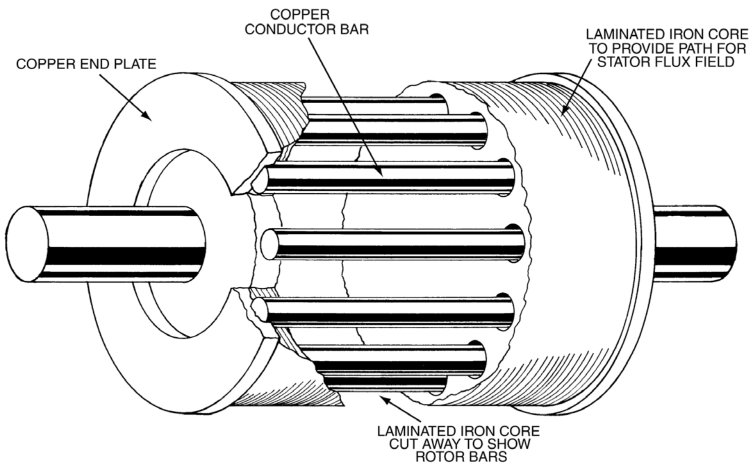
The induction motor is an asynchronous motor since it does not rotate at a speed synchronous with the applied AC frequency. There are one, two and three-phase versions; the two and three-phase motors are inherently self-starting. Once running, they tend to lose speed as load increases and vice versa, but this tendency can be overcome with automatic speed control, if necessary. This type of AC motor is commonly found in aircraft and is normally a brushless motor.

The Induction Motor

176. The principle of operation of an AC induction motor is basically simple. By sequentially energising the windings in the casing (stator) of the motor a rotating magnetic field is produced. This induces a current flow in the conductors of the rotor and the magnetic field resulting from this current flow ‘follows’ the rotating stator field due to magnetic attraction.

177. The rotor may be wound, in other words the induced current is carried in windings of copper wire, or it may be of 'squirrel cage' construction. In the latter type the rotor is drum-shaped and is made up of a ring of closely spaced parallel copper bars arranged axially and connected to flat conducting rings at either end. In either case the conductors are of low resistance, so that the emf induced by the rotating stator field sets up a large current flow and, therefore, a strong magnetic field. The construction of a squirrel cage rotor is illustrated at Figure 8-29.

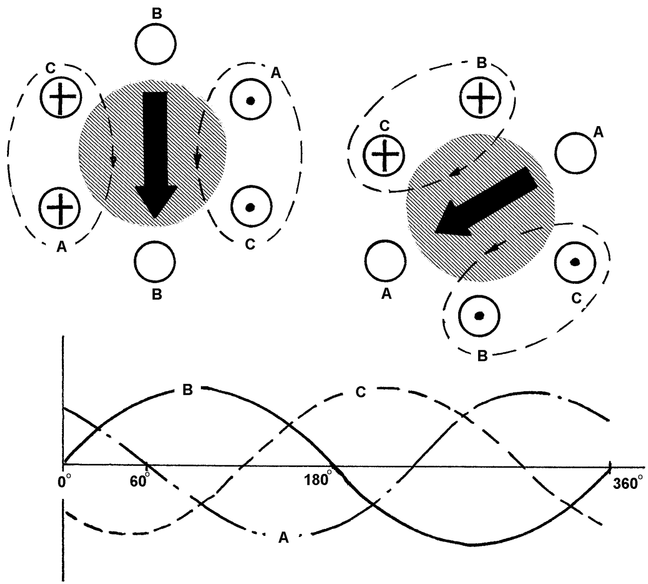
FIGURE 8-29
Squirrel Cage Motor - Construction



178. Squirrel cage motors are very low-torque machines and are typically only used where mechanical loads are light.

179. Figure 8-30 illustrates the principle of operation of a three-phase, squirrel cage induction motor. The upper part of the diagram shows the means by which a rotating magnetic field is set up in the stator of the machine. The lower part shows the sine wave relationship of the three supply phases.

FIGURE 8-30
Squirrel Cage Motor - Operation



180. Consider then the situation at instant 0°. Phases A and C are both carrying current, but of opposite polarity, and phase B is carrying no current. The magnetic fields set up by phases A and C produce a vertical resultant field, as indicated by the heavy arrow. If we now progress to the situation at instant 60°, phases B and C are carrying current, but of opposite polarity, and phase A is carrying no current. The magnetic fields set up by phases B and C produce a resultant field at 60° to the vertical. In other words the resultant field has rotated clockwise 60°. By supplying three-phase AC to the stator windings they will be sequentially energised and the stator field will rotate through 360° at supply frequency.

181. This rotating stator field will cut through the axial conductors in the rotor, inducing an emf and current flow. The magnetic field due to this current flow will be attracted to 'follow' the rotating stator field, producing torque and causing the rotor to rotate. The greater the current flowing in the stator windings the greater the strength of the rotating magnetic field they produce, consequently the greater the induced emf and current flow in the rotor and, therefore, the greater the magnetic attraction, and torque produced.

182. As the speed of rotation of the rotor increases, the interaction between the rotor conductors and the stator field decreases causing a decrease of induced emf and torque, because their relative movement decreases. Eventually a steady-state condition will be reached where the relative rotor/stator field speed is producing just sufficient torque to match the mechanical load on the motor.

183. The rotor never rotates at the same speed as the stator field, since if it did there would be no relative movement and no induced emf. With no induced emf there would be no current flow in the rotor and therefore no rotor electro-magnetic field to maintain the torque necessary for rotation. The difference between the rotor speed and stator field (synchronous) speed is called slip speed, and it increases as the torque load on the motor increases, in order to produce the necessary increases of rotor emf and current flow. Hence the term asynchronous in association with these motors.

184. As the torque load increases, so slip increases and the motor runs more slowly. Eventually, if the torque load becomes excessive, the motor will stop. Technically this is called the pullout point. Pullout torque is at least twice the normal full load torque.

185. Synchronous speed of an induction motor field is determined by the number of poles produced by the stator windings and the frequency of the AC supply. It is defined as:

$$\text{Synchronous Speed (rpm)} = \frac{\text{Frequency (Hz)} \times 60}{\text{No. of pairs of poles}}$$

186. There is another effect that must be considered when a load is applied to an induction motor. This is a lowering of the power factor caused by the inductive reactance of the rotor. When the rotor is turning at almost synchronous speed, that is the speed of the stator field, the frequency of the rotor current and the inductive reactance of the rotor are low.

187. As a load is applied to the rotor, the slip increases and there is a corresponding increase in the frequency of the rotor current. This increases the inductive reactance of the rotor and the power factor of the motor consequently decreases since the power factor is equal to the cosine of the phase angle between voltage and current. Inductive reactance increases this phase angle.

188. Open circuiting of one phase, due to breakage of the relatively fine wiring of the field coils, is a fairly common fault in induction motors. When this happens the motor will continue to run at reduced speed (and will emit a humming noise whilst doing so). The extent of the speed reduction will, theoretically, be to 60% of normal rpm. In practice the motor runs at about half speed unless it is particularly heavily loaded, when it may slow down and stop altogether. In an aircraft the fault may well not be apparent until the next occasion of requiring the motor, when it will not re-start.

189. Short-circuiting between phases, in other words between the stator windings, causes overheating and loss of power and speed, the effect is proportionally greater with increased load.

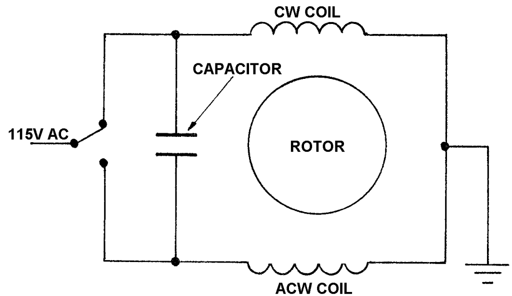
190. Reversing the supply phase sequence will reverse the direction of rotation of the motor. For example, if the phase sequence A, B, C produces clockwise rotation of the stator field, then the sequence B, A, C must produce anti-clockwise rotation of the field and therefore of the rotor. From the foregoing it can be seen that only two of the three phases need be reversed to reverse the direction of rotation and this is easily achieved with a simple switching system.

191. The induction motor is the most suitable AC motor for aircraft use and is commonly found in such applications as fuel pumps, hydraulic pumps, autopilot servos, gyro rotors, gyro torque motors and AC actuators. AC motors are less vulnerable to changes in ambient conditions than DC motors.

192. Single-phase induction motors usually have a single stator winding, split into two parts set at 90° to each other. During start up, both parts of the winding are energised and a rotating magnetic field is set up in the stator. Once sufficient rotor speed has been reached a centrifugally operated switch cuts out one part of the stator windings and the motor continues to run as a single-phase machine, albeit less powerful than an equivalent three-phase motor.

193. Figure 8-30 shows the basic circuit for a reversible single phase AC split-phase motor. When the control switch is in the clockwise position current is supplied direct to the clockwise field coil and by way of a capacitor to the anti-clockwise coil. This has the effect of causing the current in the anti-clockwise coil to lead the current in the clockwise coil, creating a clockwise rotating field. Placing the control switch in the anti-clockwise position reverses the effect. The rotor of such a motor is of squirrel cage construction.

FIGURE 8-31
Reversable Single Phase AC Split Phaser Motor



194. Single-phase induction motors are used for relatively low powered devices such as equipment cooling fans, aircraft rotating navigation lights and retracting landing/taxi light.

195. Some AC systems employ two phase induction motors (asychronous motors) where a servo control of synchronous devices is required. The windings are at 90° to each other, but they are connected to different voltage sources. One source is the main supply which is a constant voltage. The other source serves as a control voltage whose amplitude can be varied, thus allowing the speed of the motor to be varied.

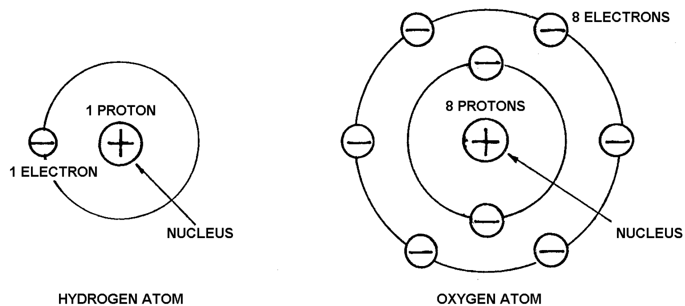
Semiconductors

196. The ability of a substance to conduct an electric current depends upon its atomic structure. Some elements, mostly metals, are good conductors because an electric current will flow easily through them. This is because the atoms of these elements will readily give up, or receive, electrons. Electrons that freely move from one atom to another are known as free electrons.

197. An atom is made up of electrons, protons and neutrons. The nucleus of the atom comprises positively charged protons surrounded by orbiting negatively charged electrons. The total electron charge is equal to, but of opposite polarity to, the total proton charge. Thus, the atom is electrically neutral. Neutrons have the same mass as protons, but have no electrical charge and they play no part in electrical conduction.

198. The electrons orbit the nucleus of the atom in layers, or concentric shells. A simple atom, with only one or two electrons, has a single shell, whilst more complex atoms may have many shells of orbiting electrons. Examples of atomic structure are shown at Figure 8-31.

FIGURE 8-32
Atomic Structure Example



199. The outermost shell of the atom is known as the valence orbit and the electrons in this orbit are known as valence electrons. It is the electrons in the valence orbit which take part in electrical conduction. To be able to do this, these electrons must be free to move within the valence orbit, or shell. The greater the number of electrons in the valence orbit, the less freedom of movement available. Elements comprised of atoms with a high number of electrons in their valence orbits are poor conductors of electricity, or insulators.

200. Elements whose atomic structure has a low number of electrons in the valence orbit offer greater freedom of electron movement and are good electrical conductors.

201. The number of electrons which are actually present in the valence orbit of any element's atoms determines whether that element is a conductor, an insulator (a non-conductor) or a semiconductor.

202. In a conductor, the outer shells (valence orbits) in adjacent atoms are able to overlap each other. When a voltage is applied to the conductor the outer shell is subjected to an electrical force which, if sufficiently strong, will cause some of the electrons to escape and be attracted to the positive pole of the electrical supply (unlike poles attract). However, when an electron escapes from an atom, the atom is no longer electrically neutral, but now has a net positive charge. This attracts any mobile electron in the vicinity.

203. Thus, when current is flowing through a conductor there is a movement of negatively charged electrons in one direction (towards the positive pole of the electrical source) and a movement of positive charges in the opposite direction. These mobile positive charges are known as holes, into which mobile electrons can fall. ‘Conventional’ current flow, from positive to negative, assumes movement of holes in the direction of current flow with movement of electrons in the opposite direction.

204. In an insulator the number of electrons in the outer shell of each atom is too great for the outer shells to overlap, which makes movement of electrons through the material extremely difficult.

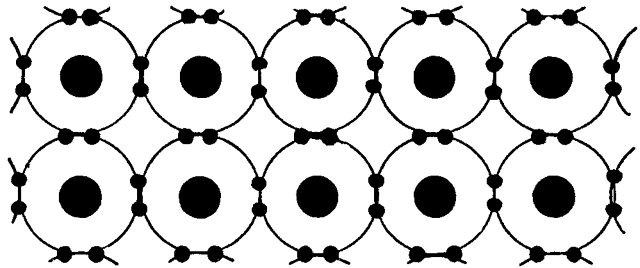
205. Some materials contain a number of electrons in the outer shell of their atomic structure that is midway between that of a conductor and an insulator. In its pure state such a material offers high resistance to current flow because there are sufficient valence electrons (electrons in the outer shell) to prevent 'overlapping'. However, if a precise number of electrons is added, or removed, the outer shells can be persuaded to overlap such that there are a few 'free' electrons, capable of mobility, and the element's resistance can be reduced to a very low value. Such elements are called semiconductors.

Semiconductor Rectifiers

206. A rectifier is a device that offers very low resistance, permitting current flow, in one direction and very high resistance, preventing current flow, in the opposite direction. A rectifier is, in effect, the electrical equivalent of a non-return valve. Solid-state rectifiers employ semiconductors to achieve this effect.

207. Typically, germanium or silicon is used as the semiconductor material. Semiconductor materials in their pure state have 4 electrons in their outer (valence) shell. Germanium atoms have a total of 32 electrons. Silicon atoms have a total of 14 electrons. Both of these elements are highly resistive because the atoms of these materials have a strong valence bond. In other words, the electrons in the outer shell of each atom naturally pair with those of adjacent atoms. Consequently there are no free electrons in these materials to act as current carriers. This is illustrated at Figure 8-32.

FIGURE 8-33
Semiconductor
Valence Bond



208. If a small quantity of another element is added to either germanium or silicon, a process called doping, the resulting impure semiconductor becomes capable of conducting current. Suppose, for example, a small amount of antimony is added to germanium. Antimony atoms have 5 electrons in their outer shell, so the fifth electron cannot pair, or bond, and becomes a free electron. Hence, the germanium becomes conductive.

209. When germanium is doped with antimony it is known as N-type germanium, because it now has some free electrons and electrons are negatively charged. The material is still electrically neutral, since the total number of protons in each atom balances the total number of electrons.

210. If germanium is doped with indium there are vacant spaces, or holes, left in the valence bond between germanium and indium atoms, because indium atoms only have 3 electrons in their outer shell. Electrons that break away from an adjacent valence bond can then fill these holes. This leaves a hole elsewhere, and so the process is repeated. A hole represents a net positive charge, and consequently germanium doped with indium is known as P-type germanium but as with N-type, the material is still electrically neutral.

211. Remember that a P-type material will accept electrons and an N-type material will give up electrons.

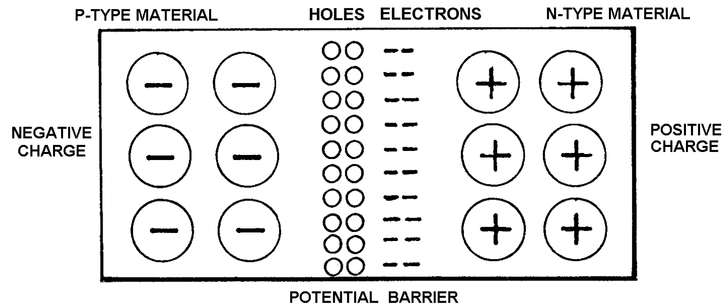
212. Were P-type and N-type semiconductor materials to be joined together the P-type material would act as an anode (a gatherer of electrons) and the N-type material as a cathode (an emitter of electrons). This device would in effect be a diode, which is a device that permits the flow of electrons in one direction only. Electrons will flow from cathode to anode (from the N-type to the P-type material), but not in the opposite direction from anode to cathode, (from the P-type to the N-type material).

213. At the junction of the two materials an interesting phenomenon would occur. The positively charged holes and the negatively charged electrons would be mutually attracted and drift towards the junction, and some of the electrons would migrate across the junction to fill holes.

214. Thus, the P-type material would gain (negative) electrons and become negatively charged and the N-type material would lose electrons and become positively charged. Note, however, that the material would still be electrically neutral overall (that is to say, the total number of electrons still equals the total number of protons).

215. At a point between the P-type and the N-type material the potential difference between the now positively charged N-type and the now negatively charged P-type materials (0.3 volts in germanium) creates a potential barrier which stops the movement of electrons across the junction. Figure 8-33 illustrates the condition that exists when a junction of P-type and N-type germanium is made.

FIGURE 8-34
P type and N type Germanium Junction

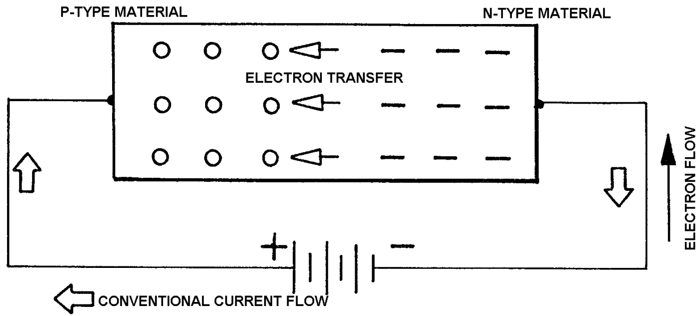


216. The preceding paragraphs describe accurately how a solid state diode works, except that the diode is not in fact two pieces of semiconductor material joined in the middle. It is rather a single piece of germanium (or silicon) which has been doped in such a way that one end is P-type and the other end N-type.

217. If the diode illustrated at Figure 8-33 is connected across the terminals of a battery, it is found that current will flow through the diode in one direction, but not in the other.

218. If the connection is made as at Figure 8-34, with the positive battery terminal connected to the (negatively charged) P side of the diode, and the (positively charged) N side connected to the negative battery terminal, current will flow. (Remember that a conventional current flow from a positive terminal through a circuit to a negative terminal is in fact a net migration of electrons in the opposite direction). This is known as a forward-biased diode. The diode has become a good conductor in the direction shown because its resistance in the forward direction has been reduced to a low value.

FIGURE 8-35
Forward Bias Diode



219. If the connections to the battery terminals are reversed, P side to negative and N side to positive, the negatively charged electrons in the N side are drawn towards the positive connection and the positively charged holes towards the negative connection as illustrated at Figure 8-36. With no movement of electrons across the junction there can be no current flow through the diode. This is known as a reverse-biased diode. The resistance at the junction has been increased sufficiently to prevent current flow through the diode. The symbol representing a diode is illustrated at Figure 8-36.

FIGURE 8-36
Symbol for a Diode

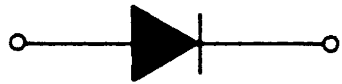
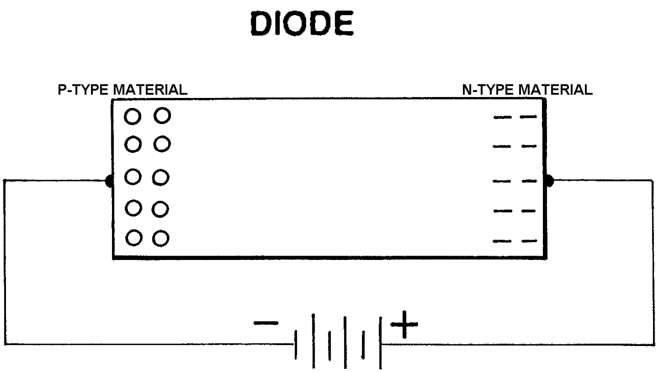


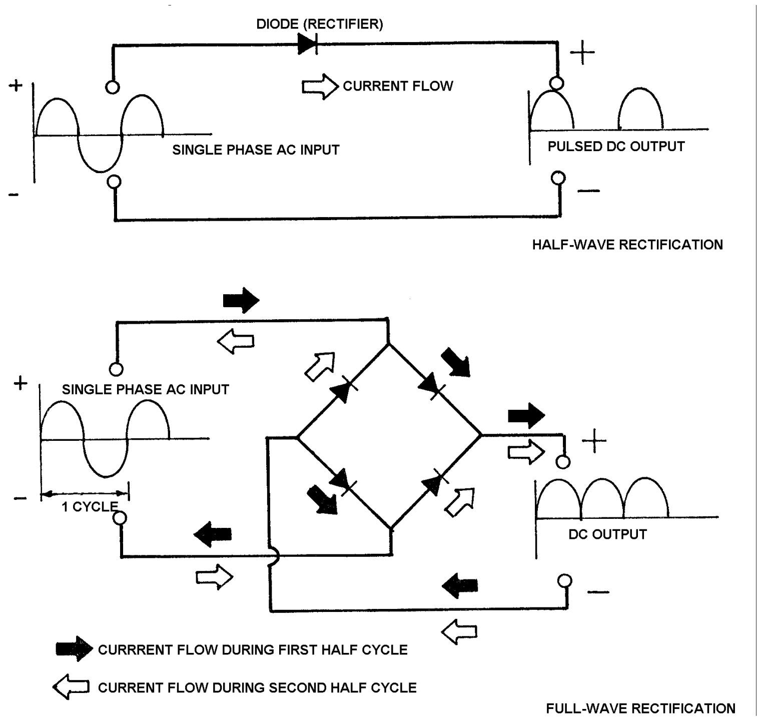
FIGURE 8-37
Reverse Bias Diode



Half-Wave and Full-Wave Rectification

220. A single diode can be used to achieve half-wave rectification of alternating current to direct current. Placed in series with a single phase AC supply, one half cycle of alternating current can pass through the diode to the DC output. When supply current direction of flow is reversed during the second half cycle the diode is reverse-biased and current flow ceases. Consequently, the DC output is in a series of pulses, the number of pulses per second being equal to the AC supply frequency. This is known as half-wave rectification and is illustrated in the upper diagram at Figure 8-37.

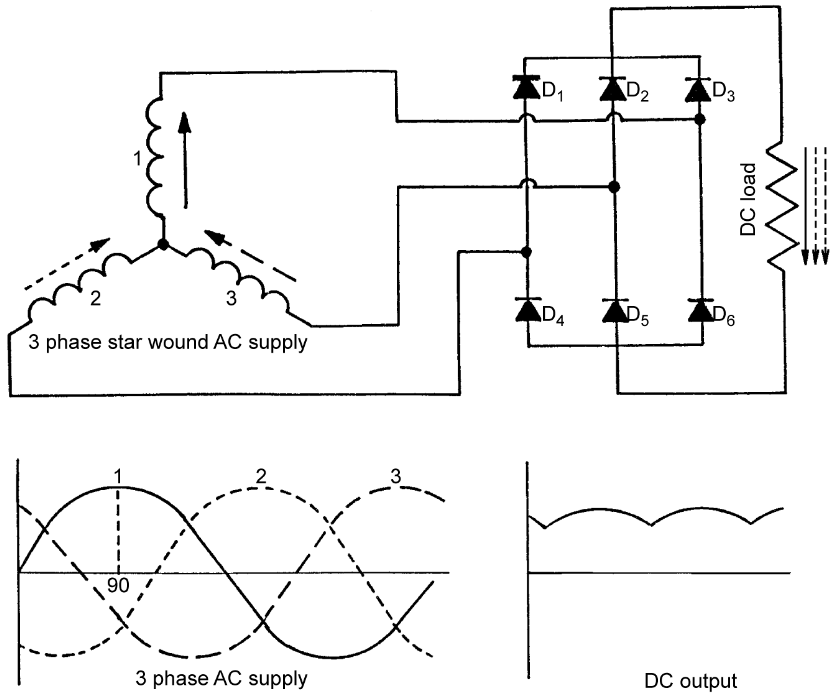
FIGURE 8-38
Half and Full Wave Rectification



221. Half-wave rectification of a three-phase AC supply requires three diodes (one per phase) with their DC outputs connected to one terminal. The opposite polarity DC terminal is connected to the AC neutral return. This arrangement also produces DC pulses, the number of pulses per second being three times the frequency of the AC supply.

222. If four diodes are connected in a bridge circuit, as shown in the lower diagram at Figure 8-37, and supplied with single phase AC, then full-wave rectification of AC to DC is achieved. In aircraft it is usually necessary to obtain DC from a three-phase alternator, in which case the rectifier needs six diodes connected in such a way as to provide one-way paths only.

FIGURE 8-39
3 Phase AC Full Wave Rectification

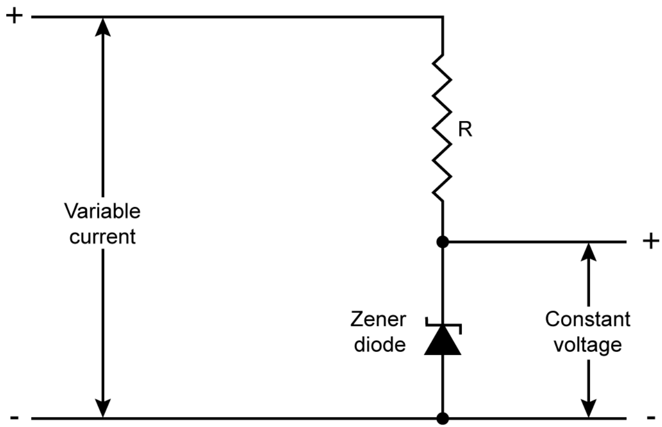


223. A circuit to provide full-wave rectification of three-phase AC to DC is shown at Figure 8-38. Current flow at instant 90° is indicated. At this instantaneous point in the AC cycle, diodes D_3, D_4 and D_5 are forward-biased and diodes D_1, D_2 and D_6 are reverse-biased. Consequently, current flow from all three phases is to the DC load through D_3 and return from the DC load is through D_4 and D_5. At any instant in the three phase supply cycle three diodes will be forward-biased and three reverse-biased to ensure constant direction current flow through the DC load.

224. Some typical diodes and there applications are listed as follows :

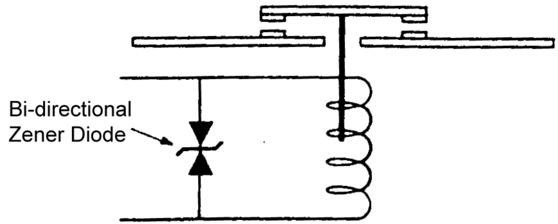
(a) **Zener Diode.** The Zener Diode as illustrated at Figure 8-39 will conduct electricity only under certain voltage conditions. Operation of the zener diode is designed around its breakdown voltage. The breakdown voltage is a set voltage at which the zener diode will conduct, below this set voltage it acts as a normal diode. If the breakdown voltage is reached in the reverse bias mode, it produces what is known as avalanche effect. At that point the diode resistance falls to zero. They are ideal for use in voltage regulation.

FIGURE 8-40
Zener Diode



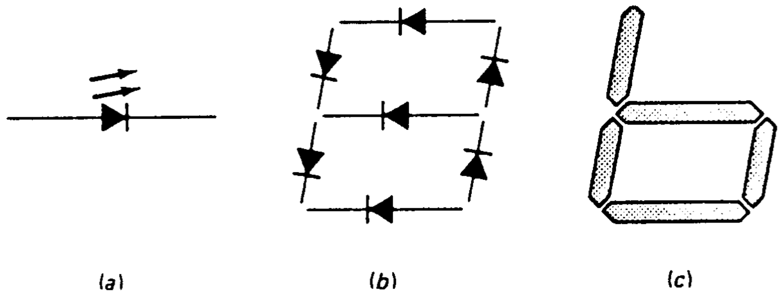
(b) **Bi-directional Zener Diode.** These diodes are often connected in parallel with the magnetic coil of a relay or solenoid. Their inclusion eliminates any transient voltages that could be created during the rise and fall of the relay or solenoids electro-magnetic field. Like the zener diode, the bi-directional zener will conduct current above a certain voltage level. (Figure 8-40).

FIGURE 8-41
Bi-directional
Zener Diode



(c) **Light-emitting Diode (LED).** Used in aircraft systems as indicator lights to display letters and numbers. For an LED to conduct, the applied voltage must be connected in the forward biased condition. Energy given off when the diode is forward biased produces light and heat. A variety of colours are available in LEDs which is determined by the active elements such as phosphorus, gallium and arsenic used in their construction. LEDs require 1.5 to 2.5V and 10 to 20mA to produce the required light.

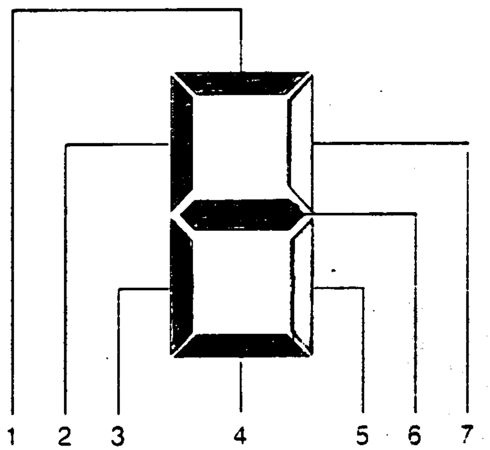
FIGURE 8-42
Light-emitting Diodes



Light-emitting diodes.
(a) A schematic symbol
(b) an LED arrangement to display a digit
(c) an LED display of the digit 6

(d) **Liquid Crystal Display (LCD).** This type of display can provide indication of letter and number patterns or even form a full picture. LCD are commonly grey but some colour systems are available. Liquid crystals are fluid materials that contain molecules arranged in crystal forms. When light is passed through the crystal, it is 'bent'. If a voltage is applied to the liquid crystal the molecules align and the light passes straight through the material. LCDs use this phenomenon to align light waves with polarised filters. The polarised filters will pass or block the light to form the pattern for display. A typical 7-segment liquid crystal display is shown at Figure 8-43. In this example, voltage is applied to the individual segment to form a number 3. The segments that do not have voltage applied are light grey. The light waves pass through these segments and reflect off a mirror mounted behind the rear polariser. The segments that form the number 3 are dark grey, because they are reflecting the light. The liquid crystals of these displays are aligned by an applied voltage.

FIGURE 8-43
Electrodes No 1,2,3,4 and 6 are Energised to Form the Number 3

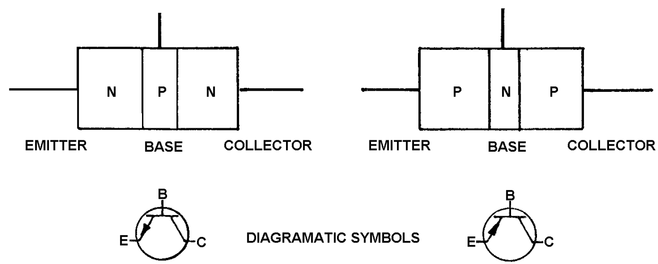


Transistors

225. The primary function of a transistor is to transfer resistance internally to either permit or prevent current flow, depending upon its connection within a circuit, and to increase the output current. In other words, its function is as a switching device or as an amplifier. The name is a combination of transmitter and resistor.

226. A semiconductor transistor is a triode (three electrodes, as opposed to two in a diode) and is typically formed of three sections of a single piece of germanium or silicon, two of which are N-type and one P-type, or vice versa. These are known as junction transistors. They may be either n-p-n or p-n-p type transistors, as illustrated at Figure 8-43.

FIGURE 8-44
Junction Transistors



227. An n-p-n junction transistor comprises two sections of N-type germanium (or silicon) separated by a very thin section of P-type germanium (or silicon). One end of the transistor is called the emitter and the other is called the collector. The thin P-type section is called the base. A p-n-p transistor is basically the same, except that the base is N-type and the emitter and collector P-type material.

228. Functionally n-p-n and p-n-p transistors are the same, except that their connections are of opposite polarity. The arrow in the transistor symbol indicates the type of transistor (pointing out for n-p-n and in for p-n-p) and defines the direction of current flow. Conventional current flow is, of course, opposite to electron flow.

229. In the transistors described there are two junctions between the N-type and P-type material. To permit the transistor to conduct both junctions must be correctly biased. The base-emitter junction must be forward-biased and the base-collector junction must be reverse-biased.

230. In an n-p-n transistor this condition is achieved when the emitter is negative with respect to the base and the collector is positive with respect to the base. A p-n-p transistor will be biased in conducting mode (both junctions forward-biased) when the emitter is positive with respect to the base and the collector is negative with respect to the base.

231. Transistors have replaced valves, or electron tubes, in electronic circuits. Their advantage over valves is that they are considerably smaller and they generate little heat and consequently require much less power than a valve performing an equivalent function. They are cheaper to produce than valves and are generally more reliable. In common with other semiconductor devices, transistors are easily damaged by heat and it is often essential to provide heat sinks and adequate air circulation, or even air conditioning, for transistorised circuits.

Computer, Binary and Logics

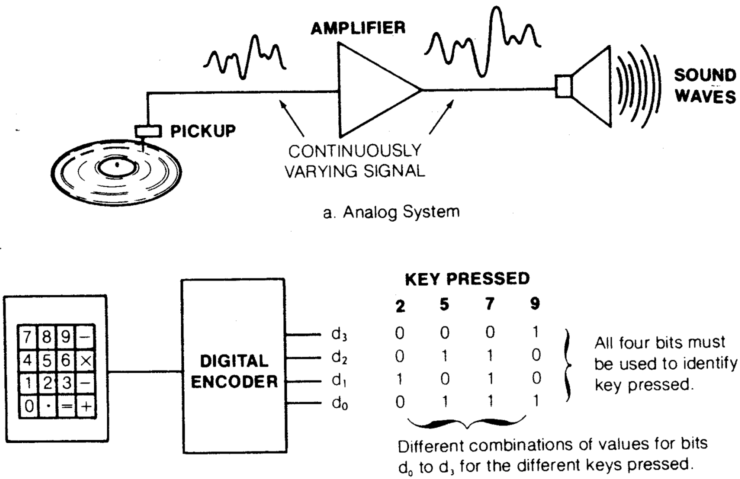
Computer, Binary and Logics

Computer Basics

1. On modern large aircraft computers operate practically every system in whole or in part. In some, such as the later series of Airbus aircraft, a fly-by-wire control system is used in which the pilot's commands are transmitted to the control surfaces only through computer-controlled electrical systems.

2. Computers can be analogue systems or digital systems. When a computer is an analogue system, the quantities to be computed, transmitted or controlled are represented with physical means that vary in a smooth continuous fashion. The representation is not broken into discrete parts or separated into set levels and the signal is usually carried on one or two wires. By contrast digital systems represent a system quantity by breaking it into discrete parts and usually several wires bundled in a group are needed to carry the signals (digital bus).

FIGURE 9-1
Analogue and Digital Systems



a. Analog System

b. Digital System

3. A system that plays records or cassette tapes is an example of an analogue system. Continuously varying signals of the audio tones from the record or tape pass from the pick-up to the amplifier and to the loudspeaker. Sound impulses on the record or tape are converted to continuous electrical signals that are amplified and reconverted to sound by the loudspeaker. The electrical signals have provided an analogy or 'analogue' of the sound signals. The circuits used to handle analogue information electronically are called linear circuits.

Digital Computer Components

4. The basic building blocks of a computer are shown in Figure 9-1. The computer is designed to perform the same sequence of steps to execute a command as the human thought process. These steps are as follows:

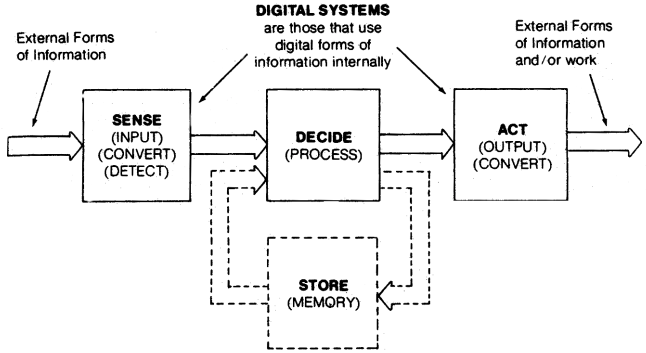
(a) **Sense Function.** The 'sense function' senses information, converts it to a form that can be read by the 'decide function'. This input information can be data about surrounding conditions such as temperature, pressure, light or it can be commands that set the machine in a given mode of operation and tell it where to start. By having sense elements, the system can receive information from humans via a key-board. The functional units that perform the 'sense function' are simply called inputs or input interface.

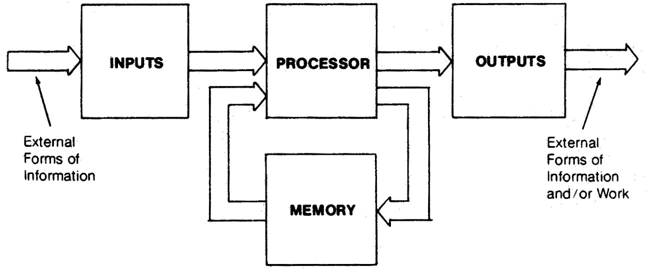
(b) **Decide Function.** The 'decide function' is like the reasoning function in the human brain. All the computations, logical operations and decisions are made here. These decisions take into account the inputs (commands and information about the surroundings) and information in the memory. In the computer, the 'decide function' is provided by the processor. It performs the basic arithmetic and logical decisions required by the computer. It also controls the operation of the computer by turning the other functional units on and off at the appropriate times.

(c) **Store Function.** The 'store function' is the computers memory. It must remember what it is asked to do, information on how it performs the function or task and the results of what it has done. It must also remember a number of rules used in making decisions, performing arithmetic and controlling the system.

(d) **Act Function.** Once a decision has been made by the processor, the 'act function' carries out the command. The 'act function' may be termed the computer output. The output interface may convert the information to a format to be displayed on a visual display unit or provide commands to operate a system or motor for example.

FIGURE 9-2
Digital Computer Components

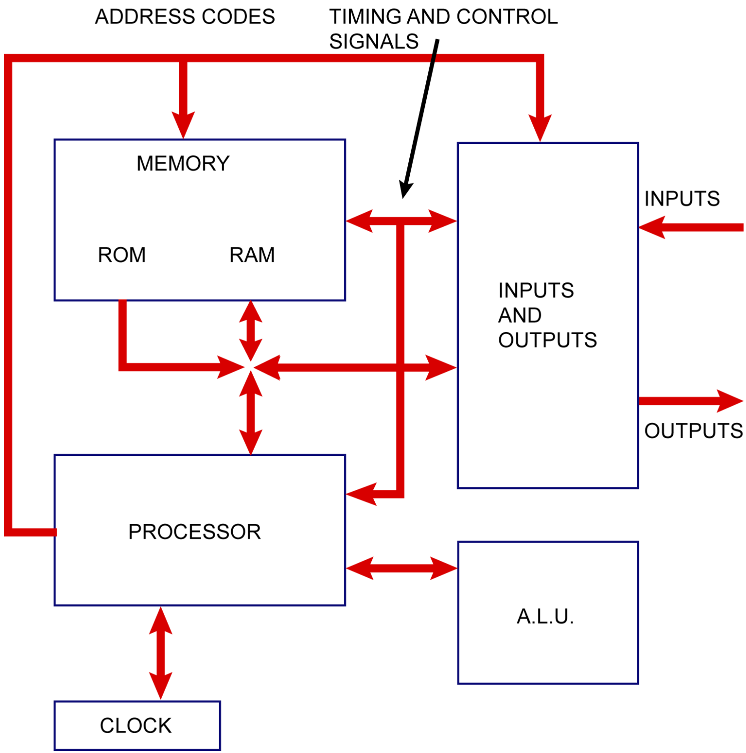




The Central Processing Unit (CPU)

5. The central processing unit is the brain of the computer. It has access to the arithmetic logic unit (ALU) to carry out calculations, the memory block for read only memory (ROM) and random access memory (RAM). The read only memory contains the control instructions for computer operation and also data for reference purposes that cannot be changed. The random access memory is the memory used by the computer to store and retrieve data as required. The clock provides the timing signal for coding the digital signals.

FIGURE 9-3
Central Processing Unit



6. When discussing computers, the terms hardware and software are bound to be encountered. Hardware is the assembly of printed circuit boards, integrated circuit chips, transistors, resistors, capacitors and all the wiring, connections and switches that make up the computer. Software is the term applied to the step-by-step programme written to control the system. The name software came about because most software is written or printed material.

7. Digital signals are termed bits, '1s' and '0s'. The arrangement of bits form codes to identify each number. A group of 16 bits is called a word, 8 bits is called a byte and 4 bits a nibble. Memory capacity in computers is expressed in bytes.

Analogue Signals and Digital Computers

8. Where analogue sensors supply input information to a digital computer the information is converted to digital form by an analogue to digital (A/D) converter. A digital output signal to an analogue system is converted to analogue form by a digital to analogue (D/A) converter.

Binary Numbers

9. The language of the computer is the binary numerical system, which is made up of only two basic numbers, 1 and 0. Each decimal number is represented by a unique combination of 1s and 0s, which can be made to mean a variety of things to the computer. For example, in a particular system 1 may represent ON and 0 may represent OFF; in another, 1 may represent PRESSURE and 0 NO PRESSURE. In what is called a positive logic circuit 1 represents positive voltage and 0 represents zero or negative voltage; in a negative logic circuit the reverse is true.

10. Since the language of the computer is the binary number system and the numerical language of humans is the decimal number system, there must be a consistent conversion between the two. As with the decimal system, the binary system increases value by adding columns to the left of the first digit. In the decimal system there are 10 basic numerals from 0 to 9, so any value above 9 is shown by adding a column, or columns, to the left of the first digit, beginning with 10.

11. In the binary system there are only two basic numerals, so the number of columns required, representing any given decimal number, is greater than in the decimal system. The first or right hand digit of a binary number is represented by 2^0, which equals 1. The digit to the left of this equals 2^1, or 2. The third digit to the left equals 2^2, or 4 and so on.

12. To find the decimal value of a binary number it is a case of adding the individual values of the binary 1s, remembering that binary 0s have a value of zero. Thus the binary number 1111 is $2^3+2^2+2^1+2^0 = 8+4+2+1 = 15$. The binary number 1001 is $2^3+0+0+2^0 = 8+0+0+1 = 9$.

13. A decimal number can be converted to its binary equivalent by repeatedly dividing by two and recording the remainder sequentially from right to left. For example, the binary equivalent of 25 is found as follows:

$$\frac{25}{2} = 12(\text{record 1 remaining})\frac{12}{2} = 6(0\text{ remaining})\frac{6}{2} = 3(0\text{ remaining})$$

$$\frac{3}{2} = 1(1\text{ remaining})\frac{1}{2} = 0(1\text{ remaining})\text{ Result :- }11001$$

Hence, the binary equivalent of 25 is 11001. (Proof: $2^4+2^3+0+0+2^0 = 16+8+0+0+1 = 25$).

14. The digital 1s and 0s that make up the binary numbers are the basis for the whole of the computer code, or language, and may be made to represent letters of the alphabet, words or components of a system. A number may represent the temperature in an air conditioning system and another number may represent the location at which the temperature is measured. For example, 0101 (5) and 0111 (7) could be the digital code indicating a temperature of 5°C at location number 7, which might be the mixing valve outlet.

15. One binary digit (0 or 1) is known as a bit, 1 being a high logic level and 0 a low logic level. A group of bits, such as a five digit binary number for example, form what is known as a byte. A group of bits forming standard computer information data is known as a word. In many computer systems words are of standard length, typically 16 or 32 bits.

Binary Arithmetic

16. As with any numerical system it must be possible to add, subtract, multiply and divide in order for the computer to perform calculations using the input data. With binary addition, subtraction, multiplication and division there are four basic rules in each case.

Rule No.	Addition	Subtraction	Multiplication & Division
1	$0 + 0 = 0$	$0 - 0 = 0$	$0 \times 0 = 0$
2	$0 + 1 = 1$	$1 - 0 = 1$	$0 \times 1 = 0$
3	$1 + 0 = 1$	$1 - 1 = 0$	$1 \times 0 = 0$
4	$1 + 1 = 10$	$10 - 1 = 1$	$1 \times 1 = 1$

EXAMPLE 9-1

EXAMPLE

Addition

0010	0101
0011	0101
0101	1010

Subtraction

1000	1111
0010	0101
0110	1010

Multiplication

0010
0011
0110

Try converting the above binary numbers to decimal, and check whether the results are correct in each case.

Computer Binary Code Systems

17. There are a number of code systems used to convert bits into letters of the alphabet, decimal numbers, symbols and so on. The principal systems in use are the binary-coded decimal (BCD) system, octal notation and the hexadecimal number system.

18. In the BCD system, four bits represent each digit of the decimal system from 0 to 9. Each byte (eight bits) therefore has four unused digit combinations, which can be used for alphabetic symbols. The BCD system is used mainly for data interchange between computers and their peripherals.

19. The octal notation system is primarily used in computer programming, which tends to involve manipulation of large quantities of binary numbers. It represents decimal numbers to the base eight (octal numbers) in a binary language comprised of three-bit groups.

20. The hexadecimal number system uses base sixteen and its primary function is the representation of very large numbers in computerised memory systems. The hexadecimal numbers are 0 to 9 and the letters A to F, representing the decimal numbers 0 to 15. A discrete 4-bit binary number represents each hexadecimal number.

Logic Circuits and Symbols

21. Logic gates represent a fundamental function performed by computerised circuits. Each gate may have several inputs and will have only one output, which will depend upon the state of the inputs. The logic gate is a basic function performed by a computer and forms part of an integrated circuit. There are six commonly used logic gates, known as the AND, the OR and the INVERTER (NOT) - the basic gates - and NOR, NAND and EXCLUSIVE OR.

22. For each logic gate there is a truth table that shows in binary form the relationship between the gate inputs and output. Each gate is represented by a symbol of a discrete shape, designed so that it points in the direction of logic 'flow'. Conventionally the inputs are shown on the left side of the gate and the output on the right. Inputs and outputs are shown as 1 or 0, where 1 represents ON, or positive voltage and 0 represents OFF, or negative voltage.

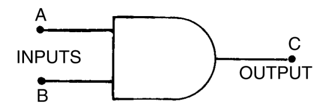
The AND Gate

23. The AND gate symbolises a situation where both inputs to the gate must be ON to produce an output. Hence, both inputs (A and B) must be 1 for an output (1) at C. If either A or B is 0 the output at C will be 0. Because the AND gate will only produce an output when both inputs are in the logic 1 state, it is sometimes referred to as the 'all or nothing' gate.

24. Figure 9-4 shows the AND gate logic symbol (a), its truth table (b), the series switching circuit it represents (c) and a solid-state AND circuit (d). Figure (c) represents a simple AND circuit where closure of both switches A and B will provide a cureent flow in Relay R_1. Figure (d) represents an integrated circuit (IC) where the requirement is to produce a positive (I) or negative (O) output at Point C. If both diodes are reverse biased with a positive (I) input then point C will be positive (I) (i.e. no volts drop across R_1). If either diode is forward biased then Point C will be negative (volts drop across R_1). if both diodes are foward biased with no input (O) then point C will be negative (i.e. volts drop across R_1).

FIGURE 9-4
The AND Gate

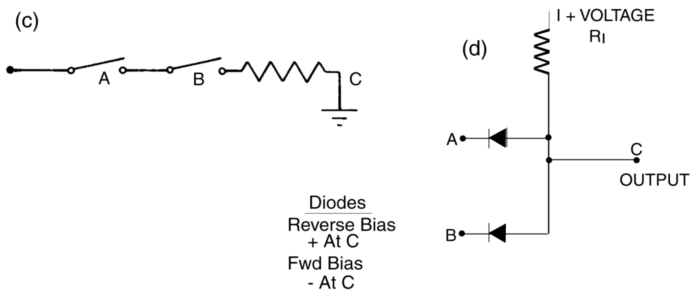
(a)



(b)

INPUTS		OUTPUT
A	B	C
0	0	0
0	1	0
1	0	0
1	1	1

(c) (d)



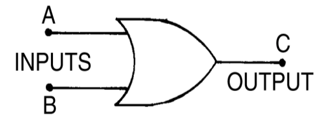
The OR Gate

25. The OR gate represents a situation where either input (A or B) being in logic state 1 (ON) will produce an output, that is to say a logic 1 output. Only if both inputs are in logic state 0 will the output be 0. Because the OR gate will produce an output if either, or both, inputs are in logic state 1 it is sometimes referred to as an 'any or all' gate.

26. Figure 9-5 shows the OR gate logic symbol (a), its truth table (b), the parallel switching circuit it represents (c) and a solid-state OR circuit (d). Figure (c) represents a simple OR circuit where any input of (1) will produce an ON (1) output. Figure (d) represents an integrated circuit (IC). In this circuit, if a positive voltage (1) is applied to either input the dioed A or B as appropriate will be forward biased and current will flow through R_1. Point C will be positive as a result of the volts drop across R_1. If there is no current flow across R_1 then C will be negative. There is no current flow across R_1 when both A and B are negative. Should either A or B on A or B inputs become positive the output is positive.

FIGURE 9-5
The OR Gate

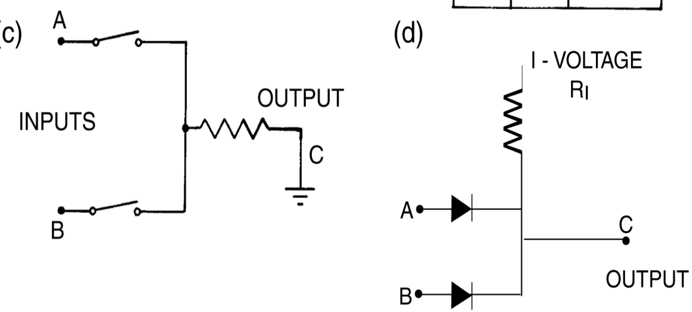
(a)



(b)

INPUTS		OUTPUT
A	B	C
0	0	0
0	1	1
1	0	1
1	1	1

(c) (d)

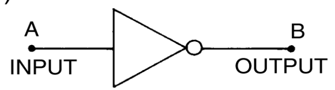


The NOT (Inverter) Gate

27. The third of the three basic logic gates, the NOT, or INVERTER, gate has only a single input and output and is used to invert a function. If the input is 1, the output will be 0, and vice versa. The logic symbol, truth table, a simple electrical circuit, and a solid state circuit that could be represented by a NOT gate are shown at Figure 9-6.

FIGURE 9-6
The NOT Gate

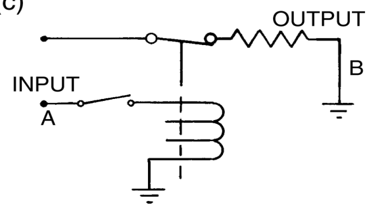
(a)



(b)

INPUT	OUTPUT
A	B
0	1
1	0

(c)

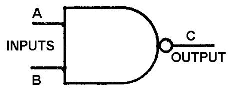


28. The NOT gate is generally used in conjunction with other gates, rather than on its own, to alter their basic functions.

The NAND Gate

29. The NAND gate is an AND gate with an inverted output. The output of this gate will be 1 if any input is 0. In other words it is an AND gate with a NOT gate added on the output side (the term NAND is the diminutive of NOT and AND). Thus, the normal outputs of the AND gate will be inverted. The NAND gate symbol, and the derivation of its truth table, is shown at Figure 9-7.

FIGURE 9-7
The NAND Gate



AND FUNCTION

INPUTS		OUTPUT
A	B	C
0	0	0
0	1	0
1	0	0
1	1	1

→

NOT FUNCTION

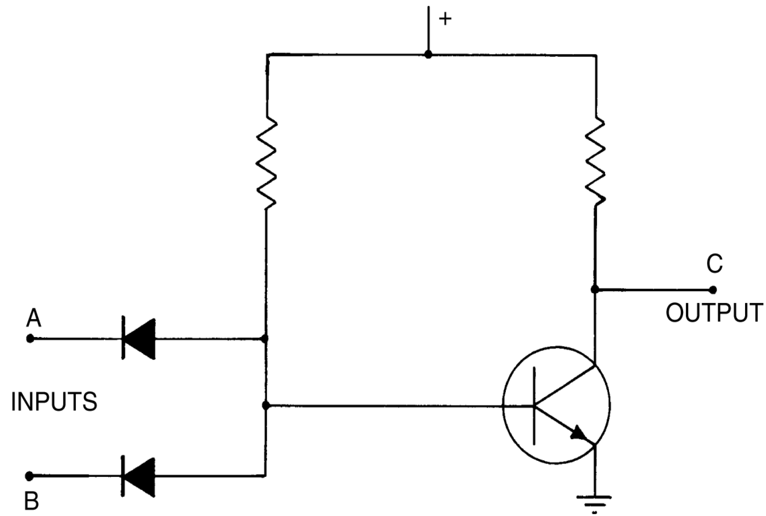
IN	OUT
0	1
0	1
0	1
1	0

NAND TRUTH TABLE

INPUTS		OUTPUT
A	B	C
0	0	1
0	1	1
1	0	1
1	1	0

30. The symbol is abbreviated, rather than show an AND gate with a NOT gate added, the NOT (INVERTER) function is symbolised by a small circle on the output side. The NAND truth table shows the result of the combined gates. A simple circuit represented by a NAND gate is shown at Figure 9-8.

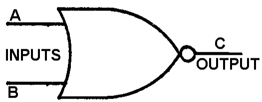
FIGURE 9-8
Simple NAND Gate



The NOR Gate

31. As with the NAND gate, a NOR gate is an OR with a NOT added (that is to say a NOT OR gate). It is an OR gate with an inverted output. The symbology and truth table derivation for the NOR gate are shown at Figure 9-9.

FIGURE 9-9
The NOR Gate



OR FUNCTION

INPUTS		OUTPUT
A	B	C
0	0	0
0	1	1
1	0	1
1	1	1

→

NOT FUNCTION

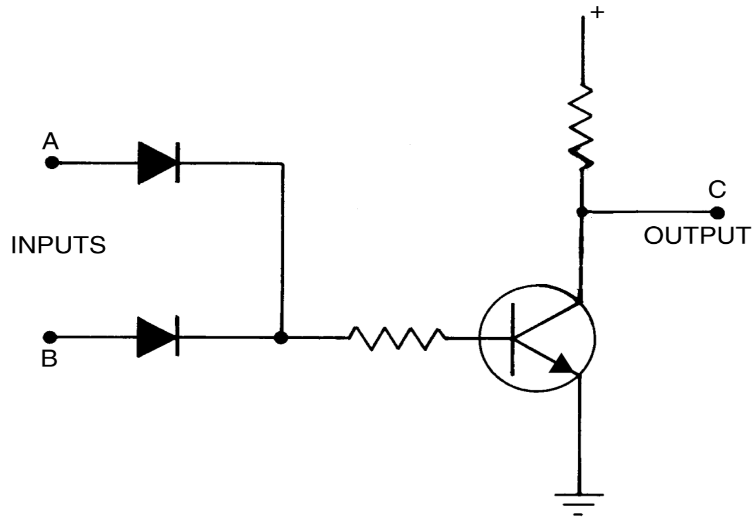
IN	OUT
0	1
1	0
1	0
1	0

NOR TRUTH TABLE

INPUTS		OUTPUT
A	B	C
0	0	1
0	1	0
1	1	0
1	0	0

32. A simple circuit represented by a NOR gate is shown at Figure 9-10.

FIGURE 9-10
Simple NOR Gate



Inhibited or Negated Gates

33. The basic logic gates can be used in a wide range of combinations to symbolise sequential functions. Having seen how a NOT gate can be added to the output of an AND or OR gate to invert the output, it follows that the functions of these two gates could be changed further by adding a NOT gate to one or other of the inputs.

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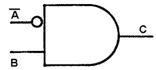
34. Figure 9-11 shows the two basic gates modified in this manner, the NOT gate position being shown by a small circle in each case. For each combination the truth table has been derived. It is current practice to indicate an inverted input or output by a 'state symbol', in the form of a horizontal bar above the appropriate letter, in the truth table and on the logic gate, as shown.

FIGURE 9-11

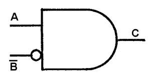


NOT FUNCTION	
IN	OUT
A	
0	1
0	1
1	0
1	0

→

AND FUNCTION		
INPUTS		OUTPUT
A	B	C
1	0	0
1	1	1
0	0	0
0	1	0

TRUTH TABLE		
INPUTS		OUTPUT
$\overline{A}$	B	C
0	0	0
0	1	1
1	0	0
1	1	0

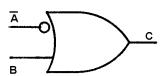


IN	OUT
B	
0	1
1	0
0	1
1	0

→

INPUTS		OUTPUT
A	B	C
0	1	0
0	0	0
1	1	1
1	0	0

INPUTS		OUTPUT
A	$\overline{B}$	C
0	0	0
0	1	0
1	0	1
1	1	0

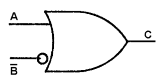


IN	OUT
A	
0	1
0	1
1	0
1	0

→

OR FUNCTION		
INPUTS		OUTPUT
A	B	C
1	0	1
1	1	1
0	0	0
0	1	1

INPUTS		OUTPUT
$\overline{A}$	B	C
0	0	1
0	1	1
1	0	0
1	1	1



IN	OUT
B	
0	1
1	0
0	1
1	0

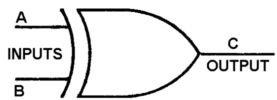
INPUTS		OUTPUT
A	B	C
0	1	1
0	0	0
1	1	1
1	0	1

INPUTS		OUTPUT
A	$\overline{B}$	C
0	0	1
0	1	0
1	0	1
1	1	1

The Exclusive OR Gate

35. The exclusive OR gate symbolises a situation in which similar inputs produce a 0 output and dissimilar inputs produce a 1 output. The symbol and truth table for an exclusive OR gate are shown at Figure 9-12

FIGURE 9-12
The Exclusive OR Gate



INPUTS		OUTPUT
A	B	C
0	0	0
0	1	1
1	0	1
1	1	0

Positive and Negative Logic

36. As previously stated, logic circuit signals are at two levels, referred to as being binary 1 and binary 0. In simplistic terms these may be regarded as ON or OFF, but in reality they represent voltage levels.

37. Positive Logic circuits are those in which binary 1 represents a higher voltage level and binary 0 a lower one. The actual voltage values may be both positive, both negative or one positive and the other negative. Negative Logic circuits, in which binary 1 always represents the lower voltage level, are rarely used and then only when system design parameters demand it.

Basic Logic Circuits

38. Every computer and its peripherals uses a variety of combinations of the basic logic gates described in the preceding text. The five most common logic circuits are Adders, Subtracters, Clocks, Latches and Flip-Flops.

39. Adders and Subtracters perform the functions their names suggest. An Adder circuit adds binary digits and a Subtracter circuit subtracts them. A Clock circuit produces a constant alternating sequence of 1s and 0s to create a square waveform, useful as a time reference. Latches and Flip-Flop circuits are used to perform basic memory functions. In both cases the output signal is retained after the inputs have been removed, enabling the computer to 'remember' input data.

40. The flip-flop circuit remembers or stores the digital signal at the input when a clock pulse arrives. Each flip-flop can store one bit of information, 0 or 1. A clock pulse causes the output of a flip-flop to 'flip' to a 1 or 'flop' to a 0, depending on which bit is being presented at the input.

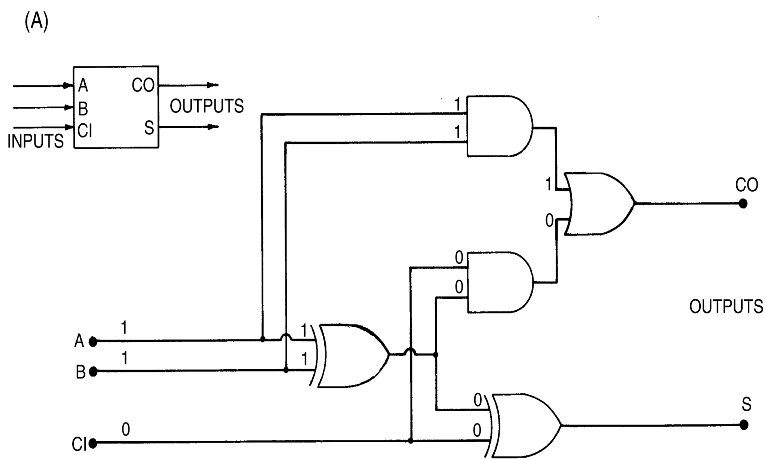
41. In order to provide an example of a working logic circuit an Adder circuit is explained in the following paragraphs.

42. As has been shown, when binary numbers are added it is almost always necessary to carry a digit to the next higher order column. For example, binary 1 plus binary 1 produces binary 10. Two single digits were added, producing a two-digit result (1+1 = 10), the 1 of the result being the digit carried to the next higher order column.

43. An Adder circuit is capable of accepting two inputs (the numbers to be added), plus a third input where necessary that is a carried digit from a previous lower order addition. Hence, a number of Adder circuits in series is required for the addition of large binary numbers.

44. Figure 9-13 shows the symbol for an Adder logic circuit (a) and the combination of EXCLUSIVE OR, AND and OR gates that make up such a circuit. Inputs A and B are the two bits to be added and input CI is the carried digit from the previous lower order column, where appropriate. Output S is the summation of the inputs and output CO is the bit to be carried to the next higher order column, where appropriate.

FIGURE 9-13
Adder Logic Circuit



45. Let us assume that the circuit is required to make the binary addition 1+1. Inputs A and B will both be 1, input CI will be 0, since there is nothing to be carried from a lower order column. By applying these inputs and following the process through each logic gate, using the truth tables, it will be seen that the outputs are CO = 1 and S = 0. 1+1 = 10.

Integrated Circuits

The Integrated Circuit

46. An integrated circuit (IC) is a complete electronic circuit containing transistors, diodes, resistors and capacitors along with their interconnecting conductors processed on and contained entirely within a single chip of silicon.

47. A process known as photolithography is used to print the actual circuits connecting the components. The conducting circuit is etched onto a silicon wafer by a process similar to that used in photographic developing. The circuit may be a relatively simple one, such as that making up an Adder, or a highly complex one containing an entire digital computer system.

48. Integrated circuits can be produced to carry out a wide variety of tasks, for example from a simple timing device to a voltage regulator or an amplifier. The applications are almost endless. The advantages of IC's are small size, low weight, reliability of operation.

Basic Radio Theory

Basic Radio Theory

1. Alternating current (AC) provides a more efficient means of conveying electrical energy than direct current (DC). With alternating current the electrons flow for half the total time in one direction, and for the other half of the total time in the opposite direction. The pattern of the change of direction of the current flow is **sinusoidal.**

2. It is not possible to produce an AC current flow from a battery. Low frequency AC may be produced by using alternators. The much higher frequencies required for radio and radar equipment are produced using oscillators, which electronically produce alternating current.

FIGURE 10-1
Simple AC Circuit

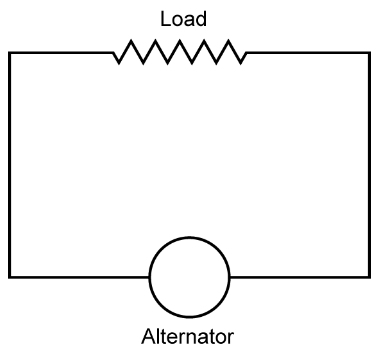
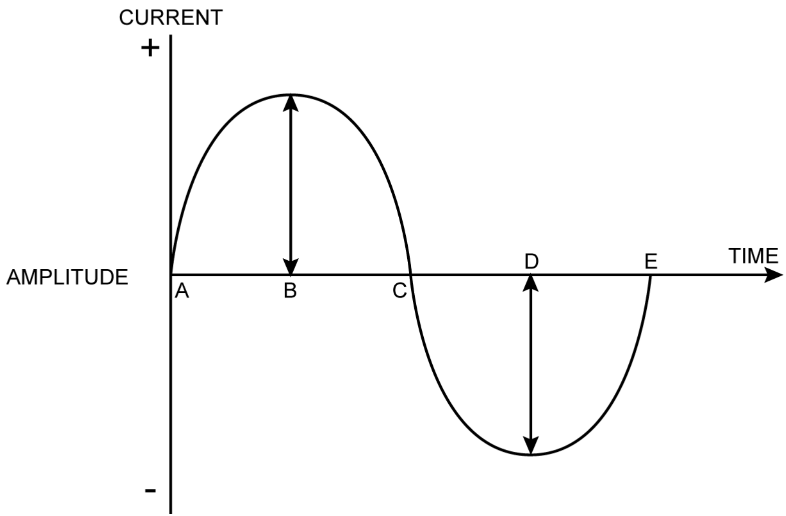


FIGURE 10-2
Current Flow in a Simple AC Circuit



3. In Figure 10-1 the battery in a DC circuit has been replaced by an alternator. Figure 10-2 shows the sinusoidal variation of current flow. Note that at times **A**, **C** and **E** no current flows, whilst at times **B** and **D** the current flow is at maximum, but in opposite directions. Note also that the **amplitude** of the wave is measured at its maximum value, at times **B** and **D**.

4. Time **A** to **E** at Figure 10-2 represents one complete cycle. The term **frequency** is used to denote the number of cycles occurring in one second. With the UK National Grid system the time taken for each cycle is one-fiftieth of a second, giving a frequency of 50 **cycles per second**, or 50 **Hertz** (50 Hz). Aircraft AC supply frequency is normally 400 Hz. The frequencies used for radio wave transmissions are very much higher than 50 Hz. In order to simplify matters the units shown below are used.

1,000 Hz	=	1×10^3 Hz	=	1 Kilohertz (1KHz)
1,000,000 Hz	=	1×10^6 Hz	=	1 Megahertz (1 MHz)
1,000,000,000 Hz	=	1×10^9 Hz	=	1 Gigahertz (1 GHz)

5. If a wire (or **antenna**) is supplied with alternating current, some of the power is radiated into space as electromagnetic energy. A similar wire (antenna), parallel to the first, will have a small alternating current induced in it as the radiated electromagnetic energy passes over it. The characteristics of the induced AC will be identical to those supplied to the transmitting wire. This is the basis of all radio transmitting and receiving systems.

6. As the frequency increases the time taken to complete each cycle decreases accordingly. Consequently it is necessary to simplify the units of time, as shown below.

$$\frac{1}{1,000}\text{ second} \quad = \quad 1 \times 10^{-3} \quad = \quad 1 \text{ millisecond (1 m sec)}$$

$$\frac{1}{1,000,000} \text{ second} \quad = \quad 1 \times 10^{-6} \quad = \quad 1 \text{ microsecond } (1\ \mu \text{ sec})$$

7. The frequencies used for radio and radar systems are delineated into convenient **bands**, each frequency band having, to a degree, its own unique characteristics, as will be seen shortly. Figure 10-3 shows the frequency bands which need concern us.

FIGURE 10-3
Frequency Bands

Band	Frequency Range
VLF Very Low Frequency	3 –30 KHz
LF Low Frequency	30 – 300 KHz
MF Medium Frequency	300 KHz – 3 MHz
HF High Frequency	3 – 30 MHz
VHF Very High Frequency	30 – 300 MHz
UHF Ultra High Frequency	300 MHz – 3 GHz
SHF Super High Frequency	3 – 30 GHz
EHF Extremely High Frequency	30 – 300 GHz

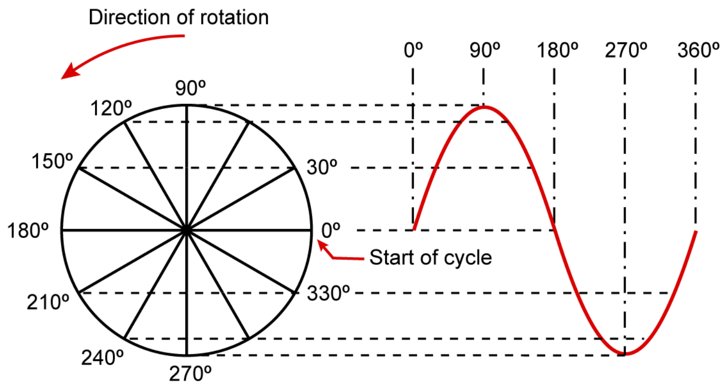
Phase Notation

8. When considering alternating current flow, or an electromagnetic force field, it is convenient to denote a particular point on the sine waveform using **phase notation**. Sine waves are constructed as shown at Figure 10-4, by tracing the progress of a given point on the circumference of a circle, the radius of which determines the amplitude of the wave. The 0° phase point always lies on the zero amplitude line at the start of the positive half of the sine wave.

9. Considering the construction it is easy to visualise phase notation. Of course the time taken for one complete 360° cycle (the horizontal axis of the graph) is dependent upon the frequency considered, the time being in fact 1/F seconds.

10. Phase notation is important since several of the equipments subsequently studied employ **phase comparison** techniques in order to determine aircraft position.

FIGURE 10-4
Construction of a Sine Wave



Phase Comparison

11. It is necessary to adopt a **precise** approach to the construction of a diagram showing a fixed phase difference between two sine waves. Of course, if the phase difference is to remain fixed, **the sine waves must be of exactly the same frequency.**

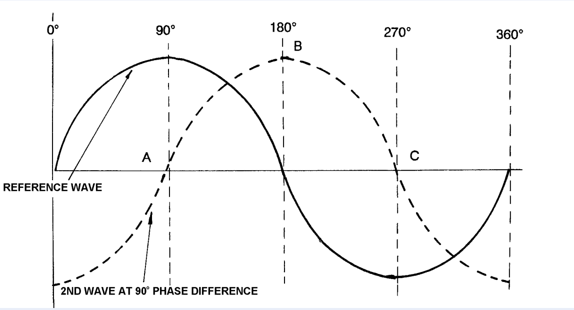
EXAMPLE 10-1

EXAMPLE

Draw a diagram to illustrate two sine waves which are of the same amplitude but 90° apart.

SOLUTION

First draw one of the sine waves as a reference waveform, the solid sine wave shown below.



In this case the phase difference is 90° and so the second waveform starts at the 90° phase point of the reference waveform, (point A). Similarly when the reference wave is at the 180° phase point the second waveform will be at its (180 – 90) 90° phase point, (position B). When the reference wave is at is 270° phase point the second waveform will be at its (270 – 90) 180° phase point, (position C) and so on.

12. Figure 10-5 and Figure 10-6 show two further examples of phase comparison diagrams.

FIGURE 10-5

Two Sine Waves, 135° Phase Difference

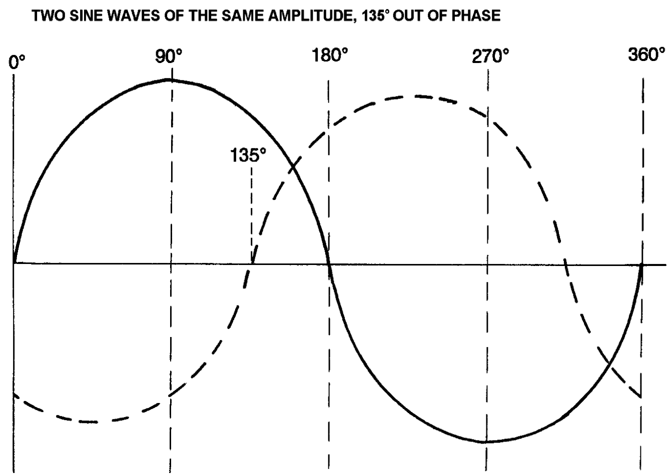
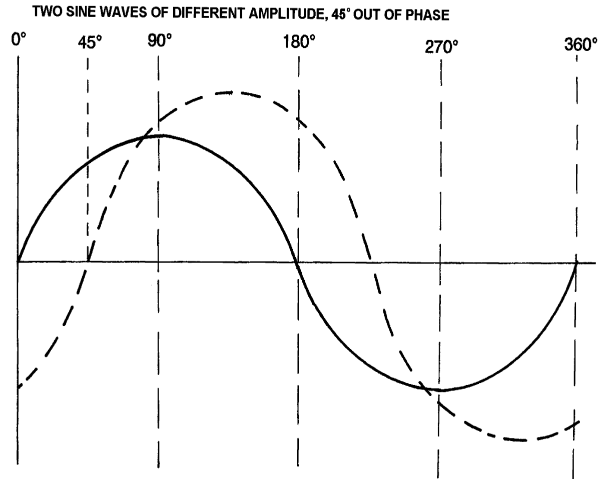


FIGURE 10-6
Two Sine Waves, 45° Phase Difference



Wavelength

13. Once an electromagnetic wave has left the transmitting aerial it is assumed to travel at the speed of light (C), which is:

300,000,000 metres per second

or

3×10^8 m/sec

14. The wave will travel a given distance in the small but finite time taken to transmit one complete cycle of energy, and this distance is known as the **wavelength** (λ). The higher the frequency the shorter the length of time required to transmit one complete cycle, and consequently the shorter the wavelength.

The relationship between frequency and wavelength is given by the formula:

$$F = \frac{C}{\lambda}$$

or

$$\lambda = \frac{C}{F}$$

where

F is the frequency in hertz

λ is the wavelength in metres

C is the speed of propagation, 3×10^8 m/sec

EXAMPLE 10-2

EXAMPLE

Given a wavelength of 30 metres, determine the frequency.

SOLUTION

$$F = \frac{C}{\lambda} \quad \text{or} \quad F = \frac{C}{\lambda}$$

$$F = \frac{300{,}000{,}000 \text{ m/sec}}{30 \text{ m}} \quad \text{or} \quad F = \frac{3 \times 10^8 \text{ m/sec}}{30 \text{ m}}$$

$$F = 10{,}000{,}000 \text{ Hz} \quad \text{or} \quad F = 1 \times 10^7 \text{ Hz}$$

$$F = 10 \text{ Mhz} \qquad F = 10 \text{ Mhz}$$

EXAMPLE 10-3

EXAMPLE

Given a wavelength of 35 kilometres, calculate the frequency.

SOLUTION

$F = \frac{C}{\lambda}$	or $F = \frac{C}{\lambda}$
$F = \frac{300{,}000{,}000 \text{ m/sec}}{35{,}000 \text{ m}}$	or $F = \frac{3 \times 10^8 \text{ m/sec}}{35 \times 10^3 \text{ m}}$
$F = 8571$ Hz	or $F = 0.08571 \times 10^5$ Hz
$F = 8.571$ Khz	or $F = 8.571$ Khz

EXAMPLE 10-4

EXAMPLE

Given a wavelength of 4.5 cm, calculate the frequency.

SOLUTION

$$F = \frac{C}{\lambda} \quad \text{or} \quad F = \frac{C}{\lambda}$$

$$F = \frac{300{,}000{,}000 \text{ m/sec}}{0.045 \text{ m}} \quad \text{or} \quad F = \frac{3 \times 10^{8} \text{ m/sec}}{4.5 \times 10^{-2} \text{ m}}$$

$$F = 6{,}666{,}666{,}666 \text{ Hz} \quad \text{or} \quad F = 0.667 \times 10^{10} \text{ Hz}$$

$$F = 6.67 \text{ Ghz} \quad \text{or} \quad F = 6.67 \text{ Ghz}$$

EXAMPLE 10-5

EXAMPLE

Given a frequency of 15 KHz, calculate the wavelength.

SOLUTION

$$\lambda = \frac{C}{F} \quad \text{or} \quad \lambda = \frac{C}{F}$$

$$\lambda = \frac{300{,}000{,}000 \text{ m/sec}}{15{,}000 \text{ Hz}} \quad \text{or} \quad \lambda = \frac{3 \times 10^8 \text{ m/sec}}{15 \times 10^3 \text{ Hz}}$$

$$\lambda = 20{,}000 \text{ m} \quad \text{or} \quad \lambda = 0.2 \times 10^5 \text{ m}$$

$$\lambda = 20 \text{ km} \quad \text{or} \quad \lambda = 20 \text{ km}$$

EXAMPLE 10-6

EXAMPLE

Given a frequency of 9.7 GHz, calculate the wavelength.

SOLUTION

$$\lambda = \frac{C}{F} \quad \text{or} \quad \lambda = \frac{C}{F}$$

$$\lambda = \frac{300{,}000{,}000 \text{ m/sec}}{9{,}700{,}000{,}000 \text{ Hz}} \quad \text{or} \quad \lambda = \frac{3 \times 10^{8} \text{ m/sec}}{9.7 \times 10^{9} \text{ Hz}}$$

$$\lambda = 0.031 \text{ m} \quad \text{or} \quad \lambda = 3.1 \times 10^{-2} \text{ m}$$

$$\lambda = 3.1 \text{ cm} \quad \text{or} \quad \lambda = 3.1 \text{ cm}$$

15. Figure 10-7 reproduces the table of radio frequency bands previously shown at Figure 10-3 but now includes the wavelengths pertinent to the various frequencies shown.

FIGURE 10-7
Frequency Bands and Associated Wavelengths

Band	Frequency Range	Wavelength Range	Wavelength Denomination
VLF Very Low Frequency	3 – 30 KHz	100 km – 10 km	Myriametric
LF Low Frequency	30 – 300 KHz	10 km – 1 km	Kilometric
MF Medium Frequency	300 KHz – 3 MHz	1 km – 100 m	Hectometric
HF High Frequency	3 –30 MHz	100 m – 10 m	Decametric
VHF Very High Frequency	30 – 300 MHz	10 m – 1 m	Metric
UHF Ultra High Frequency	300 MHz – 3 GHz	1 m – 10 cm	Decimetric
SHF Super High Frequency	3 – 30 GHz	10 cm – 1 cm	Centimetric (Microwave)
EHF Extremely High Frequency	30 – 300 GHz	1 cm – 1 mm	Millimetric

Radio Wave Emission

16. If an alternating current at a suitable frequency is fed to a transmitting element (**aerial** or **antenna**) of the required dimensions, the current flow through the aerial will result in an **electromagnetic force field** being radiated from the aerial. The **lower the frequency** the **larger the aerial** required for efficient transmission (and reception).

17. A suitable frequency for radio transmission is obtained by means of a radio frequency **oscillator.** This is then amplified and fed to the transmitting aerial via a modulator, which superimposes the signal to be transmitted upon the radio frequency (more of which later).

Oscillators

18. The most common oscillators are circuits, which produce a simple sinusoid at a particular frequency. They may be fixed in frequency (FIXED TUNE) or variable. Mechanical tuning can be used to vary frequency fairly slowly. Electronic tuning will allow very rapid variation of frequency and fine control. (VCO = Voltage Controlled Oscillator).

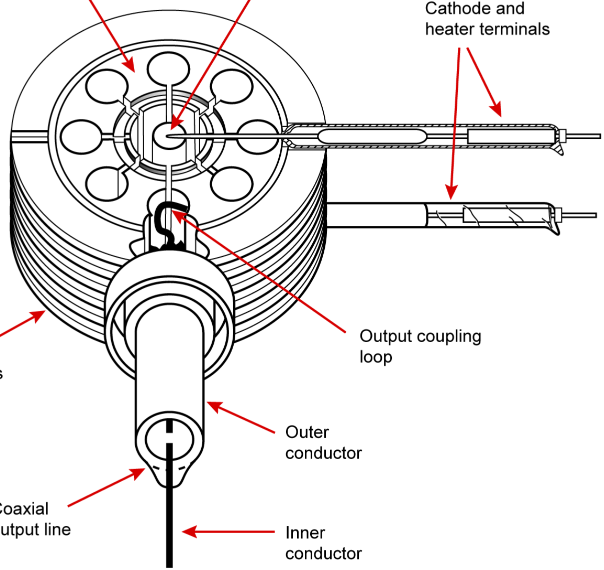
The Magnetron

19. Magnetrons are cross-field oscillators. Electrons, emitted from a central cathode, move out radially under the influence of an electrical (E) field towards the anode block. The E field is associated with a high voltage applied between anode and cathode. A powerful permanent magnet is used to supply a strong magnetic field at right angles to the electric field. Under the influence of these crossed-fields, electrons travel in curved paths in the space between cathode and anode. The anode block is made of copper and contains a number of resonant cavities.

20. The rotating cloud of electrons is made to move in synchronism with a radio wave travelling round the anode block structure. The Magnetron is an amplifier with total feedback; the output is connected directly to the input. This feedback results in oscillation and a pick-up loop in one of the cavities delivers the signal to an output coaxial line or waveguide.

21. Pulse outputs of several MW and CW outputs of a few KW are common. **Magnetrons are often used as the master oscillator in pulse radars.** Output frequency is determined by the shape and size of the resonant cavities. Figure 10-8 shows the general construction of a magnetron.

FIGURE 10-8
Schematic of a Magnetron



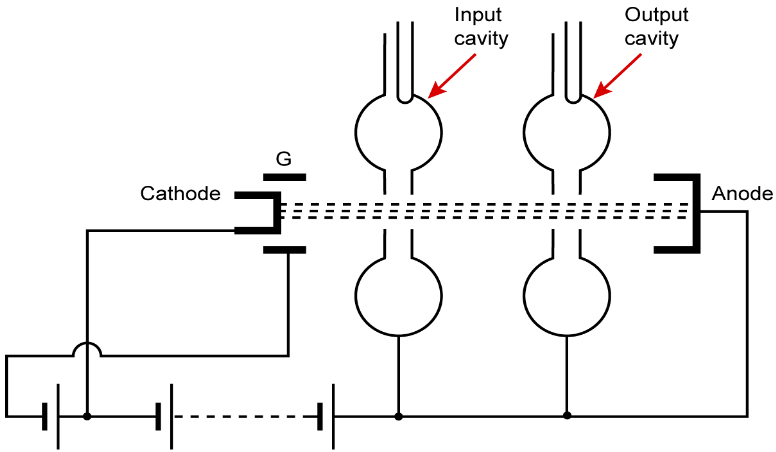
The Klystron Amplifier

22. Figure 10-9 is a schematic diagram of a very simple klystron amplifier. A powerful electron beam passes between an electron gun on the left and a collecting electrode on the right. The beam passes through hollow cavities which are tuned to support electro-magnetic fields alternating at the signal frequency. An input signal enters the input cavity along a waveguide or coaxial cable and creates an alternating field pattern in the cavity. This field disturbs the uniform electron beam creating ripples or concentrations of electrons which increase in intensity as they move along the beam. As they pass through the output cavity, the ripples on the electron beam induce alternating fields, which are stronger replicas of those fed to the input cavity.

23. In practice there are usually a number of intermediate cavities (multi-cavity klystron). These result in a larger gain and wider bandwidth than would be possible using just two cavities. A number of coils carrying direct current surround the valve in order to stop the electron beam from spreading. A longitudinal magnetic field exerts a focusing effect.

24. CW power outputs of several KW are readily available and pulse outputs of several MW. It is important to notice that in a pulse radar application the input to the klystron may be a continuous wave. The pulses are created by turning the amplifier on and off, which ensures a coherent pulse stream. Klystron amplifiers are entirely unsuitable for use as RF amplifiers in radar receivers as they are far too noisy for this application.

FIGURE 10-9
Schematic of a Twin Cavity Klystron



The Reflex Klystron Oscillator

25. A two-cavity klystron amplifier can be made to oscillate by connecting the output cavity to the input cavity in the correct manner. The reflex klystron oscillator features only one cavity but this cavity is used twice. A stream of electrons from the cathode passes through the cavity and is then turned back by the reflector electrode which is maintained at a negative potential. The stream passes through the cavity a second time. It is necessary to adjust the reflector voltage in order to achieve oscillation. The alternating signal in the cavity is extracted along a coaxial cable or waveguide. No magnetic focusing field is required.

26. A low power CW output is available from the reflex klystron, typically a few tens of mW. It can be used as a low power transmitter but is more commonly found in the role of the **local oscillator in a microwave superhet receiver.**

Quartz Crystal Controlled Circuits

27. The basic requirements for a VHF aeromobile band ground transmitter are that the equipment is capable of radiating an amplitude modulated signal (see later) of high stability containing low noise levels and that it is capable of being controlled remotely via telephone lines.

28. Frequency control is by quartz crystal. A number of crystalline substances have the ability to transform mechanical strain into an electrical charge and vice versa. This property is known as the piezoelectric effect.

29. If a small plate or bar of such material is placed between two conducting electrodes it will be mechanically strained when the electrodes are connected to a voltage. Conversely, if the crystal is compressed between two electrodes a voltage will develop across those electrodes, thus piezoelectric crystals can be used to transform electrical to mechanical energy and vice versa. This effect is frequently used in inexpensive microphones, gramophone pick-ups, and in some headphones and loudspeakers. For these purposes crystals of Rochelle salt are used. Crystalline plates also exhibit a mechanical resonance of frequencies ranging from a few thousand to many millions of cycles, the frequency depending on the material of the crystal, the angle at which the plate was cut from the crystal and the physical dimensions of the plate. Due to the piezoelectric effect the plates also exhibit an electrical resonance and act as a very accurate, and highly efficient, tuned circuit. **Such crystals are used in radio equipment in high-stability oscillator circuits and in highly selective filters**.

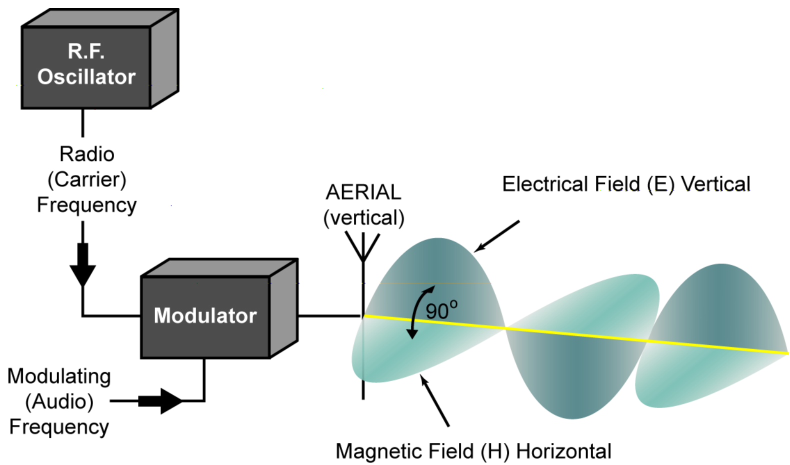
30. In ground transmitters a quartz crystal is used as the frequency determining element, controlling the oscillator circuit. Until recently it has not been possible to manufacture a crystal that will resonate in the 110 MHz to 136 MHz aeromobile band and even now these crystals are fragile and expensive. A crystal on a sub-multiple of the required frequency is therefore often used, followed by circuits designed to multiply the oscillator frequency to the final operating frequency. In early days, crystal frequencies of 5 MHz to 7 MHz were used requiring multiplication factors of 18 or 24 times, but more recently, improved techniques in cutting and mounting have produced much higher frequencies, requiring far less frequency multiplication. The crystal multiplier chain is followed by amplification stages which generate the necessary output power.

Radio Transmitters

31. A simplified block diagram of a radio transmitter, and the electromagnetic field emanating from the aerial, is shown at Figure 10-10.

32. As the name suggests, an electromagnetic force field consists of two distinct elements, the **electric field** (the E field) and the **magnetic field** (the H field). These two force fields exist in planes which are at right angles to each other and which are mutually at right angles to the direction of travel of the radio wave. The amplitude of each force field increases and collapses at the same rate as the alternating current, which is producing the radiated signal.

FIGURE 10-10
Simplified Block Diagram of a Radio Transmitter



Skin Effect

33. The current, which flows to support the H field in an aerial, would exist only on the surface of a perfect conductor. In the case of a practical conductor, the value of current is high at the surface and falls exponentially as distance within the conductor increases. For a good conductor, such as copper, the current value falls very rapidly confining the current, in effect, to a thin skin at the surface. This is the effect known as SKIN EFFECT. Note that skin depth decreases as frequency increases. For copper, skin depth is approximately 1 micron, at a frequency of 3 GHz.

Polarisation

34. The term **polarisation** describes the plane of oscillation of the **electrical field** of an electromagnetic wave. At Figure 10-10 the electrical field of the electromagnetic wave lies in the **vertical plane** and the radio wave is said to be **vertically polarised**. The electromagnetic wave shown at Figure 10-10 is transmitted from a vertical antenna and, being vertically polarised, requires a vertical antenna at the receiver to ensure efficient reception. Similarly, a signal transmitted by a horizontal aerial would be horizontally polarised (the electrical part of the electromagnetic field would oscillate in the horizontal plane), and the receiver aerial would need to be horizontal for optimum reception.

35. In general, a vertically polarised signal will achieve better ranges at frequencies up to and including VHF, however horizontally polarised signals will achieve better ranges at UHF and above.

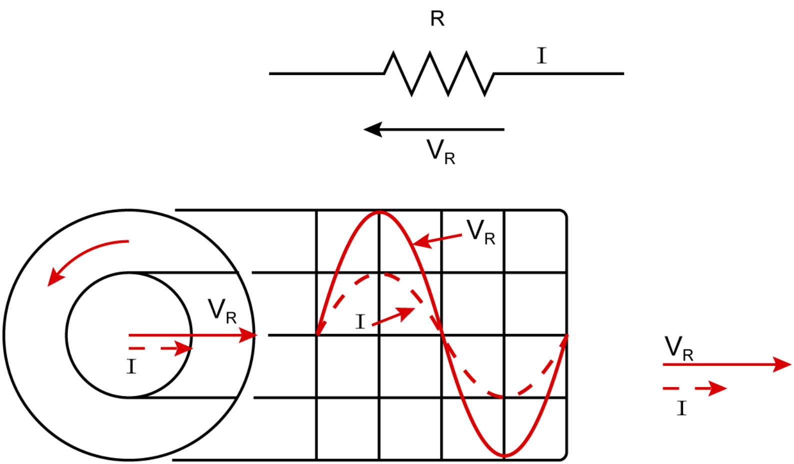
Radio Wave Reception

Electrical Resonant Circuits

36. The first stage of most radio receivers comprises of some sort of electrical resonant circuit, the workings of which will need to be understood by the student. The following paragraphs describe how such a circuit is constructed and its method of operation.

Pure Resistive Circuit

FIGURE 10-11
Phase Diagrams for Pure Resistance



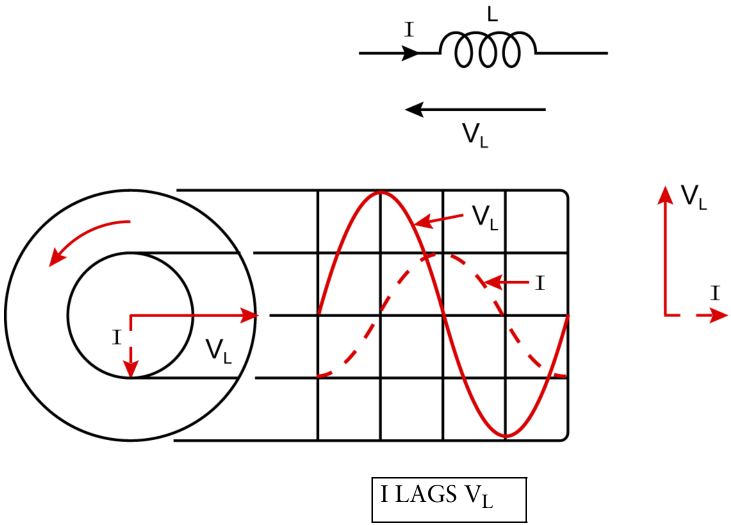
37. An ac flows through a resistance of R ohms, see Figure 10-11. From ohms law we know that V_R = IR. Thus for a pure resistance, the potential difference across it, V_R, is IN PHASE with the current flow through it.

I IN PHASE with V_R

Pure Inductive Circuit

38. Figure 10-12 shows the curve for a sinusoidal current (I), which is flowing through a coil of inductance L henrys. It can be shown that the current lags the applied voltage by 90°.

FIGURE 10-12
Phase Diagram for Pure Inductance



Inductive Reactance. In a pure resistance the ratio of voltage to current gives the resistance R. In a pure inductance the ratio of voltage to current is:

$$\frac{V_L}{I} = 2\pi FL$$

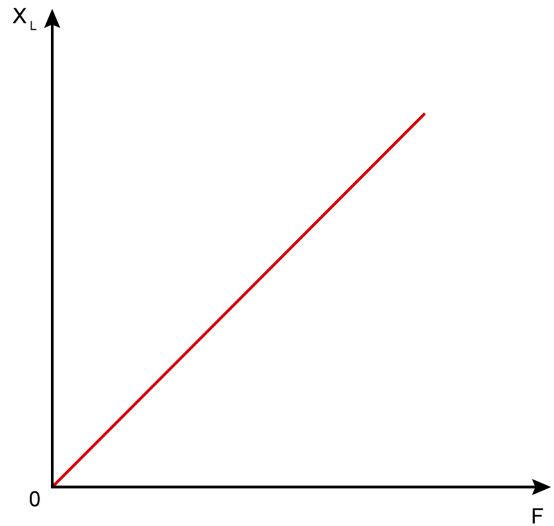
$\frac{V_L}{I}$ is called the INDUCTIVE REACTANCE, X_L, and is measured in ohms

Reactance/Frequency Graph. Figure 10-13 is the reactance – frequency graph for an inductor.

$$X_L = 2\pi FL$$

$$X_L \alpha F$$

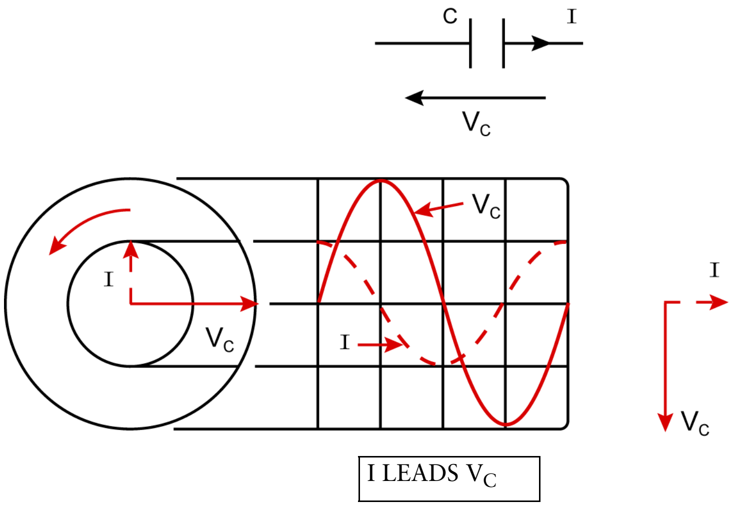
FIGURE 10-13
Effect of Frequency on Inductive Reactance



Pure Capacitive Circuit

39. Figure 10-14 shows the curve for a voltage (V_C) developed across a pure capacitor of capacitance C farads. It can be shown that the current leads the applied voltage by 90°.

FIGURE 10-14
Phase Diagram for Pure Capacitance



NOTE:

Students may find the word 'CIVIL' a useful aid memoir in that in a capacitor current leads voltage, in an inductor voltage leads current.

Capacitive Reactance. In a pure capacitance the ratio of voltage to current is:

$$\frac{V_C}{I} = \frac{1}{2\pi FC}$$

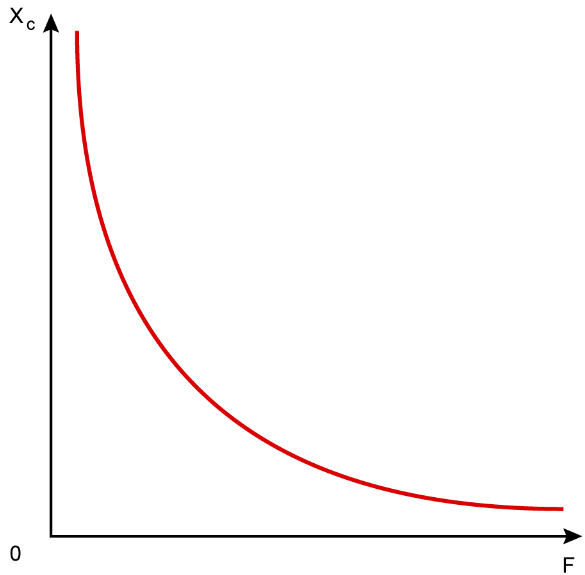
$\frac{V_C}{I}$ is called the CAPACITIVE REACTANCE, X_C, and is measured in ohms

Reactance-Frequency Graph. Figure 10-15 is the reactance frequency graph for a capacitor.

$$X_C = \frac{1}{2\pi FC}$$

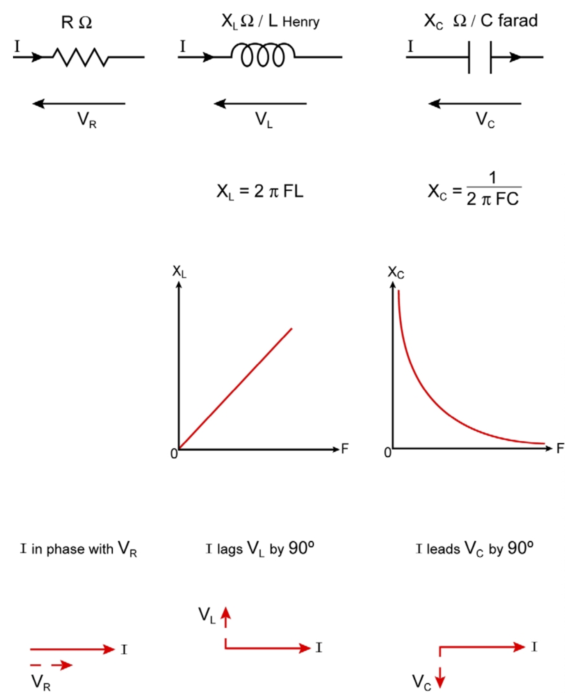
$$\boxed{X_C \alpha \frac{1}{F}}$$

FIGURE 10-15
Effect of Frequency on Capacitive Reactance



Summary

FIGURE 10-16
Summary of Pure Resistive, Capacitive and Inductive Circuits

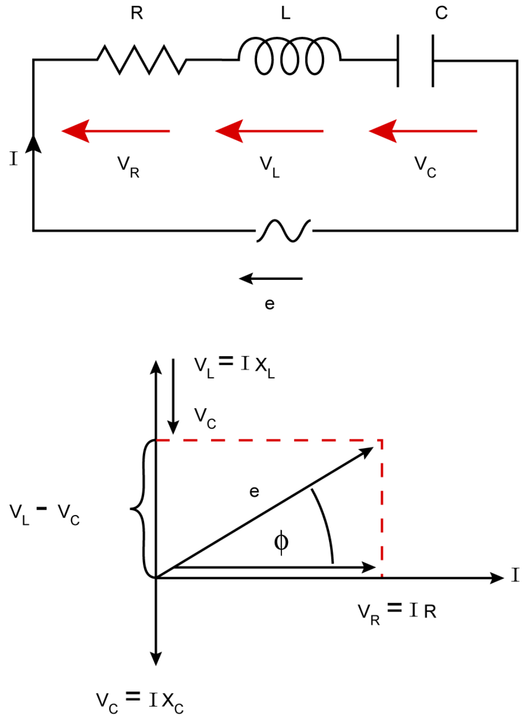


Series Circuit

40. A coil, of self inductance L henrys and resistance R ohms, is connected to a capacitor of C farads. An emf of e volts and of variable frequency is connected to the circuit. Figure 10-17 shows the circuit details.

FIGURE 10-17

Series LCR Circuit
and Phase Diagram



41. A phase diagram for the circuit is also shown in Figure 10-17. In this the potential difference (pd) across L is taken as greater than that across C and therefore the applied voltage leads the input current by the phase angle, ϕ.

Impedance

42. $\frac{E}{I}$ is called the IMPEDANCE, Z of the circuit and is measured in ohms.

$$Z = \sqrt{R^2 + (X_L \sim X_C)^2}$$

Resonance

43. When $V_L = V_C$, $\phi = 0$ i.e., the input current is IN PHASE with the applied voltage. In this special condition, the circuit is said to be at RESONANCE.

As $V_L = V_C$ then $I X_L = I X_C$ i.e. $X_L = X_C$

Z is a minimum and is equal to R (from above equation).

Resonant Frequency

44. The frequency at which $X_L = X_C$ may be determined as follows:

$$X_L = X_C$$

$$2\pi FL = \frac{1}{2\pi FC}$$

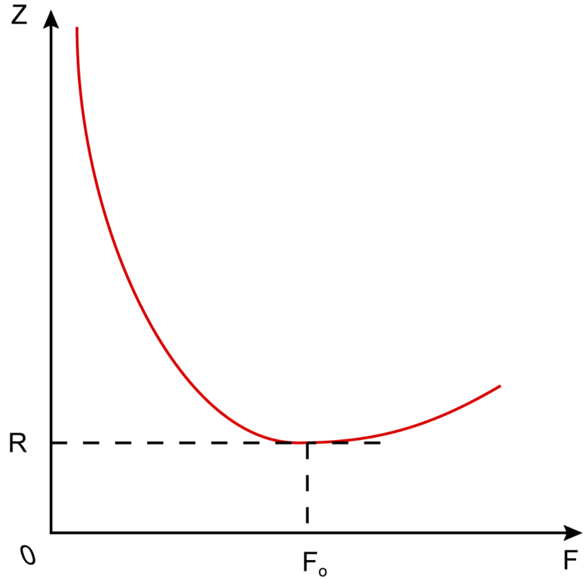
This value of frequency is denoted by F_O.

$$F_O = \frac{1}{2\pi\sqrt{LC}}$$

Response Curve

45. At frequencies other than the resonant frequency, V_L is not equal to V_C and the impedance of the circuit is higher than that at resonance, see Figure 10-18. For an applied voltage of constant amplitude, the current (rms value, I) varies as the frequency of the supply changes, see Figure 10-19. The curve shown in Figure 10-19 is called a **RESPONSE CURVE**.

FIGURE 10-18
Variation of Impedance with Frequency



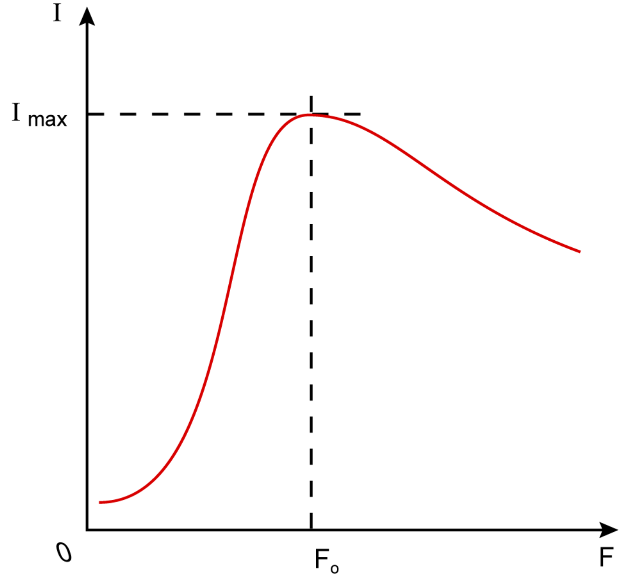
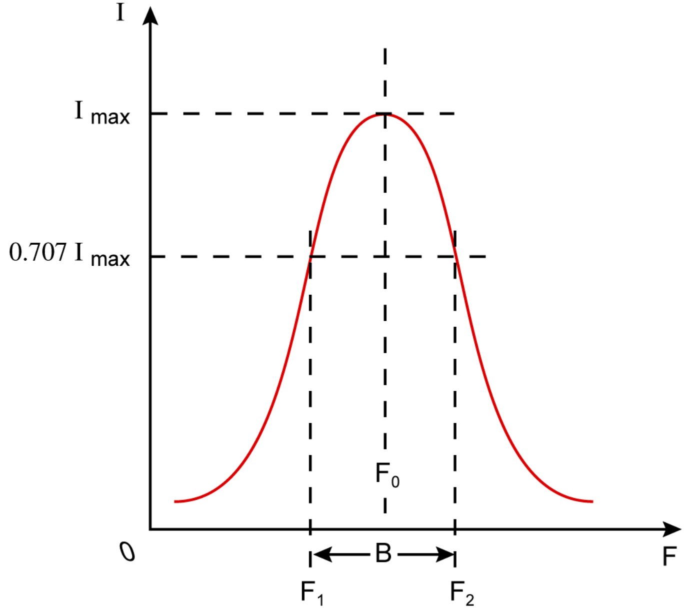


FIGURE 10-19
Variation of Current with Frequency

Bandwidth

46. The BANDWIDTH, B, of the circuit is the difference between the two frequencies either side of resonance at which the **current** has fallen to 0.707* of its maximum value, see Figure 10-20.

FIGURE 10-20
Bandwidth in a
Series LCR Circuit



The bandwidth, $B = F_2 - F_1$

NOTE:

$$\frac{1}{\sqrt{2}} = 0.707$$

Selectivity

47. The sharpness of the response curve over a range of frequencies near resonance indicates the SELECTIVITY of the circuit. Selectivity is the ability of a tuned circuit to respond strongly to its resonant frequency and to give a poor response to other frequencies either side of resonance. A sharp response curve indicates high selectivity; poor selectivity is indicated by a flat response curve. For good selectivity, a circuit should have a low value of R and a high L\C ratio.

FIGURE 10-21
Narrow Bandwidth/Good Selectivity

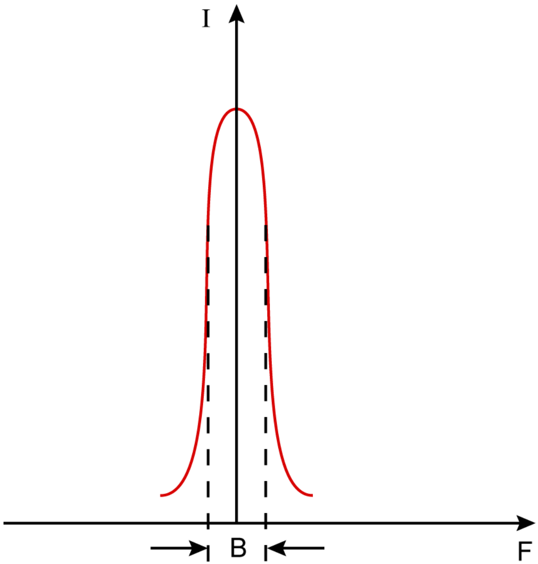
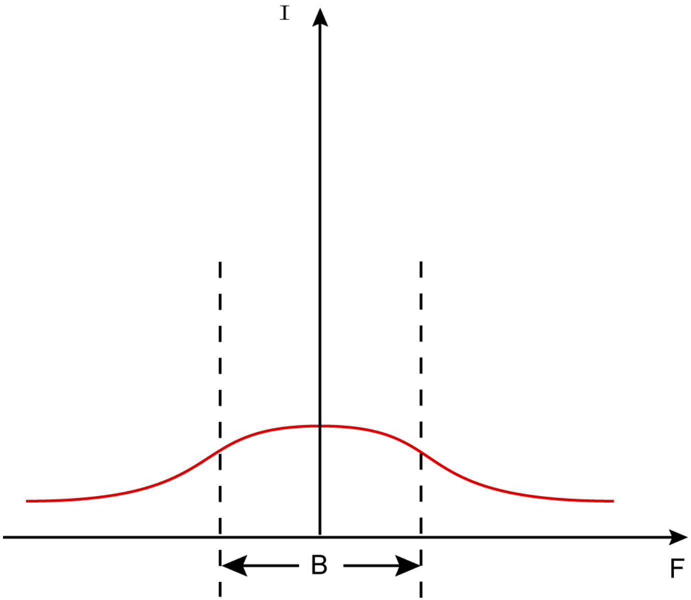


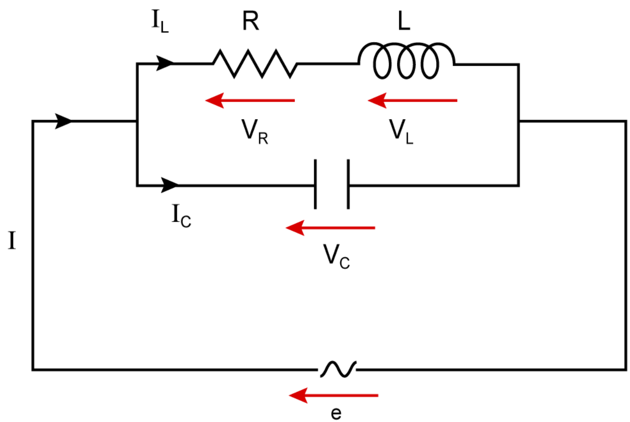
FIGURE 10-22
Wide Bandwidth/
Poor Selectivity



Parallel Circuit

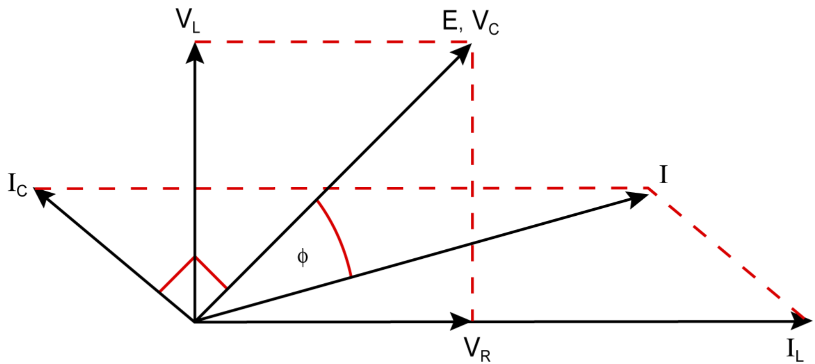
48. A coil, of self-inductance L henrys and resistance R ohms, is connected across a capacitor of C farads. An emf of e volt and of variable frequency is connected to the circuit, see Figure 10-23. This type of parallel ac circuit is very common in radio equipments and has many important applications.

FIGURE 10-23
Parallel LCR Circuit



49. The pd across the coil, the phasor sum of V_R and V_L, is equal to the pd across the capacitor, V_C. The supply current is the vector sum of I_L and I_C. A phase diagram for the circuit is shown in Figure 10-24. For the condition shown, the supply current LAGS the applied voltage by a phase angle of ϕ degrees and the circuit is therefore INDUCTIVE ($I_L > I_C$).

FIGURE 10-24
Phase Diagram for a Parallel LCR Circuit



50. For a certain value of frequency, I is in phase with E, i.e. the circuit is at RESONANCE. Once again it can be shown that the value of the Resonant Frequency (F_O) is given by:

$$F_O = \frac{1}{2\pi\sqrt{LC}}$$

51. At resonance, the impedance of a parallel circuit is a maximum and the supply current is a minimum. This circuit arrangement is called a REJECTOR circuit.

Selectivity

52. This is defined in the same way as for a series tuned circuit; namely, the ability of the circuit to respond strongly to the required signal, which is at the resonant frequency, and to give a poor response to all other signals. At resonance, the supply current is a minimum and the impedance is maximum. If the circuit is mis-tuned either side of resonance, e.g. by altering the value of C, the supply current increases and the impedance decreases, see Figure 10-25 and Figure 10-26.

FIGURE 10-25

Variation in Current with Frequency

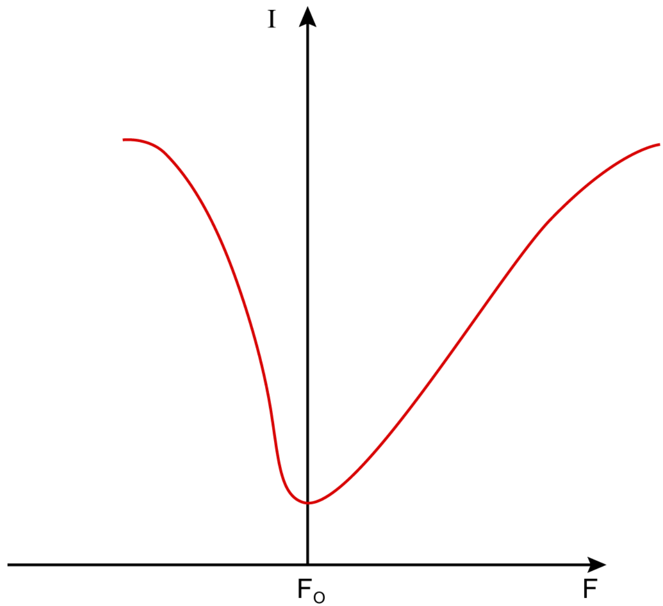
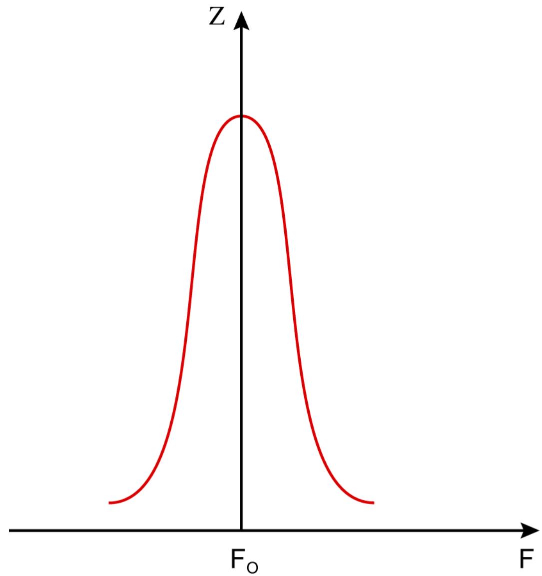


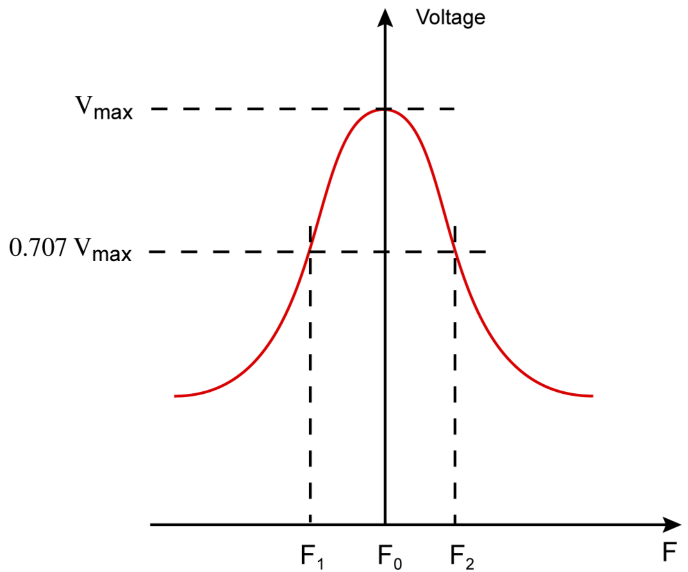
FIGURE 10-26
Variation in Impedance with Frequency



Bandwidth

53. Parallel circuits reject signals near to the resonant frequency. The BANDWIDTH of the circuit is the difference between the two frequencies, either side of resonance, at which the **voltage** has fallen to 0.707 of its maximum value. See Figure 10-27.

FIGURE 10-27
Bandwidth in a Parallel LCR Circuit



Bandwidth B = $F_2 - F_1$

In some circuits a wide bandwidth is required and one way to achieve this is to connect a resistor across the parallel circuit.

Demodulation

54. Having been filtered by the resonant circuit at the front end of a radio receiver the incoming signal is then amplified before undergoing a process called **demodulation**. Demodulation is the point at which the intelligence/information is separated from the carrier wave and therefore made available to the recipient.

Fading

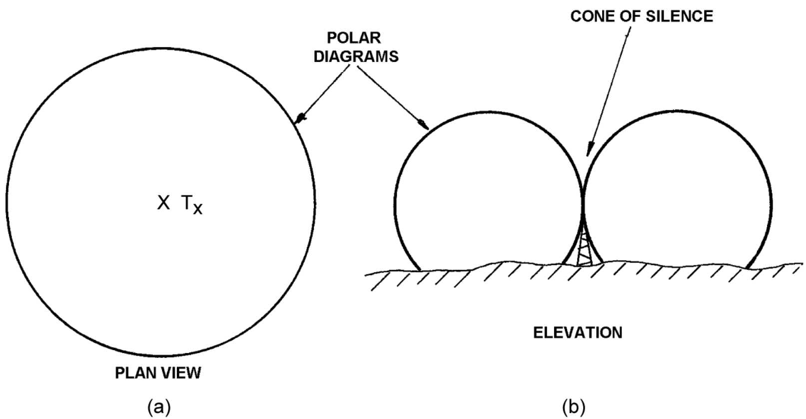
55. Occasionally radio reception suffers from the effects of fading. This is caused by the radio wave travelling by two alternative routes (e.g. surface wave and skywave) before meeting at the receiving aerial. If the two signals arrive in phase they will reinforce each other, if they arrive in anti-phase they will cancel out; if therefore, in the above example, the ionosphere is fluctuating in intensity and height the path length taken by a skywave will vary continuously, leading to fading in and out of the signal.

Polar Diagrams

56. A transmitter polar diagram is simply a pictorial presentation of the strength of the electromagnetic energy field in all directions from the transmitter aerial.

57. Figure 10-28 shows the polar diagrams in plan and side view associated with a single vertical antenna. Figure 10-28 (a) shows that the aerial is propagating radio waves omni-directionally in the horizontal plane, that is to say that at any point around the circumference of the circle, the transmitted signal strength is the same. Notice at Figure 10-28(b) that the vertical antenna is not transmitting vertically upwards. This gives rise to the cone of silence which occurs overhead, for example, VOR transmitters. Appreciate that transmitter polar diagrams do not define the limit of coverage of the transmitted signal, but rather points of equal signal strength.

FIGURE 10-28
Transmitter Polar Diagrams - Single Vertical Antenna



58. It is equally convenient to produce polar diagrams for receiver aerial arrays. Now the situation is reversed in that a signal of a given strength which is transmitted from any point on the polar diagram will induce a current of constant amplitude to flow in the receiver aerial.

59. By modifying the antenna configuration it is possible to adjust the shape of the polar diagram. The ILS transmitters give polar diagrams which are lobe shaped, whereas VOR transmitters produce a polar diagram which is known as a limacon, and which is in fact made to rotate, but more of this later in this section.

The Ionosphere

60. The ionosphere exists in the upper atmosphere above the mesospheric layer. The gaseous composition of the ionosphere is such that ultra-violet rays from the sun cause electrons to become separated from their parent atoms. The atoms which are consequently left with a **positive** charge are known as **ions**. The ionosphere is most intense, that is to say it contains the greatest concentrations of ions, during the daylight hours. During the night the displaced electrons tend to re-combine with their parent atoms, resulting in some degree of de-ionisation.

61. The areas of ionised gases tend to exist in distinctive layers, known as the **D**, **E** and **F** layers.

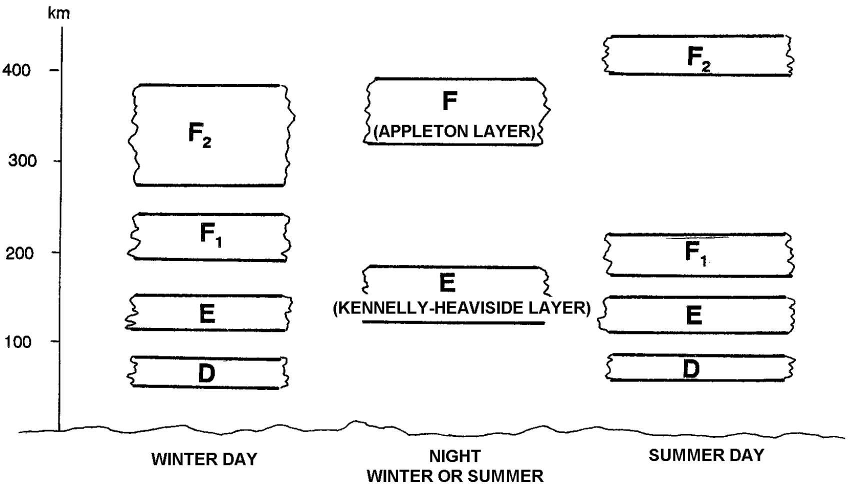
62. During the daylight hours it is normal for four distinct layers to become established, **D**, **E** **F1** and **F2**, see Figure 10-29. The height and thickness (depth) of these layers will depend upon such factors as latitude, time of the year and sun spot activity. Ionisation intensity **increases** with **increase** in height and therefore the **F** layer(s) tend to be stronger than the lower layers.

63. During the hours of darkness the four layers tend to merge into only two distinctive layers, the **E** and the **F** layers, again see Figure 10-29.

64. It is important to appreciate that, at night, the ionospheric layers are generally speaking **higher**, and are **less intensely ionised** than during the day.

65. The significance of the ionosphere is that radio waves, on entering a layer of ionised gases, will tend to both **bend** and **weaken**.

FIGURE 10-29
The Ionospheric Layers



66. The average heights of the various ionospheric layers are as follows:

(a) D Layer 75 km.

(b) E Layer 125 km.

(c) F Layer 225 km.

Attenuation

67. As any radio wave travels away from the transmitter it becomes weaker, or is **attenuated**, due to some or all of the reasons discussed below.

Range from the Transmitter

68. Assume that an electromagnetic radio wave is being transmitted in all directions, that is to say it is being transmitted **omni-directionally**. The energy is contained within a spherical wave front in this case, and as the distance from the transmitter increases the area of the wave front also increases. Consequently as the range from the transmitter increases the **field strength** of the signal decreases. Field strength is inversely proportional to the square of the range and it can be proved that, to double the range of a transmission the transmitter power must be quadrupled.

Surface Attenuation

69. If a radio wave is constrained to travel close to the Earth's surface, the wave will attenuate by virtue of its contact with the surface. Two important facts are worthy of note. Firstly, the rate of surface attenuation is likely to be three times greater over land than it is over sea. Secondly, the rate of surface attenuation **increases** as the frequency of signal **increases**.

Ionospheric Attenuation

70. As a radio wave travels through any of the ionospheric layers it will be weakened by the positively charged ions. In the extreme case a radio signal may be totally attenuated within an intensely ionised layer. The rate of ionospheric attenuation **decreases** as the transmitted frequency **increases**.

Atmospheric Attenuation

71. A radio wave travelling through unpolluted and unsaturated air would not suffer any attenuation due to the medium through which its travelling. However the atmosphere contains solid particle pollutants and water in both liquid and solid form, and these particles, droplets and ice crystals will **reflect** and **scatter** radio waves at sufficiently high frequencies.

72. The purist might say that the signal is not in fact attenuated, merely redirected in an unwanted direction. The fact remains that, if the signal is scattered, it may never reach the receiver at a given range, and is therefore of no use.

73. When considering **atmospheric signal scatter** it should be appreciated that the particles, droplets or ice crystals must be of **significant size** when compared to the **wavelength of the transmitted signal.** Radio signals in the **EHF** band may suffer considerable attenuation under these conditions.

Refraction

74. It is normal to assume that radio waves travel in straight lines, that is to say along great circle paths. Any departure from such a straight line path is said to exist as a result of **refraction** of the radio wave.

75. The principal causes of refraction are discussed below.

Ionospheric Refraction

76. Radio waves tend to bend, or refract, as they travel through any of the ionospheric layers. The rate at which this refraction occurs **decreases** as the frequency of the radio signal **increases**.

Coastal Refraction

77. It was previously convenient to accept that radio waves travel at a constant speed. In fact radio waves travelling close to the Earth's surface will travel slightly faster over the sea than over the land. In consequence any radio wave crossing a coastline at other than 90° will bend slightly towards the land mass and this phenomenon is termed **coastal refraction.** Again, the degree of refraction **decreases** as the frequency of the radio signal **increases**.

Diffraction

78. In the low frequency and medium frequency bands radio waves tend to refract to an extent such that they remain in contact with the Earth's surface, despite the curvature of the Earth. There are several theories as to why this occurs, suffice to say that is a very useful phenomenon since it increases the surface range of these frequencies.

Atmospheric Refraction

79. Atmospheric refraction sometimes occurs in certain meteorological conditions, and produces a situation known as **ducting.** This is discussed in detail at a later stage.

Propagation Paths

80. Six of the possible paths which may be taken by a radio wave are discussed below.

The Direct Wave

81. This is the simplest case, and is **normally** the only possible path for radio waves in the **VHF band or above.** This is because, at VHF and above, a signal which remains close to the ground (a surface wave) will be totally attenuated over a very short distance. Conversely, at these frequencies a signal which is beamed at the ionosphere would not be sufficiently refracted to return to the Earth (as a sky wave).

82. The curvature of the Earth is the factor which limits the **maximum theoretical range** of any direct wave, see Figure 10-30.

83. Subject to the power transmitted, the higher the transmitter and/or the receiver the greater the direct wave range.

84. The maximum theoretical range of a direct wave signal is given by the formula:

$$\text{MAX RANGE (NMS)} = 1.25(\sqrt{H_1} + \sqrt{H_2})$$

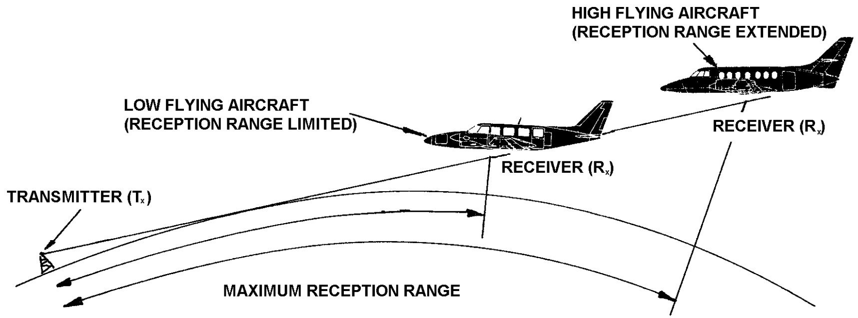
where

H1 is the height of the transmitter in feet, amsl

H2 is the height of the receiver in feet, amsl

85. Obviously the presence of **intervening high ground** will invalidate the above formula.

FIGURE 10-30
The Direct Wave



The Surface Wave

86. Fortunately, the maximum range from a transmitter at which radio signals can be received is not always limited to the direct wave range. At frequencies in the LF and MF bands diffraction has the effect of altering the direction of travel of the radio wave such that some will more or less follow the Earth's curvature. Thus in the LF and MF frequency bands, **surface wave** (**or ground wave**) propagation is possible, see Figure 10-31.

FIGURE 10-31
The Surface Wave

87. At the relatively low frequencies concerned, the amount of attenuation suffered by the radio wave as it travels across the surface is not too great, and the problem of surface attenuation can be overcome if transmitters of sufficient power are used. Also, at lower frequencies the amount of diffraction increases, so increasing the reception range at the surface.

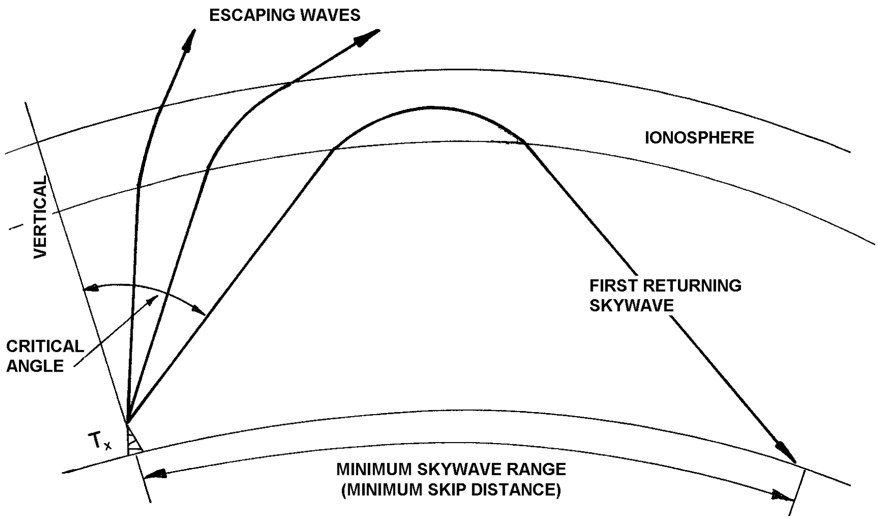
88. Remember that the rate of attenuation of a surface wave is approximately three times greater over the land than it is over the sea. Consequently maximum surface wave ranges of 1000 nms are achievable over the sea, but only 300 nms over the land.

The Sky Wave

89. So far we have considered the normal propagation paths for radio waves in the VHF band and above (**direct waves**) and the LF and MF bands (**surface waves**). The obvious gap in the middle is the HF band, and it is within this band that the refractive properties of the ionospheric layers may be usefully employed.

90. If a radio wave of a suitable frequency within the HF band is directed towards the ionosphere it will **refract** within the ionosphere. As the angle between the vertical and the outgoing radio wave increases, the radio wave will eventually bend sufficiently within the ionosphere to return to the Earth's surface. When this happens, the angle between the vertical and the radio wave is termed the **critical angle**, and the returning wave is termed the **first returning sky wave**, see Figure 10-32.

FIGURE 10-32
The Sky Wave

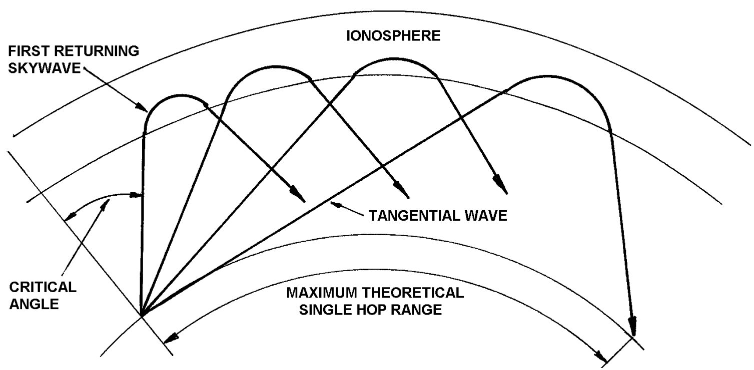


91. The critical angle will depend upon the frequency of the transmitted signal, as well as the state of ionisation of the ionosphere. As the frequency is **increased** the rate of refraction **decreases**, and therefore, the critical angle **increases**. This is because the radio wave must travel further within the ionospheric layer in order for there to be sufficient refraction to produce a returning sky wave.

92. If the signal enters the ionosphere at angles in excess of the critical angle, sky waves will continue to return to the Earth's surface, subject to transmitter power, until the angle coincides with the **Earth's tangential wave** shown at Figure 10-33.

93. Please note that at Figure 10-32 and Figure 10-33 the signal is shown **refracting within** the ionosphere, and **not bouncing off** the bottom of the ionosphere.

FIGURE 10-33
Earth Tangential Waves

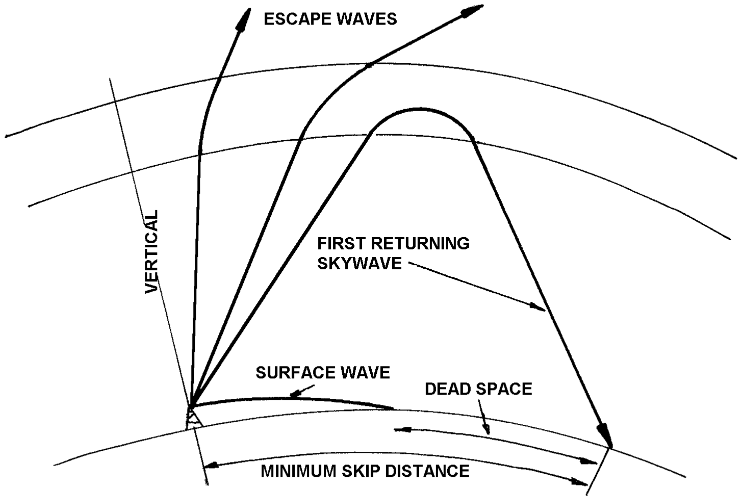


94. In all of these diagrams it is convenient to show only one ionospheric layer, however the **height** of the ionosphere is significant. For example, in the case of the tangential wave the theoretical skip distance is 1300 nm from an 'E' layer refraction and 2500 nm from the 'F' layer.

95. Because the signals used lie within the HF band, the rate of surface attenuation is fairly high and therefore surface wave ranges will rarely exceed 100 nm. It is quite possible that the first returning sky wave will not arrive at the Earth's surface below a range, which is well in excess of 100 nm. There will therefore be an area within which no signal will be received, and this is termed the **dead space**, see Figure 10-34. The distance between the transmitter and the point at which any sky wave returns to the surface of the Earth is termed the **skip distance**.

96. The distance between the transmitter and the point at which the **first returning sky wave** returns to the surface of the Earth is termed the **minimum skip distance**, and defines the far end of the dead space, as shown at Figure 10-34.

FIGURE 10-34
The Dead Space and Minimum Skip Distance



97. The preceding diagrams have shown only **single hop sky waves.** Providing that the signal has sufficient power there is no reason why it should not make the return journey to the Earth's surface via the ionosphere two or even three times. This is known as **multi-hop propagation.**

98. The final problem for your consideration in terms of sky wave propagation is the diurnally changing state of the ionosphere. Obviously there is no point in using a frequency such that the receiver lies within the **dead space.** Conversely the range between transmitter and receiver should ideally be only slightly in excess of the **minimum sky wave range**, so that the strongest possible signal is received.

99. In an ideal world the perfect situation would be for the distance between the transmitter and the receiver to be exactly the same as the skip distance for the first returning sky wave (the minimum skip distance). The frequency which would have to be transmitted in order to achieve this, for a given transmitter-receiver distance set of ionospheric conditions, is termed the **optimum frequency**. Since it is not possible to predict precisely the state of the ionosphere, the **maximum usable frequency** is used rather than the **optimum frequency**. The maximum usable frequency will be slightly lower than the optimum frequency and will therefore give a slightly shorter minimum skip distance. This ensures that slight variations in ionospheric intensity or height will not cause the receiver to lie within the dead space.

100. Figure 10-35 shows the ideal situation with the maximum usable frequency for a low and intensely ionised layer during the day. Figure 10-36 shows the situation with the same frequency at night, when the ionosphere is higher and is partially de-ionised. The receiver now lies within the dead space, and a new maximum usable frequency is required, as shown at Figure 10-37.

FIGURE 10-35
Day Time Sky
Wave Propogation

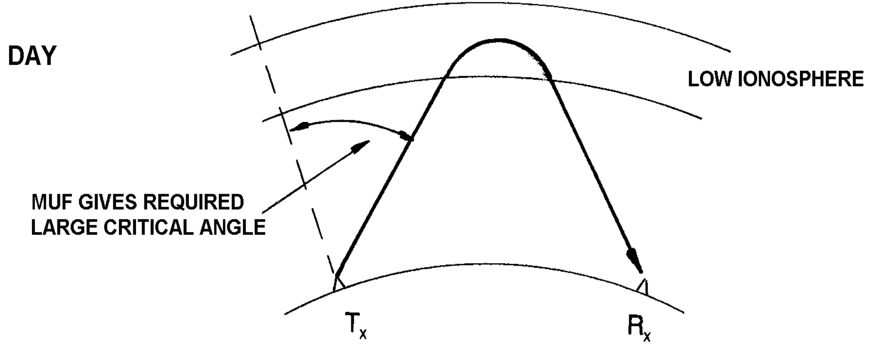


FIGURE 10-36

Night Time Sky
Wave Propogation
- same Frequency

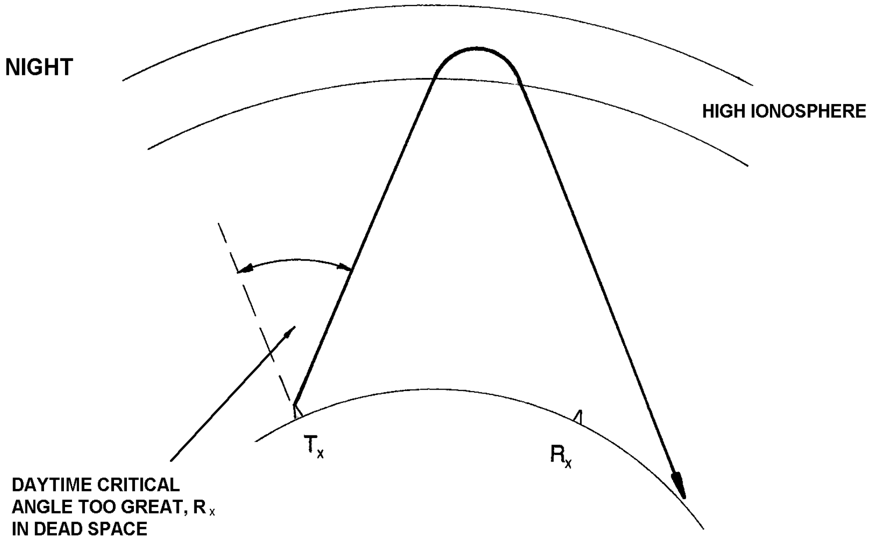
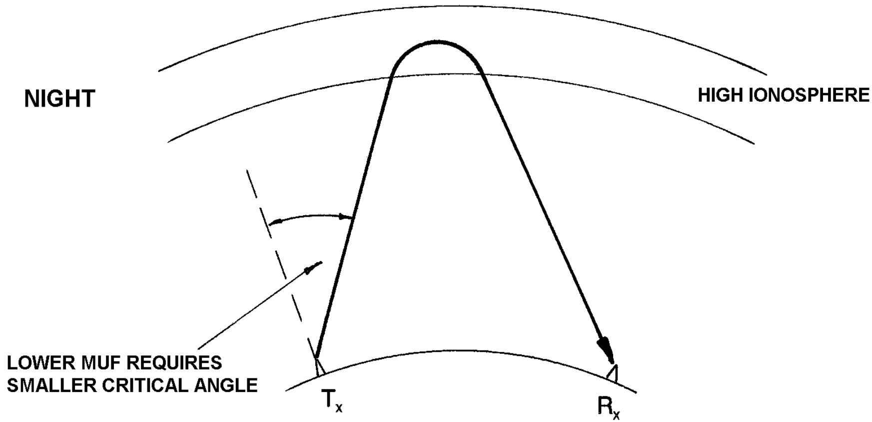


FIGURE 10-37
Night Time Sky Wave Propogation with New Max Usable Frequency



101. The maximum usable frequency required by night will be approximately **half the daytime maximum usable frequency.** This lower frequency will have a smaller critical angle, to overcome the geometry of the higher ionosphere. Additionally the lower frequency will refract at a similar rate, despite the partial de-ionisation of the layer. Finally and fortunately, the less intense level of ionisation will mean that the signal will not be unduly attenuated, despite the lower frequency, which is being used.

102. In the LF and MF bands sky waves do not generally exist by day, since the lower frequency signals are totally attenuated within the ionosphere. By night it is normal for some sky waves to survive, since the ionosphere is now somewhat **weaker.** This poses the problem of **sky wave interference** by night in such equipments as ADF.

Propagation at VLF

103. By now you should be absolutely familiar with the following general statements concerning propagation paths:

(a) **LF** and **MF** signals are propagated as **surface (ground)** waves. Within these bands sky waves are not normally present by day, and their presence by night reduces useful equipment range due to **night effect**.

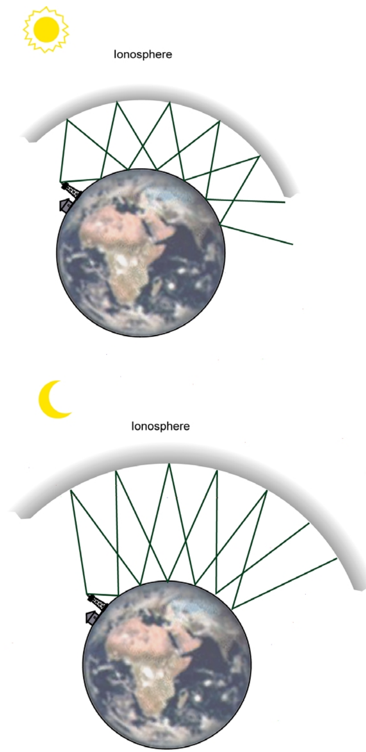
(b) **HF** signals are propagated principally as **sky waves**, which are present both by day and by night.

(c) At **VHF and above**, signal range is limited to **line of sight**, since the propagation path is **direct wave** only.

104. At VLF the wavelength is obviously much longer than at VHF or UHF and now a process which is similar to ducting (see below) occurs between the surface of the Earth and the underside of the lowest layer of the ionosphere, which may now be considered as a reflecting surface. The name which now seems to be accepted for this type of propagation path is the **conduit wave**. As with VHF ducting the VLF signals travel great distances with little attenuation and consequently ranges of more than 11,000 nm can be achieved with transmitter power outputs in the region of ten kilowatts. The principle of conduit wave propagation is shown at Figure 10-38.

FIGURE 10-38

Conduit Wave Propogation



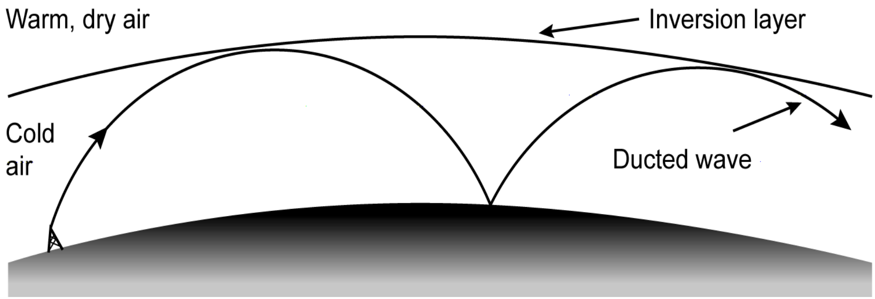
The Ducted Wave

105. The four propagation paths which have now been discussed, namely direct waves, surface waves, sky waves and conduit waves are all predictable and therefore useful. The fifth option, **ducting,** is unpredictable and ducted waves cannot therefore be usefully employed.

106. Under certain meteorological conditions, radio waves in the VHF, UHF and SHF bands, which normally travel only in straight lines, may behave in a way which is at first sight similar to sky waves.

107. The meteorological conditions required for duct propagation are **a marked temperature inversion and a rapid decrease in humidity with height**. Figure 10-39 shows ducting which, in this case, is occurring between the surface and a low-level inversion. The signal is effectively **trapped** under the inversion and may travel hundreds of miles with little attenuation. In this way, when high pressure systems prevail, interference may be heard from distant **VHF** communications stations which are far beyond the normal direct wave range.

FIGURE 10-39
The Ducted Wave



108. Ducting may also occur above the surface of the Earth, between two inversion layers.

109. Whilst ducting cannot be used to advantage, it can seriously affect receipt of signals and will cause unexpected station interference. For example, a VOR or VHF communication transmission may be trapped beneath an inversion layer so that it cannot be received by aircraft flying above the layer. Similarly, transmission which would normally be limited to short surface ranges because of their direct wave propagation may travel distances well in excess of 1000 nm.

Tropospheric Scatter

110. Small random irregularities or fluctuations in the refractive index of the atmosphere can cause a scatter of radio waves in the frequency band 200 MHz to 10 GHz. Provided that sufficient transmitter power is available, the signal will be scattered towards the receiver from a volume of atmosphere approximately 3 to 8 km above the Earth's surface. Tropospheric scatter offers extended over-the-horizon range for high power communication systems operating from the upper VHF band through to the SHF band. The maximum range achieved in this manner is considered to be in the region of 400 nm. This propagation phenomenon is akin to dust scattering a beam of sunlight in a dark room or hallway.

111. The advent of satellite communication systems has more or less halted research into the development of communications systems relying on tropospheric scatter for better than line-of-sight range at VHF and above, however the development of military **over the horizon radars** using tropospheric scatter continuous.

112. Having discussed radio frequency bands and propagation paths in this section, it is perhaps appropriate at this stage to produce a table of the radio and radar systems which you will subsequently study, showing the frequency range and frequency band in which each of these equipments operate, see Figure 10-40.

FIGURE 10-40
Individual Equipment Operating Frequencies and Frequency Bands

System	Frequency Range	Frequency Band
Decca	70 to 130 KHz	LF
Loran C	100 KHz	LF
ADF	190 to 1750 KHz	LF/MF

HF Communications	2 to 25 MHz	HF
ILS Markers	75 MHz	VHF
ILS Localiser	108.1 to 111.95 MHz	VHF
VOR	108.0 to 117.95 MHz	VHF
VHF Communications	118 to 136 MHz	VHF
ILS Glidepath	329.15 to 335.0 MHz	UHF
DME	960 to 1213 MHz	UHF
SSR	1030 and 1090 MHz	UHF
GPS	1575.42 MHz (L1) 1227.6 MHz (L2)	UHF UHF
Satcom (Inmarsat)	1500 to 1600 MHz (Aircraft to Satellite) 4000 to 6000 MHz (Satellite to Ground)	UHF SHF
Radio Altimeter	4200 to 4400 MHz	SHF
Weather Radar	9375 MHz	SHF
MLS	5031 to 5091 MHz	SHF
ATC Surveillance Radars	600 to 1300 MHz	UHF
ATC Ground Manoeuvre Radars	10 to 16 GHz	SHF

Modulation Techniques

113. The preceding section covered the various propagation options for transmitting electromagnetic radio waves from one point to another. It is now necessary to consider the various techniques which may be employed to superimpose **intelligence** on to the basic radio wave. When this is done the radio wave is termed the **carrier wave**, and the **intelligence** is said to be **modulated** on to it.

114. One way of modulating a carrier wave to convey intelligence is to vary the **amplitude** of the carrier wave in sympathy with the modulating wave form which is at a lower frequency (the intelligence). This technique is known as **amplitude modulation.**

115. Another way of modulating a carrier wave to carry intelligence is to vary the **frequency** of the carrier wave in sympathy with the modulating intelligence. This technique, logically, is known as **frequency modulation.**

116. The third modulation technique which is considered is **pulse modulation.**

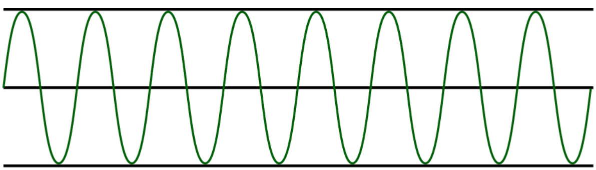
Amplitude Modulation

117. For convenience the various sub-divisions of Amplitude Modulation (AM) are coded, and the particular codes which are pertinent to this syllabus are discussed below.

Unmodulated Carrier Wave

118. As the name suggests, unmodulated (or continuous) carrier wave is simply a radio wave which is transmitted as a constant amplitude and a constant frequency, and therefore carries no intelligence. In other words the wave is **unmodulated.** This type of signal is ideal for direction finding equipment such as the ADF, although it is obviously necessary to superimpose a morse identifier onto the wave from time to time, so that the operator knows which transmitter he is tuned into. The designation normally given to this type of signal is **NON.** An unmodulated carrier wave is illustrated at Figure 10-41.

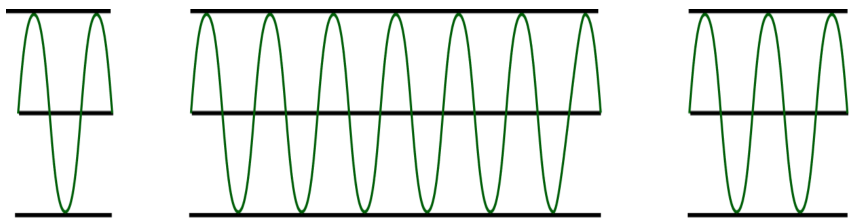
FIGURE 10-41
Unmodulated Carrier Wave



Keyed Carrier Wave

119. Keyed (or interrupted) carrier wave is the simplest form of modulation. Here the radio signal is not continuously transmitted but is switched on and off at desired intervals, see Figure 10-42. The primary use of keying is to convey intelligence using the morse code. The designation given to this type of signal is **A1A**. It is principally employed on long range NDBs, the advantage being that all of the power is contained in the carrier wave, and none in the modulating wave, since there isn't one. Range for a given transmitter power output is therefore enhanced, however the disadvantage is that a BFO (Beat Frequency Oscillator - see later) must be incorporated into the receiver to make the morse audible.

FIGURE 10-42
Keyed Carrier Wave



Simple Amplitude Modulation

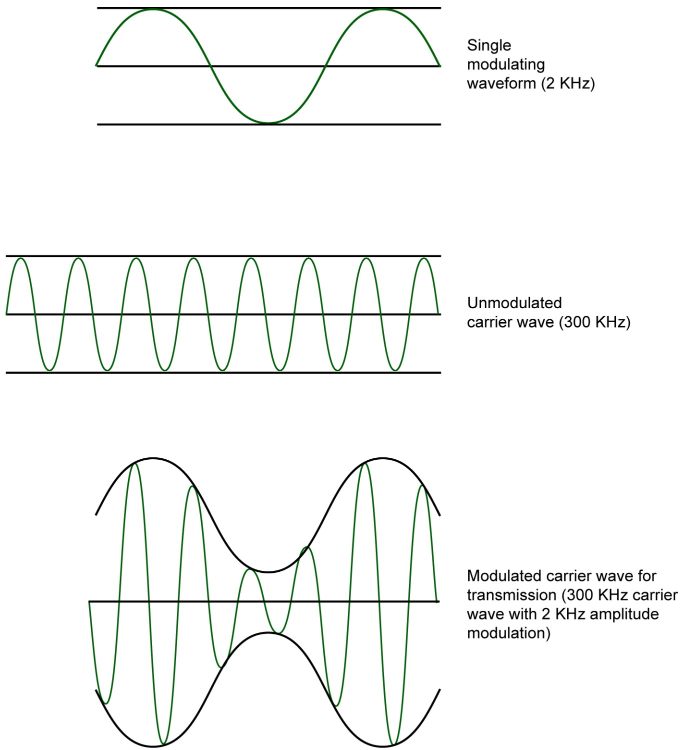
120. Now to, as it were, real amplitude modulation where the amplitude of the carrier wave consistently varies in sympathy with the intelligence wave form. The significance of **simple amplitude** modulation is that **the modulating wave form is constant in amplitude and constant in frequency**, see Figure 10-43. The intelligence normally used for this type of transmission would be a simple steady audio frequency tone.

121. By keying either the modulating signal, or the modulated carrier wave itself, the audio tone can be used to convey simple morse idents.

122. If you check any short range NDB in the COMS section of the Aeronautical Information Publication you'll find that it is given as **NONA2A**. The NON is the continuous carrier wave, which occurs between the ident sequences, and which gives the ADF receiver a nice steady signal for direction finding. The A2A which is tacked onto the end is the NDB identifier, which in this case is achieved by keyed amplitude modulation, using a steady audio tone.

123. The designation given this type of transmission is either **A2A** with an NDB (where it is keyed to achieve station identification) or **A8W** in ILS (where the depth of modulation is made to vary across the transmitted lobes).

FIGURE 10-43
Simple Amplitude Modulation



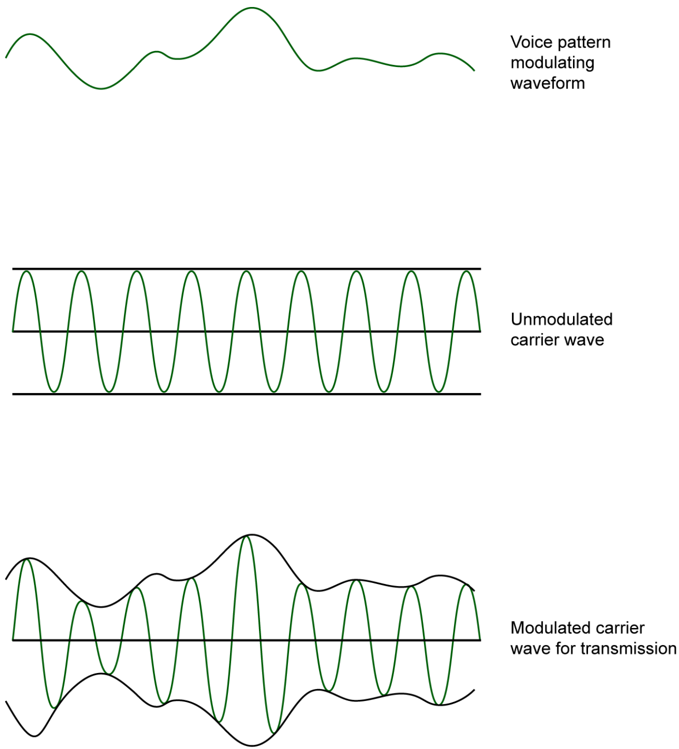
Complex Amplitude Modulation

124. The modulating waveform in the previous paragraph was simple in nature, that is to say constant in amplitude and in frequency. The human voice produces a complex waveform which is often modulated on to a carrier wave as intelligence.

125. The human voice produces a complex modulating waveform in that it is varying in both amplitude (a shout as against a whisper) and in frequency (a groan as against a scream). You may well be either groaning or screaming right now, if you have a microphone and an oscilloscope handy you can prove the complexity of the wave form! Voice modulation is shown schematically at Figure 10-44.

126. This type of signal is designated **A3E** when used in VHF communications, and **J3E** when used in HF communications (normally on single sided band networks).

FIGURE 10-44
Complex Amplitude Modulation



Depth of Modulation

127. Before proceeding on to frequency and pulse modulation techniques it is necessary to consider briefly **depth of modulation** as it applies to both simple and complex amplitude modulated signals. Quite simply, the depth of modulation is the ratio of the amplitude of the modulating waveform to the amplitude of the carrier wave prior to modulation, expressed as a percentage, or:

$$\text{Depth of Modulation \%} = \frac{\text{Amplitude of the Modulating Waveform} \times 100}{\text{Amplitude of the Carrier Wave (before Modulation)}}$$

An alternative formula for determining the depth of modulation is:

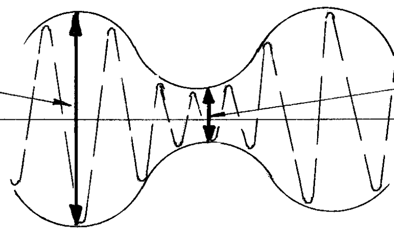
$$\text{Depth of Modulation \%} = \frac{\text{Maximum Amplitude} - \text{Minimum Amplitude} \times 100}{\text{Maximum Amplitude} + \text{Minimum Amplitude}}$$

128. Figure 10-45 shows a simple amplitude modulated carrier wave. The carrier wave (as always with amplitude modulation) is varying in amplitude in sympathy with the modulating wave. In this case maximum amplitude of the modulated carrier wave is 10 volts and the minimum amplitude of the modulated carrier wave is 3 volts. Using the second formula:

$$\text{Depth of Modulation \%} = \frac{10-3}{10+3} \times 100 = \frac{7}{13} \times 100 = 54\ \%$$

129. Since it is the power contained in the carrier wave at its **lowest amplitude** which governs the range of the signal, it is normal to reduce the percentage depth of modulation when extreme range reception is required.

FIGURE 10-45
Depth of Modulation

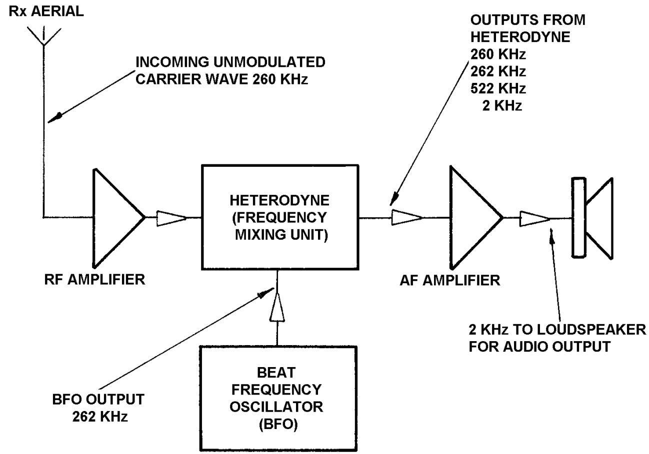


The Beat Frequency Oscillator (BFO)

130. An amplitude modulated signal is **demodulated** in a conventional receiver without any difficulty since the amplitude of the carrier wave is varying in sympathy with the intelligence waveform. With a **NON** or **A1A** (see earlier text) signal the amplitude of the carrier wave remains constant and therefore it is impossible to achieve an audible output from a receiver using conventional demodulation techniques.

131. Figure 10-46 shows how the receiver is modified when the BFO function is selected. The BFO is made to generate an alternating current, the frequency of which differs from the incoming carrier wave frequency by, typically, 2 KHz. The incoming signal and the BFO-generated signal are both fed to the heterodyne unit which **mixes** the two to give four output frequencies. The output of the heterodyne unit comprises the two input frequencies, the sum of the two input frequencies, and the **difference** frequency. It is only the difference frequency (2 KHz) which is audible, and this is fed to the loudspeaker, producing the audio tone. The difference frequency is known as the **Beat Frequency**.

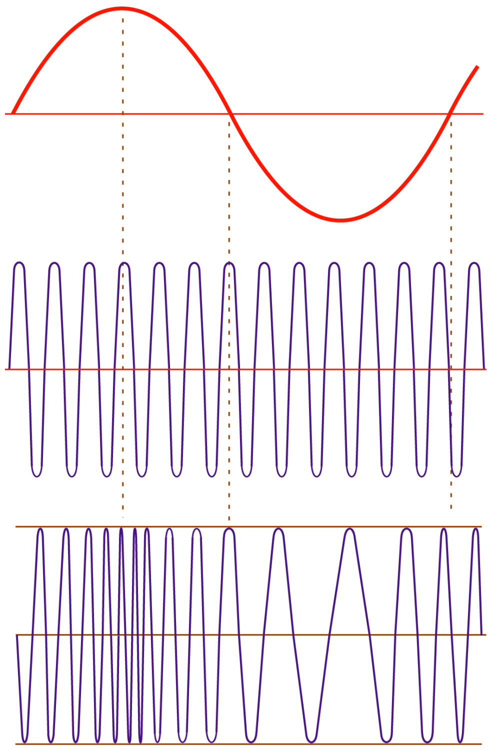
FIGURE 10-46
The Beat Frequency Oscillator



Frequency Modulation

132. When pure frequency modulation (FM) techniques are employed the **amplitude** of the carrier wave normally remains constant, however the **frequency** of the carrier wave is made to vary in sympathy with the modulating wave form (the intelligence), see Figure 10-47. The **amplitude** of the modulating waveform is represented by the **amount** by which the frequency of the carrier wave changes and the **frequency** of the modulating waveform by the **rate of change** of the carrier wave frequency.

FIGURE 10-47
Frequency Modulation



133. With VOR the carrier wave is both amplitude and frequency modulated. The designation given to this type of signal (as it applies to VOR) is **A9W.**

AM Versus FM

134. Comparing the technique of frequency modulation with amplitude modulation, **frequency modulation (FM) transmitters are simpler than AM transmitters,** the necessary modulating power is relatively lower and the reception is practically static-free. This last benefit is due to the fact that the VHF band is practically free from static, and where it is present, it is normally an amplitude-oriented disturbance. Of the disadvantages, **FM receivers are more complex** and the modulated transmission calls for a much wider frequency band to cover its multi-sidebands (see later). This is why FM broadcasters operate in the VHF band; the congestion in lower frequency bands would not permit accommodation of the necessary bandwidth. Being in the VHF band, as a side benefit they can cover a complete range of human audio frequencies (up to 15 KHz) and thus provide high fidelity reception whereas in the MF band they would have to be content with staying inside the limit of a spread of 10 KHz.

VHF Communications

135. VHF Communication Systems operate in the frequency range 118-136.975 MHz and use the frequency modulation technique for superimposing the intelligence onto the Carrier Wave. A breakdown of the VHF Communication Band is shown in Figure 10-48:

FIGURE 10-48
VHF Communication Frequency Band

Block Allotment Of Frequencies (MHz)	World-Wide Utilisation	Remarks
118 to 121.4 Inclusive.	International and National Aeronautical Mobile Services	Specific international allotments will be determined in the light of regional agreement.
121.5	Emergency Frequency.	In order to provide a guard band for the protection of the aeronautical emergency frequency, the nearest assignable frequencies on either side of 121.5 MHz are 121.4 MHz and 121.6 MHz, except that by regional agreement it may be decided that the nearest assignable frequencies are 121.3 MHz and 121.7 MHz.
121.6 to 121.975 Inclusive.	International and National aerodrome surface communications.	Reserved for ground movement, pre-flight checking, air traffic services clearances, and associated operations.
122 to 123.05 Inclusive.	National Aeronautical Mobile Services.	Reserved for national allotments
123.1	Auxiliary Frequency SAR.	
123.15 to 123.675	National Aeronautical Mobile Services.	Reserved for national allotments

Block Allotment Of Frequencies (MHz)	World-Wide Utilisation	Remarks
123.7 to 129.675 Inclusive.	International and National Aeronautical Mobile Services.	Specific international allotments will be determined in the light of regional agreement.
129.7 to 130.875 Inclusive.	National Aeronautical Mobile Services.	Reserved for national allotments.
130.9 to 136.875 Inclusive.	International and National Aeronautical Mobile Services.	Specific international allotments will be determined in the light of regional agreement.
136.9 to 136.975 Inclusive.	International and National Aeronautical Mobile Services.	Reserved for VHF Air-Ground Data Link Communications.

136. The ICAO Special Communications/Operations Divisional meeting (Montreal/April 1995) decided that, in order to increase channel capacity, the VHF communications band should be split from 25 to 8.33 KHz channel spacing. It was noted that this measure was a short term improvement for regions experiencing severe VHF frequency spectrum congestion, in anticipation of the development and implementation of the future digital VHF radio system.

European Implementation

137. As detailed above, the mandatory carriage of 8.33 KHz capable radios was required for operation above FL 245 in the ICAO EUR Region with effect from 1 January 1999. However, as a result of current fitment rates, Eurocontrol took the decision in July 1998 to delay operational implementation of 8.33 KHz channel spacing until **7 October 1999**.

138. The States that will initially implement 8.33 KHz channel spacing are Austria, Belgium, France, Germany, Luxembourg, Netherlands, Switzerland and the United Kingdom (known as 8.33 States).

139. Although the initial 8.33 States do not comprise all the States in the ICAO EUR Region, the mandatory carriage above FL 245 applies to the whole of the ICAO EUR Region.

140. Parallel operation of 25 and 8.33 KHz spaced VHF channels for the same airspace sector will not be achievable. Accordingly, from 7 October 1999, the flight plans of all aircraft not equipped with radios compatible with the new channel spacing, requiring the provision of an Air Traffic Service as GAT in the airspace designated for 8.33 KHz channel spacing, will be rejected.

141. Aircraft VHF radio equipment used in the ICAO EUR Region will still require to be capable of tuning to 25 KHz spaced channels and, additionally, to receive in an environment which uses off-set carrier systems (CLIMAX operation). Airspace users should take into account that these offset carrier systems may continue to be used throughout Europe for many years.

UK Implementation

142. The UK, although an 8.33 kHz participating state, will not implement 8.33 kHz channel spacing until **June 2000** due to related technical dependencies. Therefore, the UK will file a difference with ICAO to the effect that the UK has no requirement for the carriage of 8.33 kHz radios prior to June 2000. However, all non 8.33 kHz equipped aircraft intending to fly through UK airspace above FL 245 are to comply with the flight planning requirements detailed in the relevant AIC in order to prevent rejection of flight plans by IFPS.

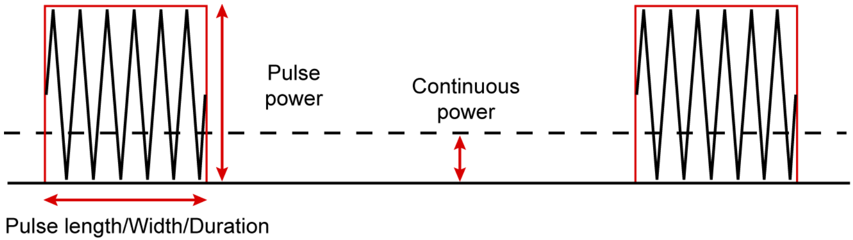
French Implementation

143. Notification had been given that France had intended to introduce 8.33 kHz channel spacing above FL 195. However, a decision has now been taken by France that initial implementation will be above FL 245 with effect from 7 October 1999.

Pulse Modulation

144. Pulse modulation is similar in principle to keyed carrier wave (A1A). The difference is that the pulses used in typical radar systems are of very short duration, typically one microsecond, whilst the period between pulses is **relatively** very long, typically one millisecond. The designation given to pulse modulation is **PON**.

FIGURE 10-49
Radar Terminology



145. The terms in Figure 10-49 should be understood by the student. The continuous power of a radar can be considered to be the average power taking into account the time interval between the pulses, the pulse width and the pulse power (i.e. it allows for the 'duty cycle' of the transmitter).

Pulse Coded Modulation

146. In some equipments (e.g. SSR) the time interval between the individual pulses of a transmitted pulse sequence is made to vary in accordance with a pre-determined code. This allows additional information to be passed between a Tx and Rx, or Rx and Tx, and is called Pulse Coded Modulation.

Multiplex Modulation

147. When two types of modulation are used together, typically AM and FM, the carrier wave is able to carry two separate sets of information. This technique is called Multiplex Modulation.

Modulation Designators

148. We can now summarise the modulation designators as they apply to some of the equipments which we will study in the Radio Navigation section:

A3E	Amplitude modulated double side-band radiotelephony (VHF communications)
J3E	Amplitude modulated single side-band radiotelephony (HF communications).
NONA1A	Unmodulated carrier wave (NON) periodically interrupted by keyed carrier wave (A1A) to give the Morse identifier (long range NDBs).
NONA2A	Unmodulated carrier wave (NON) with a simple keyed amplitude modulation periodically superimposed to give the Morse identifier (short wave NDBs)
PON	Pulse (radars)
A8W	Simple amplitude modulation, however the depth of the modulation is made to vary (ILS).
A9W	Composite amplitude/frequency modulation (VOR).
NOX.G1D	Microwave Landing System

Bandwidth

149. We tend to assume that if the published frequency of a transmitter is, for example 123.2 MHz, that the frequency transmitted is 123.2 MHz and only 123.2 MHz. This is not in fact the case.

150. With a little thought it should be obvious that a frequency modulated signal must necessarily cover a band of frequencies, this band being termed the **bandwidth** of the transmitter. For perfect reproduction of the **intelligence** the radio receiver must have the same bandwidth. That is to say that the receiver must possess the ability to accept the same band of frequencies as that transmitted. The upper and lower limits of the frequency band are equally spaced above and below the published spot frequency (except in single side band systems).

151. With FM signals the bandwidth is quite **broad**, and consequently the number of channels which can be fitted into any given part of the radio frequency spectrum, without risk of overlap and consequent interference, is somewhat limited.

152. With amplitude modulation it would appear at first sight that only one frequency is transmitted. Unfortunately life is never that simple. The example of simple amplitude modulation illustrated at Figure 10-43 shows a 300 KHz carrier wave modulated with a 2 KHz audible tone. The range of frequencies actually transmitted in this case would be 298 KHz to 302 KHz, giving a bandwidth of 4 KHz.

153. The **bandwidth** of any **amplitude modulated** signal is twice the value of the highest frequency of the modulating waveform. The range of frequencies actually transmitted is the basic carrier wave frequency **plus** and **minus** the highest frequency of the modulating waveform.

154. **The bandwidth of an amplitude modulated signal will be narrower than the bandwidth of a frequency modulated signal conveying the same intelligence.** Despite this narrow bandwidth the situation still arises whereby certain parts of the radio frequency spectrum, particularly the HF band, have become congested, and that as a result of this, interference from stations on adjacent channels often occurs. The greater the bandwidth transmitted, and therefore required of the receiver, the greater the scope for interference. Speech transmission using amplitude modulation requires a 3 KHz modulating waveform, whilst music requires about 15 KHz. In order to alleviate this problem single side band (SSB) systems are now widely used.

Single Side Band Transmission

155. With amplitude modulated signals we have established that the frequencies actually transmitted are equi-spaced about the carrier wave frequency. The **intelligence** is contained in two bands of frequencies, one above the carrier wave spot frequency, the **upper side band** (**USB**), and one below the carrier wave spot frequency, the **lower side band** (**LSB**). **The same information is conveyed in the USB as in the LSB**, and it is therefore possible to suppress one or the other without losing any of the intelligence. The effect of this is to halve the bandwidth, enabling closer channel spacing without risk of interference. Furthermore, because the side bands absorb at least 25% of the transmitter power, suppressing one of the side bands leaves more of the total transmitter power available for transmission of the carrier wave – significantly increasing the range at which the transmitted signal can be received.

156. In summary, the advantage of SSB transmission are that:

(a) It occupies less of the available radio spectrum (the bandwidth is halved).

(b) Interference from other transmissions is less likely (a better signal to noise ratio).

(c) Greater range is achieved for a given power (the same power is concentrated into the narrower bandwidth).

Interference

157. Radio interference may arise from several sources. Such interference is generally termed **noise**. It should be noted that interference generated in the atmosphere is correctly known as **static**, whereas interference generated by man-made equipment is correctly termed **noise**. **Station interference** was briefly discussed in the preceding paragraphs. This can occur when stations are operating on adjacent channels and bandwidth overlap exists. Equally station interference can occur when two stations operate on the same frequency, and the geographic spacing between them is insufficient (see ADF and VOR).

Static

158. The vast amounts of energy contained within thunderstorm cells cause the storm clouds to emit high levels of electromagnetic energy. These emissions are particularly troublesome in the VLF, LF, MF and to a degree the HF bands.

159. In the HF band, static may additionally result from ionospheric disturbances such as are caused by sun spot activity. Generally, below VHF, the lower the frequency the greater the problem. For a given location, ionospheric static will be more troublesome in summer than in winter and will be more troublesome at low latitudes than at high latitudes. Ionospheric disturbances will pose more of a problem at night than during the day, since at night sky waves spend more time within the ionised layers.

160. Precipitation static occurs, for example, when rain at a given electrical potential strikes an aircraft at a different electrical potential. On impact a current will flow between the airframe and the water droplet; dust and sand in the atmosphere can produce the same effect. Whenever a current flows it produces a magnetic field which is termed **precipitation static**. Precipitation static, including that caused by snow, is particularly troublesome within the LF and MF bands.

Noise

161. **Electrical noise** is primarily caused by **sparking**, which readily occurs at generator and electric motor commutators, at relays, and at poor electrical connections. As we probably all know from experience, this noise can seriously interfere with radio reception.

162. **Electronic noise** is troublesome at VHF frequencies and above. When alternating current electron flow occurs at these very high frequencies within the equipment circuitry, the wiring itself tends to **emit** electro-magnetic energy. In order to prevent these emissions from causing interference it is necessary to **screen** sensitive areas of the equipment.

Q Code and Radio Bearings

163. The Q code was introduced as a shorthand to assist wireless telegraphy (Morse) operators. With the advent of voice communications networks (telephony), wireless operators disappeared from flight decks and with them much of the Q code. Some Q notations still survive, and four of them which are particularly pertinent to this part of the syllabus are listed below.

(a) **QDM**. The **magnetic great circle** bearing of the **station from the aircraft**. Sometimes defined as being the great circle heading to fly to the station in still-air conditions.

(b) **QDR.** The **magnetic great circle** bearing of the **aircraft from the station.** The term **radial** is often used as an alternative to QDR.

(c) **QTE.** The **true great circle bearing** of the **aircraft from the station.**

(d) **QUJ.** The **true great circle bearing** of the **station from the aircraft.**

Aerials

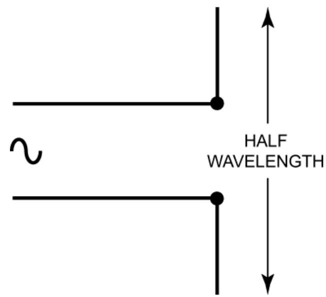
164. Alternating current flowing in a conductor is associated with a combination of electric and magnetic fields. The fields vary in time with the frequency of the alternating current and electromagnetic waves radiate away carrying energy supplied by the transmitter. A transmitting aerial is simply a conductor supplied with alternating current.

165. If the radiated electromagnetic wave is intercepted by a passive conductor, an alternating current is generated in the conductor. This conductor is acting as a receiving aerial. The receiver which it feeds is designed to detect the current, amplify it and convert it into a form appropriate to the desired display.

Half Wave Dipole

166. The half wave dipole is probably the most common practical aerial. It is a straight conducting rod, approximately half a wavelength long. Alternating current is fed to the aerial along a transmission line, which is usually connected at the centre of the rod.

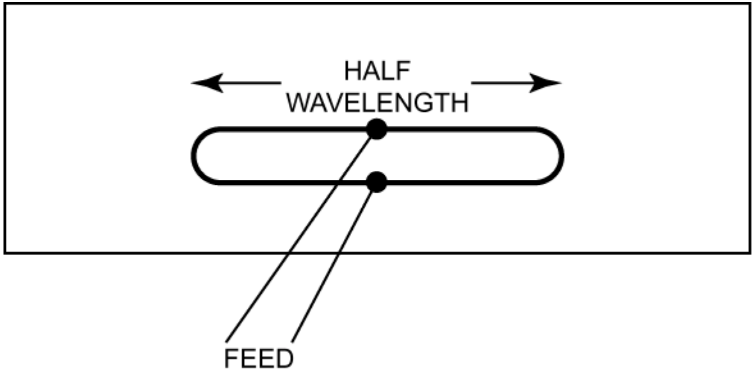
FIGURE 10-50
Half Wave Dipole Aerial



Slot Aerials

167. A half-wave slot cut in a sheet of metal behaves very like a half-wave dipole. The slot aerial in Figure 10-51 has a radiation pattern exactly like that of a vertical dipole. Slot aerials are polarised at right angles to the slots, whereas rod aerials are polarised parallel to the rods.

FIGURE 10-51
Half Wave Slot Aerial

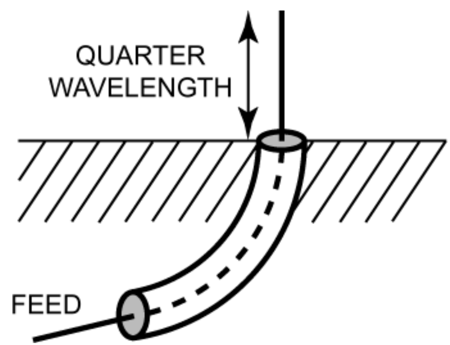


168. Some types of aircraft suppressed aerial are based on the slot. In these aerials, the slot is cut in the aircraft skin. Half-wave slots cut in the wall of a waveguide, feature in the design of certain microwave aerials.

Quarter Wave Unipole

169. The physical length of a half-wave dipole depends on design frequency and, at low frequencies, it may be excessive. The quarter-wave unipole consists of a single conductor a quarter of a wavelength long mounted at right angles to a conducting surface. This conducting surface acts as a reflector and creates the image of the unipole leading to a performance very like that of the half-wave dipole. Obviously radiation is only present in the space above the conducting surface but in this space the radiation pattern is identical to that of the dipole. The radiation resistance of the unipole is about half that of the dipole.

FIGURE 10-52
Quarter Wave Unipole Aerial



Parasites

170. The half-wave dipole is not particularly directional. A simple modification, which improves the directional qualities of a dipole, involves the use of parasitic radiating elements. Parasites are conducting rods placed near to the energised (driven) dipole. The signal transmitted from the dipole induces currents in the parasites, which reradiate the energy received. By adjusting the lengths and spacings of the parasites relative to the dipole, constructive interference can be encouraged in one direction and destructive interference in others. A parasite, longer than a half wavelength placed near to a dipole enhances the radiation in the direction of the dipole. Such a parasite is called a reflector. A short parasite tends to enhance the radiation in the direction of the parasite and such a parasite is therefore called a director.

The Yagi Array

FIGURE 10-53
The Yagi Array

171. The Yagi aerial array is formed from a driven dipole, a reflector and a number of directors. The polar diagram is similar in all planes and beamwidths of a few tens of degrees are available.

The Folded Dipole

172. An effect of the parasite elements is to reduce the radiation resistance of the driven dipole. This can cause serious mismatch problems and an increase in system losses. For these reasons, the folded dipole, shown in Figure 10-54, is often used as the driver in a Yagi array. It has a higher radiation resistance which, when reduced by the effect of the parasites, offers a reasonable match to the feeder.

FIGURE 10-54
The Folded Dipole

Radar Aerials

173. The basic function of a radar aerial is to act as a coupling device or transducer between free-space propagation, and guided-wave (transmission line) propagation. When the radar transmits, the function of the antenna is to concentrate the radiated energy into a beam of the desired shape, which points in the desired direction in space. On receive, the aerial collects the energy contained in the signal and delivers it to receiver.

174. Fundamentally radar aerials do not differ at all from aerials used for communications at other frequency bands. The most important practical difference between radar aerials and most communications aerials, is the requirement to be able to direct a narrow beam of energy in any direction at will. The narrow beam property is often achieved by focussing the transmitted energy by means of a specially designed reflector, in much the same way as a searchlight beam is formed, or by utilising an array configuration in which it is arranged that the radiation from a number of individual elements adds up in one direction only. Scanning of the beam may be achieved by mechanical or electronic means.

175. The shape of the beam is a function of the aerial shape. The aerial may be designed to produce a narrow-beam pattern with a uniform beamwidth in all planes; this is usually called a pencil beam. Other applications may require a beam which is narrow in azimuth, but much wider in elevation, or vice-versa. Such a beam is called a fan beam, and is achieved by using an aerial with an aperture which is wide in the dimension requiring the narrow beam, and narrow in the other direction.

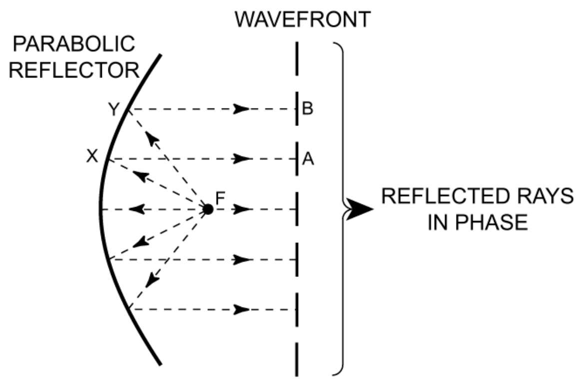
Sidelobes

176. An unavoidable feature of radar aerials is the existence in the radiation pattern of small subsidiary beams, known as sidelobes, which are generated in directions other than that of the main beam. The existence of sidelobes is usually undesirable since they represent wasted power and can give rise to spurious echoes.

Parabolic Antennas

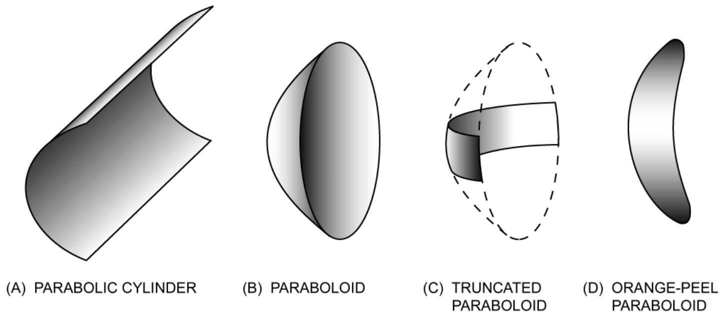
177. The paraboloid or parabolic dish is widely used as a reflector because of two basic geometrical properties. These are illustrated in Figure 10-55 for a simple parabola. Firstly, all the rays from a fixed point (called the focal point, F) to the parabola, are reflected as parallel rays. Secondly, in Figure 10-55, the path lengths FXA, FYB, etc. are all equal. Thus the reflected wave is made up of parallel rays which are all in phase. Notice that this does not imply that the beam will not diverge. In fact, except in the region very near to the antenna, the beam **will** diverge. The significant effect of a parabolic reflector is that it converts a point source of energy at the focus into a plane wavefront of uniform phase.

FIGURE 10-55
Properties of the Parabola



178. The beam shape may be modified by cutting away parts of the paraboloid and making an 'orange-peel' (Figure 10-56(d)) or a 'truncated' 'paraboloid' reflector (Figure 10-56 (c)). Notice that the beam is narrower in the plane in which the reflector aperture is greater. Another common configuration is the parabolic cylinder shown in Figure 10-56 (a), and the parabolic 'cheese' reflector which is a narrow parabolic cylinder enclosed on either side by flat plates.

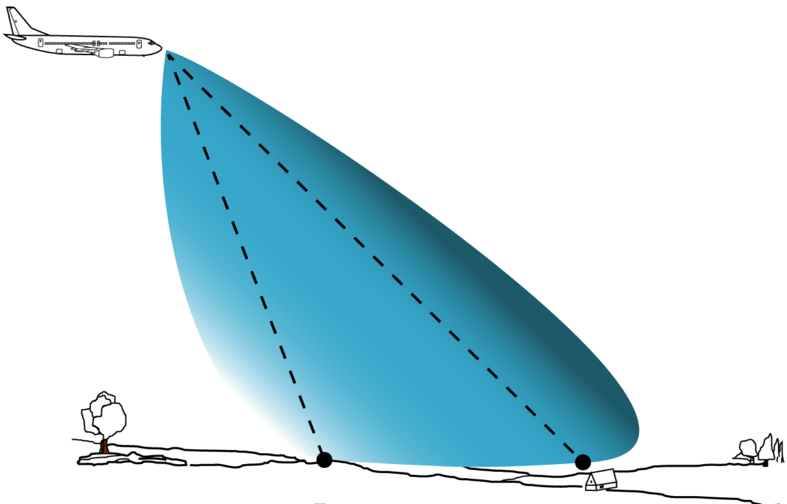
FIGURE 10-56
Types of Parabolic Reflector



Cosecant Squared Aerial

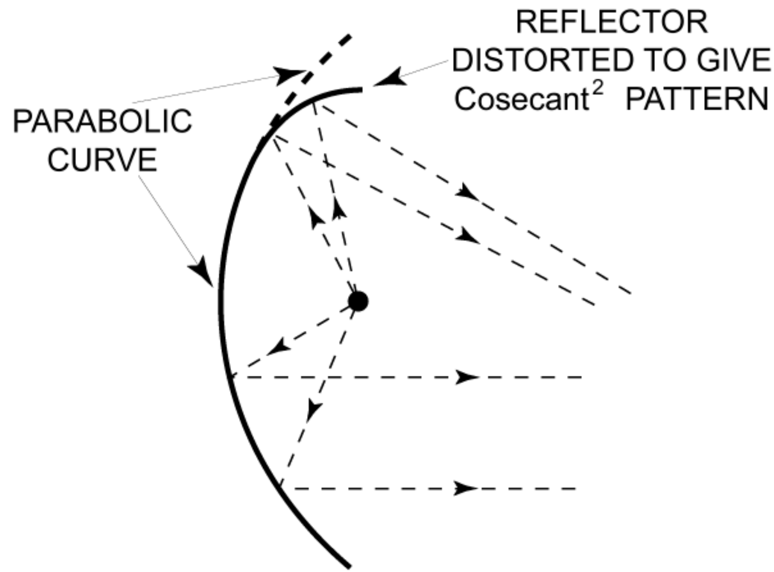
179. The shape of the vertical radiation pattern required from a radar aerial, depends on the function of the radar. Special beam shapes can be produced by appropriately configured aerials. For example the cosecant squared reflector, widely used in airborne mapping radars, produces a beam which is narrow in azimuth but wide in elevation, in order to provide equal strength returns from similar ground targets at different ranges (Figure 10-57).

FIGURE 10-57
Cosecant Squared
Radiation Pattern



180. This specialised type of pattern is often produced by distorting the upper portion of a parabolic reflector, so that more energy is directed at the longer range targets than towards those at short range (Figure 10-58).

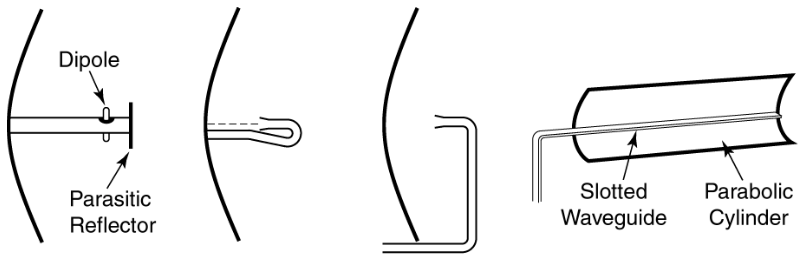
FIGURE 10-58
Cosecant Squared Reflector



Feeds for Parabolic Reflectors

181. Parabolic Reflectors are usually energised by an elementary aerial placed at the focal point, or in the case of a parabolic cylinder, a line of such aerials placed along the axis. The feed element often consists of a dipole or a waveguide horn, or for the parabolic cylinder a length of slotted waveguide. Typical arrangements are shown at Figure 10-59.

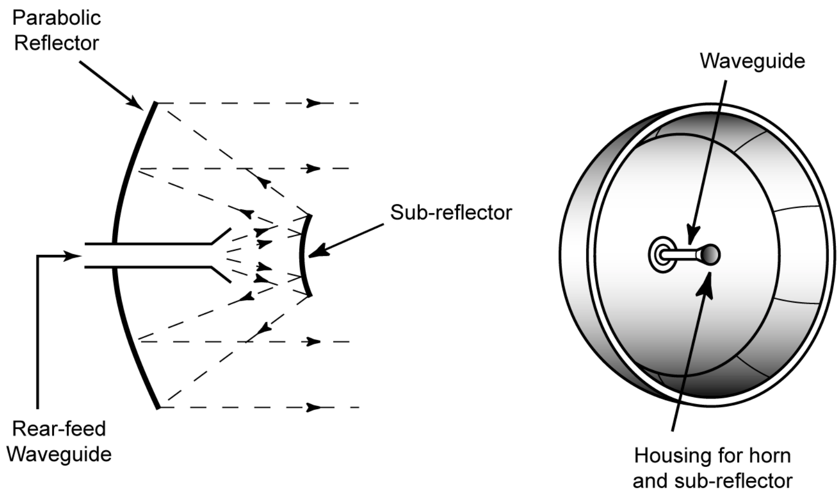
FIGURE 10-59
Feeds for Parabolic Reflectors



The Cassegrain Aerial

182. The aerial shown at Figure 10-60 is developed from a principle used to reduce the length of optical telescopes, and is known as the Cassegrain Aerial.

FIGURE 10-60
The Cassegrain Aerial

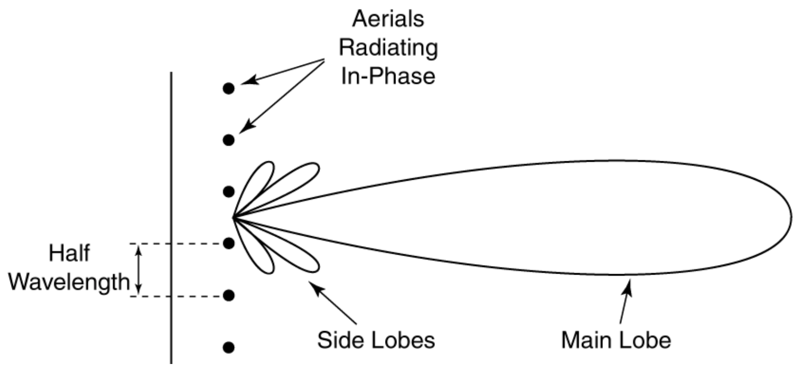


183. A parabolic reflector is fed from the rear by a waveguide horn. The waveguide energy is reflected from a small convex sub-reflector into the main parabolic reflector. This ‘double-focussing’ results in a beam considerable narrower than would be achieved from a conventional aerial of similar dimensions. As well as the advantages of smaller size and simpler mounting, the rear feed means that a shorter waveguide run between aerial and radar is required, and waveguide losses are thus reduced. The Cassegrain configuration is often used in low-noise applications where a low-noise amplifier may conveniently be mounted directly behind the main reflector, making the waveguide run as short as possible.

Slotted Waveguide Arrays

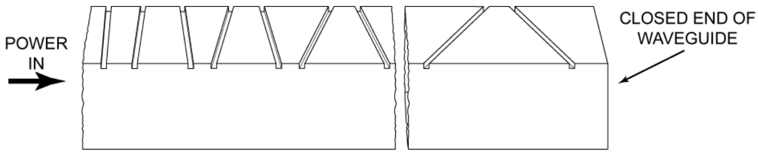
184. It has already been shown that when a number of aerials are spaced half a wavelength apart and fed in phase, the energy radiated from the aerials adds in some directions and cancels in others depending on the phase relationships involved. This type of aerial array is the linear broadside array, which gives a beam at right angles to the line of the constituent aerials (Figure 10-61).

FIGURE 10-61
The Broadside Array



185. A similar effect may be achieved by means of a series of half wavelength slots cut in the wall of a length of waveguide, and so arranged that each slot radiates equal amounts of in phase energy. Such a device constitutes a microwave broadside array. Typically the slots would be cut in the narrow dimension of the waveguide as shown at Figure 10-62.

FIGURE 10-62
Inclined Slot Waveguide Array



Instrument Landing System Aerial Radiation Patterns

186. The Instrument Landing System (ILS) is a runway approach aid which provides the pilot with accurate guidance both in azimuth and elevation during an approach in bad weather.

ILS Ground Equipment

187. The ground installation consists of:

(a) A localiser transmitter which defines the extended centreline of the instrument runway, and indicates any deviation from this centreline.

(b) A glidepath transmitter which defines a safe descent slope (normally three degrees), and again indicates any deviation from this safe approach.

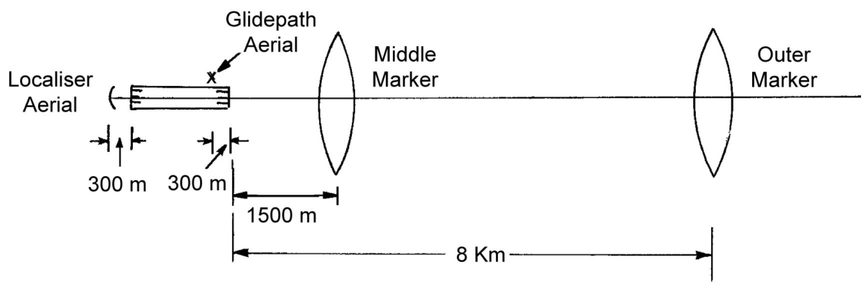
(c) Normally two (occasionally three) marker beacon transmitters for a typical installation. That is to say that with many installations the inner marker is omitted, leaving only the middle and outer markers.

The primary purpose of the markers is to define specified ranges from the runway threshold.

Localiser Transmitter

188. A localiser antenna array is approximately 25 metres wide and four metres high, and is normally situated some 300 metres beyond the upwind end of the instrument runway, see Figure 10-63.

FIGURE 10-63
ILS Aerial Locations

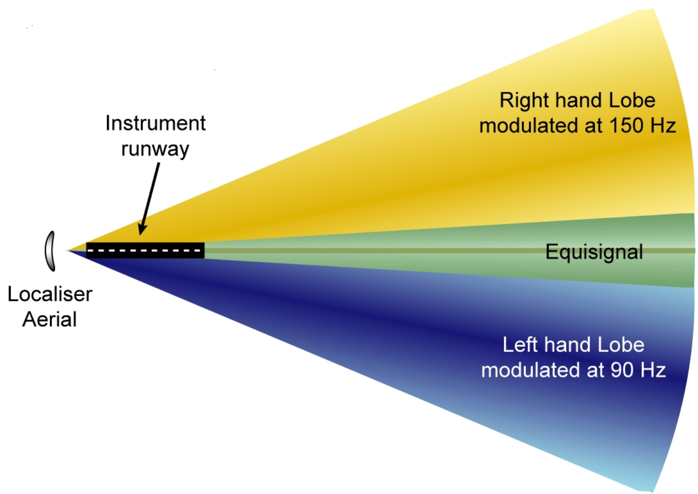


189. Should it not be possible to locate the localiser aerial on the extended centreline, it may be located to one side of the runway, giving what is known as an **offset** ILS. In this case, the QDM of the localiser centreline will differ from the runway centreline QDM by a few degrees.

Localiser Radiation Pattern

190. The localiser transmits two overlapping lobes of electro-magnetic energy (designated A8W) on the same VHF carrier wave frequency. The centre of the overlap area, **the equisignal**, defines the ILS QDM, see Figure 10-64.

FIGURE 10-64
ILS Localiser Radiation Pattern



191. The lobe on the pilot's left during the approach is amplitude modulated at 90 Hz, whilst the right lobe is amplitude modulated at 150 Hz. The depth of modulation of both the lobes is made to vary, being greatest at the centre and least at the sides of the lobes. The airborne localiser receiver compares the depth of modulation of the 150 Hz and 90 Hz waves. When they are of equal depth the localiser needle will be centralised.

192. When the depth of modulation is uneven the localiser needle is deflected in the appropriate direction. The greater the difference in modulation depths, the greater the displacement of the localiser needle from the centre of the instrument.

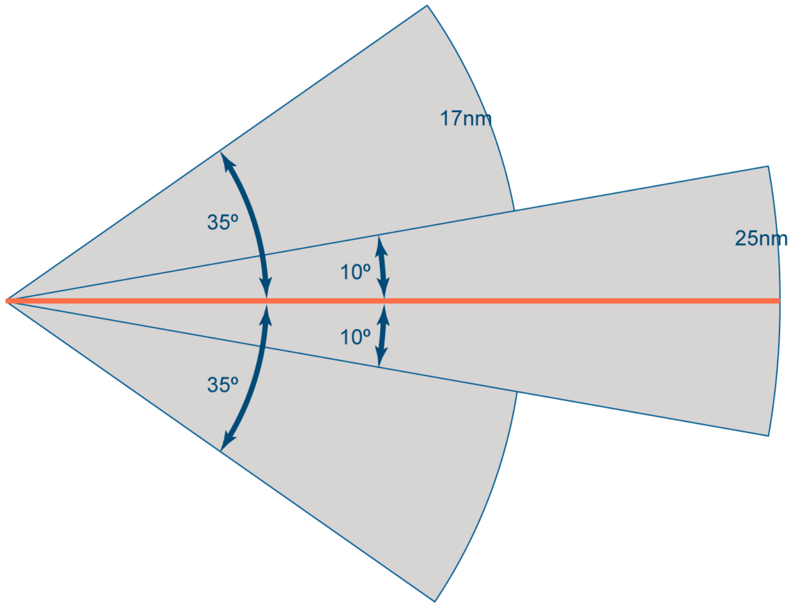
193. In the United Kingdom ILS localisers which are associated with normal glidepath transmitters provide coverage from the centre of the localiser antenna to distances of:

(a) 25 nm within plus or minus 10° of the equisignal (centre) line.

(b) 17 nm between 10° and 35° from the equisignal (centre) line.

As illustrated at Figure 10-65.

FIGURE 10-65
ILS Localiser Coverage



194. In the United Kingdom ILS localisers which are associated with steep angle glidepath transmitters provide coverage from the centre of the localiser antenna to distances of:

(a) 18 nm within plus or minus 10° of the equisignal (centre) line.

(b) 10 nm between 10° and 35° from the equisignal (centre) line.

195. As far as the above coverage areas are concerned, a **normal** glidepath transmitter should be considered to be one which produces a glidepath angle of approximately three degrees above the horizontal, and a **steep** glidepath should be considered to be one which defines an angle from the horizontal of 4° or more.

196. Pilots are warned that use of the localiser outside these areas, even on the approach side, can lead to **False Course** and **Reverse Sense** indications being received. Such use should not be attempted. In particular it must be noted that there is no provision for localiser **Back Beams** to be used in the United Kingdom, and any indications from them must be ignored.

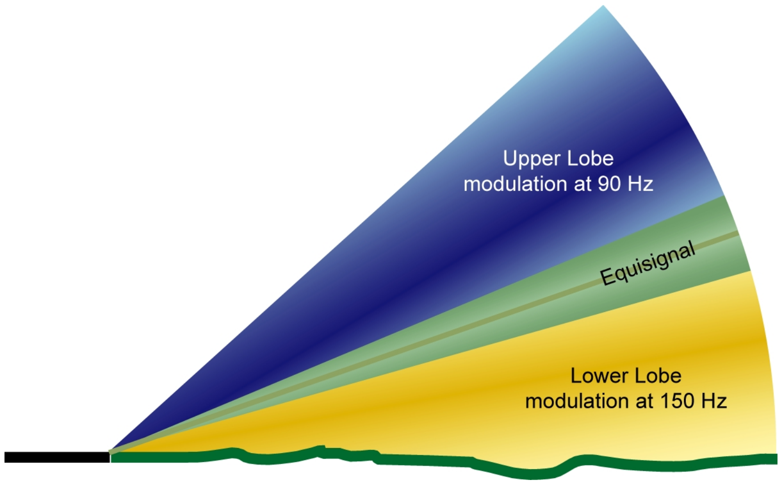
Glidepath Transmitter

197. There are two glidepath aerials which are both mounted on a mast approximately ten metres tall, which is displaced some 150 metres from the runway centreline and 300 metres upwind of the threshold markings.

Glidepath Radiation Pattern

198. As with the localiser, the glidepath transmitter emits two overlapping lobes of electro-magnetic energy (designated A8W) on the same carrier wave frequency. The frequency range used to glidepath transmissions lies in the UHF band, and in this case the lobes overlap in the vertical plane. Again the lobes are continuously amplitude modulated at 90 Hz and 150 Hz. Figure 10-66 shows the idealised radiation pattern with the equisignal defining the glidepath at a typical value of 3° above the horizontal plane passing through the touchdown zone.

FIGURE 10-66
ILS Glidepath
Radiation Pattern



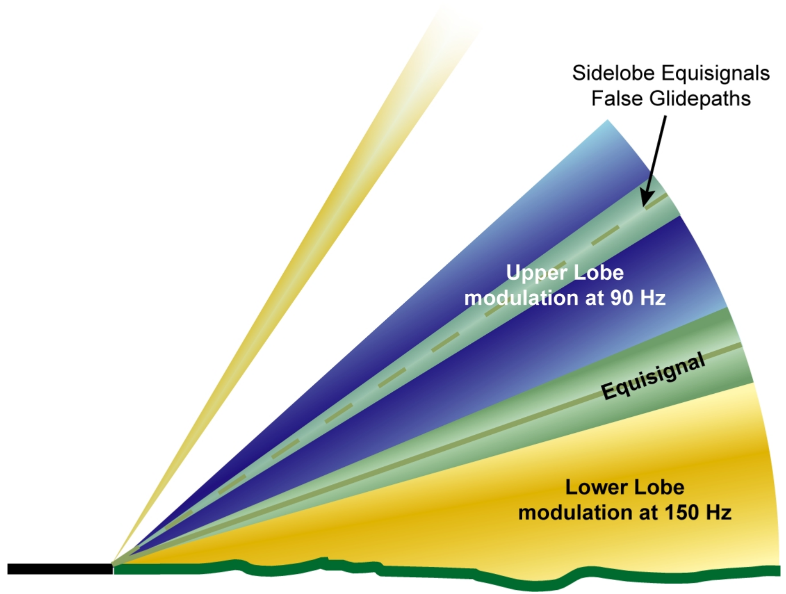
199. Since the lower (150 Hz) lobe lies adjacent to the surface ground-reflected waves result, giving side lobes (Figure 10-67). These side lobes may produce additional equisignals and consequently false glidepaths. Fortunately, these false glidepaths will be situated above the main glidepath and cannot therefore result in an aircraft flying dangerously low during the approach should the false glidepath be inadvertently followed. Indications that the aircraft is flying a false glidepath are listed below:

(a) During a normal ILS procedure, the aircraft captures the glidepath from below. This being the case, the true glidepath (being the lowest) will be the first one to be intercepted. The Civil Aviation Authority has issued a warning to pilots emphasising that special care must be taken at certain airfields around the world where procedures are published involving capture of the glidepath from above.

(b) The first (lowest) false glidepath will give a descent slope which is inclined at least 6° to the horizontal for a normal 3° glidepath, or 5° to the horizontal for a 2.5° glidepath. This will result in a rate of descent of at least twice the expected value.

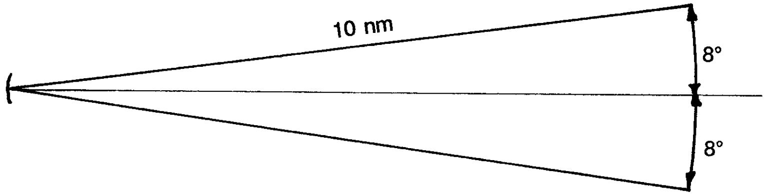
(c) The approach plate used by the pilot during an ILS approach shows check heights and altitudes at the marker beacons, and locator beacons (see later) if appropriate. If a false glidepath has been captured, a check of the altimeter will verify this. A typical check height over the outer marker would be 1500 feet (QFE), whereas on the first false glidepath the altimeter would read 3000 feet (QFE) or above.

FIGURE 10-67
ILS False Glidepaths



200. Glidepath coverage in azimuth (for United Kingdom installations) is provided through an arc of 8° on either side of the localiser centreline out to a range of 10 nm from the threshold, as illustrated at Figure 10-68.

FIGURE 10-68
ILS Glidepath Coverage - Horizontal Plane

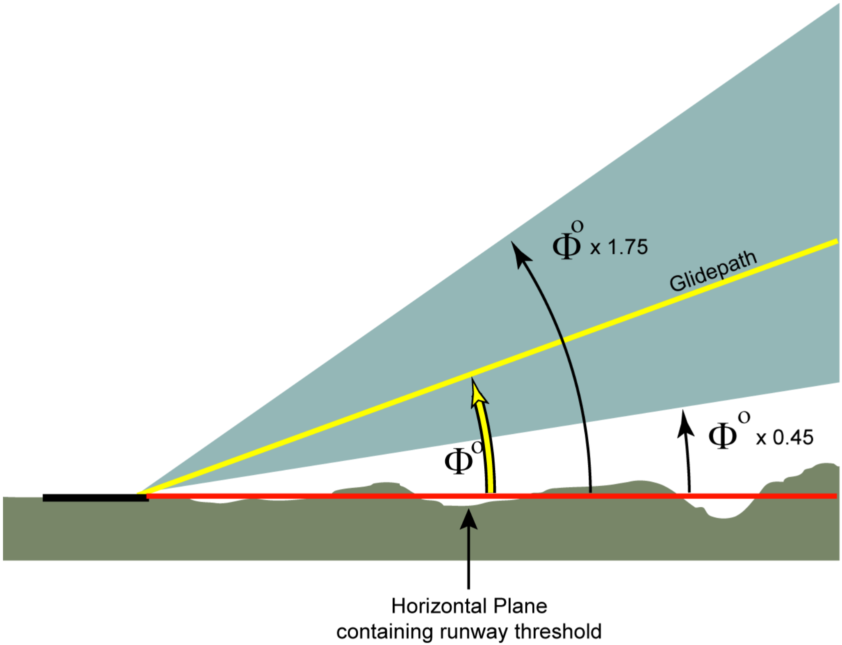


201. For glidepath transmitters which produce a steep glidepath, the coverage is reduced to a range of 8 nm from the threshold, again through an arc of 8° on either side of the localiser centreline.

202. Glidepath coverage in elevation is provided through an arc of 1.35° above the horizontal to 5.25° above the horizontal. These figures apply to a standard 3° glidepath installation, and are based on the formulae which state that glidepath coverage (in elevation) is provided through an arc (measured from the horizontal) of between **glidepath angle x 0.45** and **glidepath angle x 1.75**, as illustrated at Figure 10-69.

FIGURE 10-69

ILS Glidepath Coverage - Vertical Plane



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203. Pilots are warned that use of the glidepath outside these limits can lead to intermittent and incorrect indications being received. In particular, use of the glidepath at very shallow approach angles, (that is below 1500 feet aal at 10 nm range), should only be attempted when the promulgated glidepath intercept procedure **requires** such use.

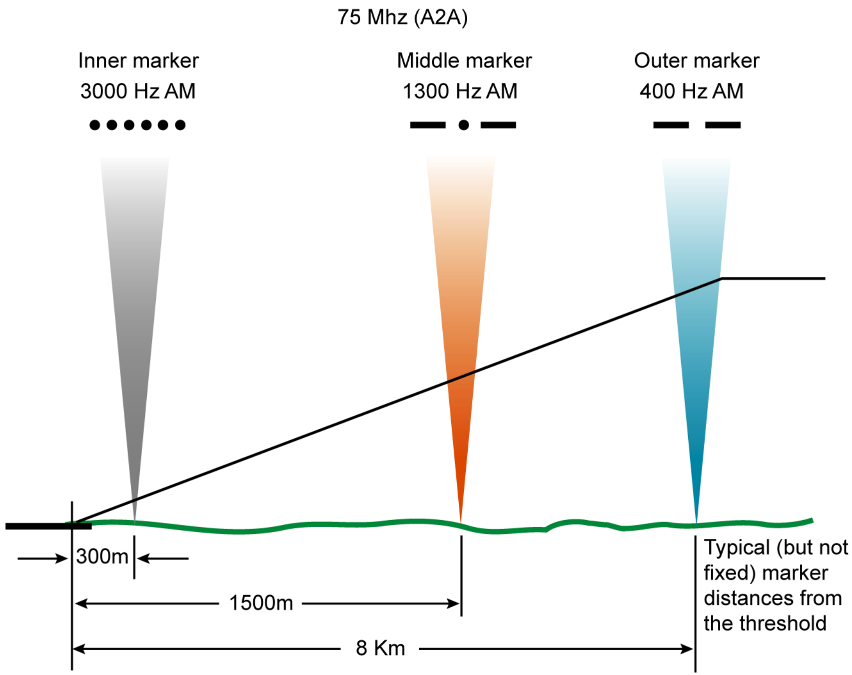
204. The glidepath indication **must** be ignored if the approach angle is so shallow as to put the aircraft at a height of 1000 feet or below at a range from touchdown of 10 nm or more.

205. Certain glidepaths in the United Kingdom do not exhibit correct deflection sensitivity to one side of the localiser course line. This effect is caused by terrain or other problems and can lead to inadequate **fly up** indications being received. When this situation exists a warning will be promulgated by NOTAM and subsequently appear in the appropriate columns of the COM2 section of the UK AIP.

Marker Beacons

206. Marker beacons radiate **fan-shaped** patterns of energy vertically upwards. Figure 10-70 shows an installation using three marker beacons, although the inner marker is not often used these days. All market beacons transmit on a set frequency of 75 MHz. Notice from Figure 10-63 and Figure 10-70 that there is no interference between adjacent beacons because of the narrow extent of the radiation patterns **along** the glidepath.

FIGURE 10-70
ILS Marker Beacon Radiation Patterns



207. The marker beacon transmissions are amplitude modulated with dots and/or dashes at given tones. As the aircraft flies through the radiation pattern associated with a given marker beacon, the pilot will receive both aural and visual indications as described at Figure 10-71.

FIGURE 10-71
Marker Beacon - Aural and Visual Indications

Outer Marker	Aural:	Low Pitch (400 Hz) Dashes
	Visual:	A Blue light flashing in synchronisation with the audible dashes at the rate of two per second.
Middle Marker	Aural:	Medium pitch (1300 Hz) alternate dots and dashes.
	Visual:	An amber light flashing in synchronisation with the audible dots and dashes at the rate of three characters per second.
Inner Marker	Aural:	High Pitch (3000 Hz) dots.
	Visual:	A White light flashing in synchronisation with the audible dots at the rate of six per second.

Airways Fan Markers or Z Markers

208. Marker beacons are still sometimes found straddling airway centrelines to denote reporting points. As with the ILS marker beacons, the airways fan markers radiate a fan-shaped pattern on a fixed carrier wave frequency of 75 MHz, however the power transmitted by an airways marker is considerably greater to facilitate high altitude reception. The aural identifier is a single Morse letter of high pitch tone (3000 Hz) which activates the white (inner marker) light on the aircraft marker beacon panel.

VOR Aerial Radiation Patterns

209. VHF Omni-directional Radio Range (VOR) is a system which gives accurate bearings with reference to ground-based stations using the principle of Phase Comparison between two waveforms.

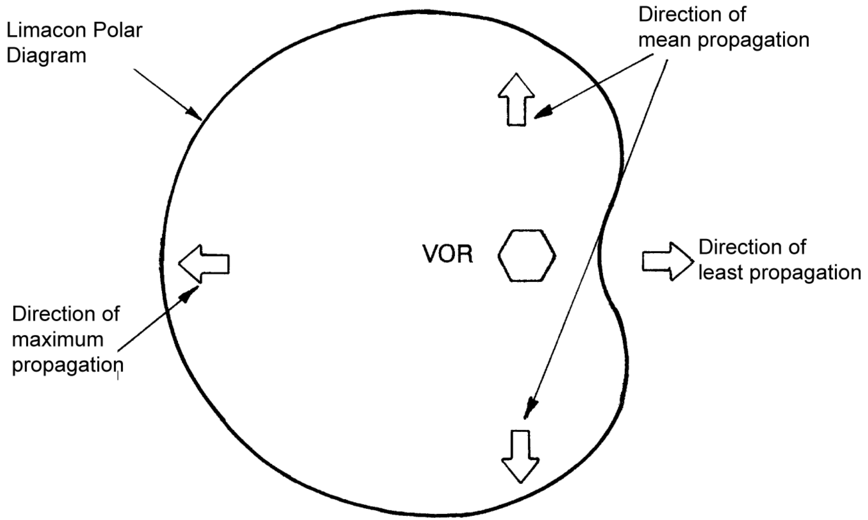
Principle of Operation

210. VOR stations transmit a carrier wave which is modulated in a manner previously described as **A9W** in the section entitled Modulation Techniques. This is to say that the single carrier wave is both **frequency** and **amplitude** modulated at the same time.

211. By **frequency** modulating the carrier wave with a simple 30 Hz waveform, the **reference** signal is achieved. This signal is so named since all airborne receivers at a given range from the station will receive a reference signal which is at the same phase, regardless of the aircraft bearing from the station.

212. The VOR station transmits in all directions (omnidirectionally), however the signal strength varies depending on the bearing from the station at a given point in time. The polar diagram for the VOR transmitter, which is known as a **limacon**, is illustrated at Figure 10-72.

FIGURE 10-72
Limacon Polar Diagram



213. The limacon itself is rotated at the rate of 30 revolutions per second and this has the effect of **amplitude** modulating the carrier wave arriving at an airborne receiver. The phase of the amplitude modulated signal will depend upon the bearing of the airborne receiver from the ground station. The amplitude modulated is therefore known as the **variphase** signal.

Doppler VOR

214. Conventional VOR transmitter aerials should be sited on flat terrain to minimise site errors. If such a site is not available, a complex aerial system may be employed to transmit the VOR signal. This type of station is know as a **Doppler VOR** (DVOR) beacon and produces a signal which is reasonably free of site errors even when the transmitter is sited in hilly terrain.

215. The way in which the bearing signal is produced is quite different from conventional VOR, the received signals are indistinguishable from each other and the airborne receiver will operate on either with equal facility. In Doppler VOR the reference signal is **amplitude** modulated at 30 Hz, whilst the bearing signal is **frequency** modulated at 30 Hz. Because this is the reverse of a conventional VOR, the bearing (or variable) modulation is made to **lead** the reference signal by a phase angle equal to the aircraft's magnetic bearing **from** the VOR ground station.

216. The Doppler VOR transmitter comprises a circle of about 50 antennae surrounding a single omni-directional antenna. The latter transmits the AM reference signal, whilst the circle of antennae are sequentially energised in an anti-clockwise direction at 30 revolutions per second (30 Hz). From any given direction, it will appear as though the transmitter is advancing and retreating at 30 Hz – in other words there will be a Doppler shift. The phase relationship between the doppler shift and the steady reference signal is arranged to be zero when received on a bearing of 000° (M) from the transmitter. Since both signals have the same modulating frequency (30 Hz), at 180° (M) from the VOR the phase difference will be 180°, at 270° (M) it will be 270° and so on.

Piston Engine Principles and Construction

Piston Engine Principles and Construction

Introduction

1. A single cylinder piston engine consists of a cylinder, closed at one end and in which a piston is free to slide up and down. The movement of the piston is transmitted to the crankshaft by means of a connecting rod, hinged to the piston at one end by the gudgeon pin and to the crank at the other end by a crank pin. The crank arm is fixed to the rotating crankshaft. The function of the connecting rod and crank is to convert the reciprocating motion of the piston into rotary movement of the crankshaft.

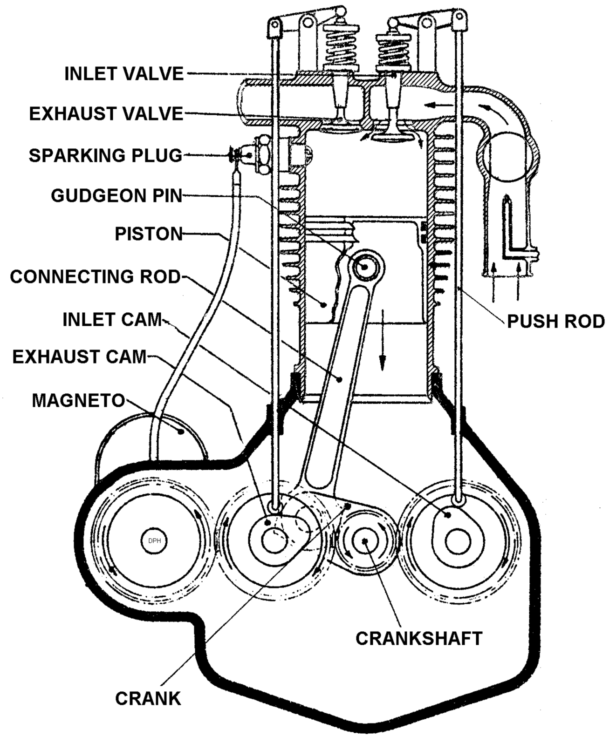
2. The closed end of the cylinder, known as the cylinder head, includes an inlet valve and an exhaust valve, each of which is held closed by strong springs. The inlet valve admits a combustible mixture of air and fuel to the cylinder whilst the exhaust valve allows the waste products of combustion to escape to atmosphere. Push rods, driven by cams attached to rotating gear wheels, open the valves through the lever action of rocker arms. The cam drive gears are meshed to a spur gear on the crankshaft.

3. The force to move the piston is derived from the expansion of hot gas, heated by the combustion of fuel. The fuel is ignited, in the case of a petrol engine, by an electrical discharge creating a large spark across the electrodes of a sparking plug. Aircraft piston engines usually employ a magneto, driven from the engine crankshaft, to provide the electrical energy to the sparking plug, which is located in the upper part of the cylinder.

4. These features are illustrated in the cross-sectional diagram of a single cylinder, spark ignition piston engine illustrated at Figure 11-1. In this diagram, for clarity, the inlet and exhaust valves are shown as being driven from separate cam drives. In practice the two cams would be mounted on a common camshaft.

FIGURE 11-1

Cross-Section through Single Cylinder Piston Engine



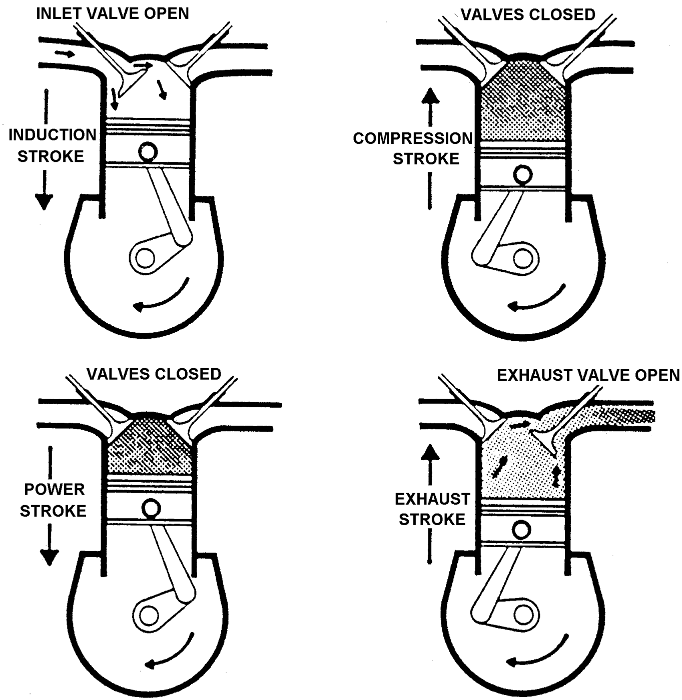
Working Cycles

The Four-Stroke Cycle

5. In aircraft piston engines the sequence of operations by which the heat energy is converted into mechanical energy, is known as the four-stroke cycle. A fuel-air mixture is inducted into the cylinder and compressed to raise the temperature to a point where the fuel will readily ignite. At close to maximum compression an electrical spark ignites the mixture and the heat of combustion causes a rapid increase in pressure, driving the piston downwards. The waste products of combustion are then ejected during the final stage of the cycle.

6. The cycle can be divided into four separate, but mechanically related operations each requiring one full stroke of the piston. They are induction, compression, power (the result of ignition and combustion) and exhaust. The basic principle of operation of the four-stroke engine is described in the following paragraphs. Figure 11-2 summarises the four-stroke cycle.

FIGURE 11-2
Four-Stroke Cycle



The Induction Stroke. During the induction stroke the inlet valve is open, and the piston is travelling down the cylinder from the upper extreme of travel, known as the top dead centre (TDC) position, to the lower extreme of travel known as the bottom dead centre (BDC) position. The increasing volume above the piston causes a pressure decrease such that cylinder pressure is lower than ambient atmospheric pressure. Air therefore enters the cylinder through the inlet pipe, or manifold, where fuel is added from the carburettor or fuel injection system.

7. The mass of mixture that enters the cylinder during the induction stroke depends upon a number of factors. Whilst the inlet valve is closed the fuel-air mixture in the inlet manifold is effectively at rest and will not instantly begin to flow into the cylinder at the start of the induction stroke. This problem is accentuated if the inlet manifold has rough sides or sharp corners that impede the flow of the mixture. Since the power achieved by the engine is dependent on the mass of mixture being burnt, it is normal to polish the inlet manifold to minimise the resistance to the flow of mixture. Additionally the mixture can be blown into the cylinder by means of superchargers or turbochargers, as discussed in a later section.

The Compression Stroke. Having completed the downward induction stroke, the piston now starts moving upwards from the BDC position, driven through the connecting rod by the inertia of the crankshaft. The inlet valve is closed by the action of the cam-driven push rod and the mixture is compressed in the enclosed space of the cylinder. By compressing the mixture into a smaller space, its pressure and temperature is increased. As the mixture is compressed it is heated adiabatically, and also gains heat from the hot surroundings. The pressure therefore rises to a higher value than that which would be expected from volumetric reduction alone. The purpose of the compression stroke is to raise the temperature of the fuel-air mixture to a value at which the fuel will readily ignite and burn efficiently.

The Power Stroke. Just before the piston reaches TDC on the compression stroke, a spark at the spark plug ignites the mixture. As the flame spreads through the mixture, the intense heat rapidly raises the pressure to a peak value, ideally when the piston is just past TDC. The mixture burns, and pressure falls as the hot gases expand, forcing the piston down the cylinder to complete the power stroke. During this stroke the piston is driving the crankshaft through the connecting rod and crank mechanism.

The Exhaust Stroke. Having descended to BDC on the power stroke the piston now starts up the cylinder once more. The exhaust valve is opened by the action of its cam and push rod mechanism and the combustion gases are forced out to atmosphere through the exhaust manifold. Ideally the exhaust manifold should be polished and free from sharp corners which would otherwise impede the exit of the burnt gases. This is important, since any exhaust gases remaining in the cylinder at the end of the exhaust stroke would impede the entry of fuel-air mixture at the start of the subsequent induction stroke. Expulsion of the combustion gases is known as scavenging.

Valve and Ignition Timing

8. The inference in the preceding description of the four-stroke cycle is that the inlet valve opens at TDC at the start of the induction stroke, and closes at BDC at the end of the induction stroke. Similarly it appears that the exhaust valve opens at BDC and closes at TDC at each end of the exhaust stroke. In fact it is necessary to adjust the valve sequence to account for a number of factors.

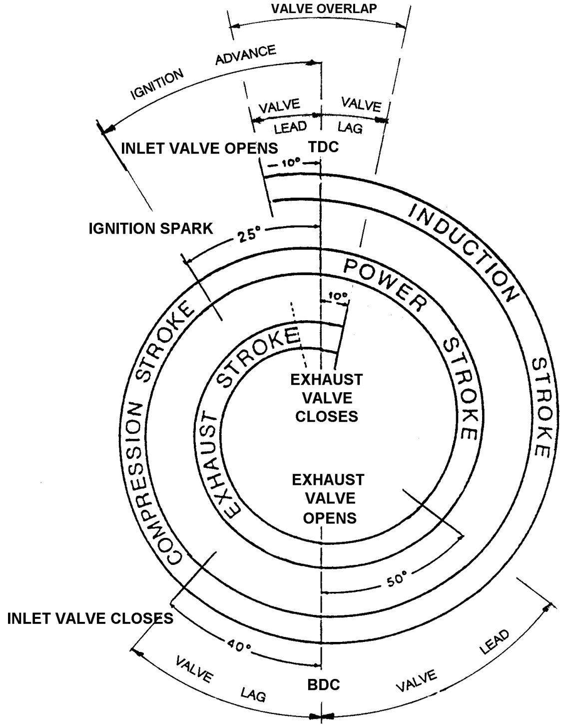
9. During the time that the valves are opening and closing the flow rate of gases past the valves is less than maximum and so the valves must be arranged to open before the piston commences its stroke and close after it has completed its stroke.

10. The mass of fuel-air mixture that enters the cylinder during the induction stroke is determined by the instant of closure of the inlet valve. If this were to close as the piston reached BDC on the induction stroke the cylinder pressure would be less than atmospheric pressure, because of the inertia of the gas. In order to allow a little more mixture into the cylinder, and thus increase the mass of the induced fuel-air charge, the inlet valve is kept open during the first part of the compression stroke. In this way the cylinder and atmospheric pressures have almost equalised when the inlet valve closes.

11. If the exhaust valve were to close as the piston reached TDC on the exhaust stroke there would inevitably be some waste gas remaining in the cylinder, again due to gas inertia. By keeping the valve open a little beyond TDC, during the start of the induction stroke, residual gas pressure ensures that the last of the waste gas is exhausted.

12. Typical timing of the opening and closing of the inlet and exhaust valves during the four-stroke cycle is illustrated in the timing diagram at Figure 11-3.

FIGURE 11-3
Valve Timing Cycle



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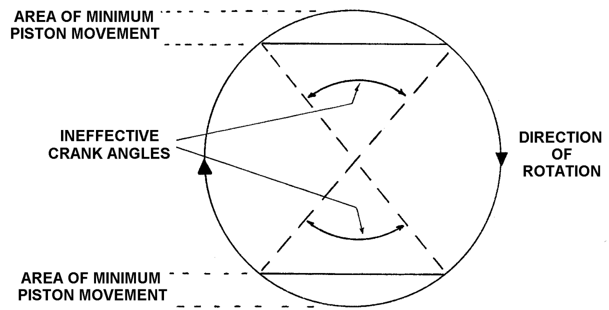
13. It will be seen from Figure 11-3 that the spark occurs shortly before the end of the compression stroke, so that the burning mixture achieves maximum pressure early on in the power stroke. The sequence of valve lead, valve lag and ignition advance is illustrated at Figure 11-3. The period at the end of the exhaust stroke/beginning of the induction stroke when both valves are open is termed the valve overlap.

14. There are two periods during each revolution of the crankshaft during which the linear movement of the piston per degree of rotation of the crankshaft is very small. These times are termed the ineffective crank angles. They occur at either side of TDC and BDC, and are shown at Figure 11-4.

FIGURE 11-4
Ineffective Crank Angle



15. Opening and closing valves during the ineffective crank angle periods does a lot to achieve the aims listed above, whilst preventing undue stresses occurring to the valves and valve gear.

16. The various factors that affect valve timing increase as the rotational speed of the engine increases. Consequently high-speed engines have considerable valve lead, lag, and overlap. With such engines, as engine speed increases it is necessary to advance the spark, in other words to cause the spark to occur at a greater angle before TDC. This ensures that maximum combustion pressure still occurs early in the power stroke. The strength of the mixture used will affect the amount of ignition advance that is required, since rich mixtures burn faster than weak mixtures.

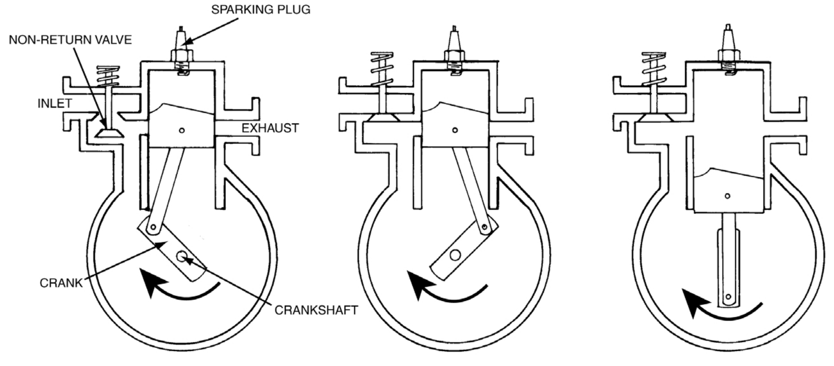
17. This advance of the ignition timing with increasing engine rpm is achieved automatically.

18. Modern, air-cooled aircraft piston engines do not rotate at sufficiently high speed to require automatic ignition advance and the ignition timing remains constant over the operating speed range of the engine. It is worth noting however that the point at which ignition of the mixture takes place, after the spark has occurred, will be progressively retarded (i.e. nearer to TDC) with increasing engine rpm. This is because the time taken for the fuel to ignite is constant, but the time taken for the crankshaft to rotate a given number of degrees is less with increasing rpm.

The Two-Stroke Cycle

19. A very few small single-engine aircraft use an engine that operates to the two-stroke cycle, in which the complete cycle of operations is completed in two strokes of the piston instead of four. A simple diagram of a two-stroke engine is shown at Figure 11-5.

FIGURE 11-5
Two-Stroke Cycle



20. There is no inlet or exhaust valve. Inlet and exhaust ports in the side of the cylinder admit the fuel air mixture and allow the expulsion of exhaust gases as the movement of the piston uncovers them. The crown of the piston is specially shaped to aid the flow of the gases into and out of the cylinder.

21. Referring to Figure 11-5, let us start the description of the two-stroke cycle with the piston about halfway up the cylinder, as in Figure 11-5a. A charge of fuel-air mixture is being compressed in the upper part of the cylinder whilst the upward movement of the piston is creating reduced pressure in the lower part of the cylinder and crankcase. The pressure difference, between the outside atmosphere and the crankcase causes a non-return valve in the inlet manifold to open and admit a fresh charge of fuel-air mixture into the crankcase. This is the compression-induction stroke.

22. As the piston approaches the TDC position a spark from the sparking plug ignites the mixture and the heat of combustion expands the gas to force the piston down, driving the crankshaft through the connecting rod. The downward movement of the piston compresses the new charge in the crankcase, raising its pressure above atmospheric and causing the intake non-return valve to close, as shown in Figure 11-5b.

23. Continuing its downward travel the piston first uncovers the exhaust port, allowing the waste combustion gases to begin escaping to atmosphere, and then uncovers the inlet port. The slight overpressure in the crankcase forces the new charge into the cylinder through the inlet port and the shape of the piston crown directs its flow towards the top of the cylinder. The incoming charge assists with the final expulsion of the exhaust gases, known as scavenging. This is illustrated at Figure 11-5c.

24. Although simpler in construction than the four-stroke engine, the two-stroke spark ignition engine is much less efficient and is difficult to cool. It is therefore unsuitable for use as an aircraft engine except where the power required is relatively small, such as in powered gliders and microlights.

Power and Efficiencies

Power

25. Work is said to be done when the point of application of a force moves and is measured by the product of the force and the distance moved in the direction of the force.

Work = force x distance moved (in the direction of the force)

Power is defined as the rate of doing work.

$$\text{Power} = \frac{\text{Work Done}}{\text{Time Taken}}$$

Horse Power. When James Watt was developing and marketing his steam engines he used the horse, the most familiar 'engine' of the day, as a reference for comparison of power. He estimated (perhaps rather generously) that the average horse was capable of a rate of doing work of 550 foot pounds per second (ft lb/s) or 33,000 ft lb/min. and he arbitrarily used this rate to define one horsepower (1HP). Thus, when an engine is said to be rated at 20 HP it means that it is capable of doing work at the rate of (20 x 33,000 ft. lb/min) 660,000 ft. lb/min. One horsepower is equal to 745.7 watts

26. A piston engine produces power by converting the heat energy of a burning fuel/air mixture into the reciprocating motion of a piston in a cylinder, subsequently mechanically converted to rotary motion by the connecting rod and crank.

27. The power produced at the crankshaft may be quantified in terms of the theoretical or indicated power available from the burning mixture (IHP), or in terms of the effective power, after frictional losses, (BHP).

28. Indicated Horsepower (IHP) takes no account of any work done within the engine to overcome friction. It is calculated using the formula:

$$\text{IHP} = \frac{\text{PLANK}}{33,000}$$

Where:

- P = The mean effective pressure (lb/in²) as measured in the cylinders by a special pressure gauge.
- L = The length of the piston stroke in feet.
- A = The area of the pistons (in²).
- N = The number of working strokes per minute.
- K = The number of cylinders.

29. The indicated horsepower is a purely theoretical value and is reduced by the friction horsepower to give brake horsepower (BHP). This is the power actually delivered to the propeller gearing.

30. The ratio of BHP to IHP is known as the mechanical efficiency of the engine and is expressed as a percentage. A figure of about 80% is normal.

31. To complete the power train, the power that is converted into thrust, after the various propeller losses have been taken into account, is known as thrust horsepower.

32. Whilst the mechanical efficiency of a piston engine is not great, when compared with a gas turbine, it is its poor thermal efficiency which is its greatest disadvantage.

Engine Efficiencies

Thermal Efficiency is a measure of the heat losses that occur during the conversion of the heat energy of the fuel to useful mechanical work. In a typical engine only approximately 30% of the heat energy of the fuel is converted into useful work. The remainder is lost to engine cooling (25%), the heat carried away in the exhaust (40%) and mechanical work to overcome friction and 'pumping' losses (5%).

33. The thermal efficiency of an engine is the ratio of the heat energy converted into useful work to the heat energy of the fuel. It may be based upon either brake horsepower (BHP) or indicated horsepower (IHP). It is given by the formula:

$$\text{Brake Thermal Efficiency} = \frac{\text{BHP x 33000}}{\text{wt/min fuel burned x heat value (BTU) X 778}}$$

34. (778 is a conversion factor, 1 BTU (British thermal Unit) = 778 ft lb)

35. For Indicated Thermal Efficiency (ITE) the word "indicated" is substituted on both sides of the equation.

36. For many modern aircraft piston engines an excellent thermal efficiency at full power would be about 34%. At slightly reduced power the thermal efficiency may improve slightly. By using high compression ratios and high octane fuels (both discussed in detail at a later stage) it is possible to achieve thermal efficiencies in a petrol engine as high as 40% although, for mechanical reasons, this is not attempted in many practical aero engines. Even at 34%, the petrol engine has a better thermal efficiency than other types of piston engine.

Volumetric Efficiency is the ratio of the volume of the induced fuel-air charge at atmospheric pressure to the piston displacement.

$$\text{Volumetric Efficiency} = \frac{\text{volume of charge at atmospheric pressure}}{\text{piston displacement}}$$

37. An un-supercharged (normally aspirated) engine induces the fuel-air charge into the cylinder by creating a depression (a pressure lower than ambient atmospheric pressure) in the cylinder. Hence the induced charge will be at a lower-than-atmospheric pressure. Remembering the gas laws it will follow that, if this mass of induced charge were at atmospheric pressure, its volume would be less.

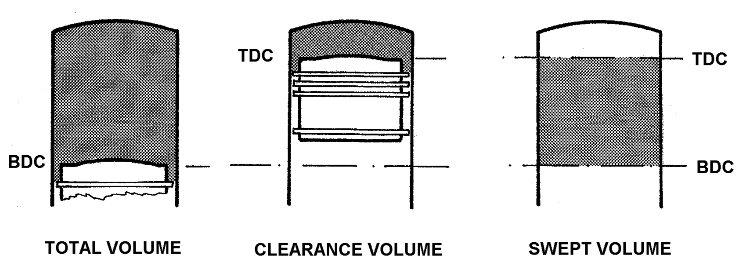
38. Factors which tend to decrease the mass (and therefore volume at atmospheric pressure) of the induced charge have an adverse effect on the volumetric efficiency of an engine.

39. Volumetric efficiency will be affected by the following factors: (1) Partial throttle opening causing restricted mixture flow to the cylinders. (2) Long induction systems of restricted cross-sectional area. Friction increases directly with length of manifold, and inversely with cross-sectional area. (3) Sharp bends in the induction system, causing deceleration of the airflow. (4) High inlet air temperature, causing low density. (5) High cylinder head temperature, also causing low density. (6) Incomplete scavenging. If the spent charge within the cylinder is not completely exhausted through the exhaust valve, there is less available volume for the incoming charge. (7) Incorrect valve timing. The degree of opening of the inlet valve determines the amount of inflow to the cylinder and the degree of opening of the exhaust valve determines the outflow (escape) of exhaust gases. The inlet valve must be opened as wide as possible throughout the induction stroke of the piston, the exhaust valve must close immediately exhaust flow from the cylinder ceases. (8) Engine rpm. Volumetric efficiency may be limited at high engine rpm because of increased friction in the intake, carburettor and valve ports.

40. The ambient atmospheric pressure opposes the escape of exhaust gases from the engine, this is known as back pressure. As altitude is increased the fall in ambient pressure means that the effective exhaust back pressure decreases. This means that the exhaust gas will escape more easily and less will remain in the cylinder after the exhaust valve closes during the induction stroke. Consequently, volumetric efficiency improves with altitude.

Compression Ratio. The compression ratio is the ratio of the volume of the cylinder with the piston at bottom dead centre (BDC), to the volume of the cylinder with the piston at top dead centre (TDC). Putting it another way, the compression ratio is the ratio of total volume to clearance volume. Total volume, clearance volume and swept volume are explained diagrammatically at Figure 11-6.

FIGURE 11-6
Cylinder Volumes



The total volume of the cylinder is the volume above the piston when it is at BDC.

41. The clearance volume of the cylinder is the volume above the piston when it is at TDC.

42. The swept volume is the difference between the total and the clearance volume, and is equal to the cross-sectional area of the piston multiplied by the stroke. Remember that the stroke is the crank throw multiplied by two.

43. Piston displacement is the volume displaced by the piston during its induction stroke. It is the same as the swept volume.

44. The power developed by a piston engine depends upon the mass of the fuel-air mixture induced during the induction stroke. Bearing in mind the gas laws and the definition of density (mass per unit volume) it will be seen that any factor that affects the density of the induced charge must affect the mass of the charge and therefore the power developed.

45. Factors that affect density of a gas are temperature and pressure. If altitude is increased temperature decreases, which tends to increased air density. However, the reduction in air pressure has a much more marked effect in reducing density. Since density decreases with altitude, the mass of the induced charge decreases and consequently engine power developed decreases.

46. Similarly, if the temperature of the induced charge is increased, its density is lowered. Use of heated intake air (to prevent carburettor icing) will reduce the density (and therefore mass) of the induced charge, resulting in a power reduction.

47. The density of the fuel/air mixture will also vary with humidity. The more humid the air the lower its density and therefore the lower the weight of the induction charge.

48. The power reduction with increasing altitude, referred to above, is of the order of 3.5% per 1000 feet of altitude increase. The diagram at Figure 11-7 shows graphically the change of power with altitude with a normally aspirated petrol engine.

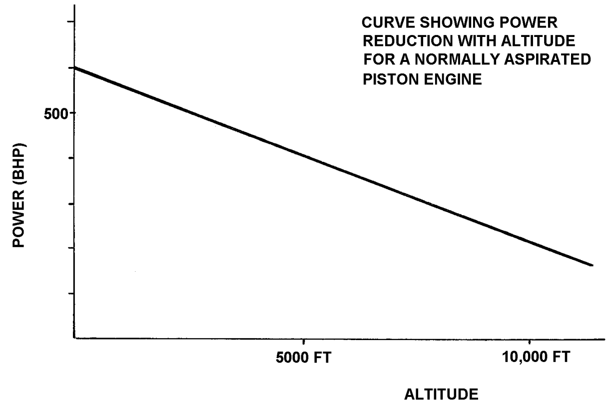


FIGURE 11-7
Power Reduction with Altitude (Normally-Aspirated)

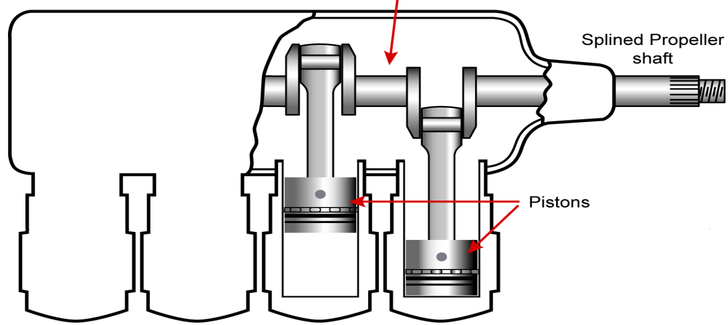
Engine Design Types

49. Aircraft piston engines are typified by the configuration of the cylinders relative to the crankshaft. The principal configurations are In-Line, V-type, Radial and Horizontally Opposed.

In-Line Engines. In this configuration the cylinders are arranged in a single line from front to rear. The arrangement was popular for racing aircraft in the period between the World Wars and continued in training aircraft for many years. Probably the best known is the De Havilland Chipmunk. In this, as in a number of other examples the cylinders were mounted beneath the crankshaft to give what is known as an inverted engine. The small frontal area of the engine enabled it to be housed in a narrow, low drag cowling. In-line engines were usually air-cooled, the air being scooped in on one side of the engine and directed over the cylinders by baffles.

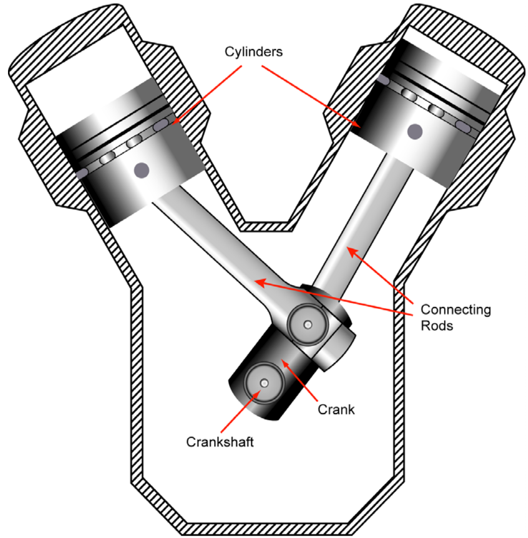
50. The cylinder arrangement required a long crankshaft, which led to the major disadvantage of the type since the crankshaft is the heaviest component of any piston engine. Hence the power-to-weight ratio of the in-line engine was poor. In general the type used four or six cylinders. Figure 11-8 shows a partly sectioned side view of an inverted in-line engine.

FIGURE 11-8
Part-Section View of Inverted In-line Engine



V-Type Engine. A natural development of the in-line engine was to mount two lines of cylinders, set apart by 45° or 60° and operating onto a common crankshaft. The small frontal area of the engine made it popular for use in high performance aircraft. V-12 versions, that is six cylinders in each bank, were developed and improved just before and during the Second World War, probably the best known being the Rolls Royce Merlin of Spitfire fame. The type was liquid-cooled, since air cooling of such a large and powerful engine would have been totally inadequate. The power-to-weight ratio was better than the straight in-line engine, and a great deal of effort was expended trying to improve this even further by adding a third and even a fourth bank of cylinders on a common crankshaft, but with no great success. Figure 11-9 shows a sectioned end view of a V-type engine.

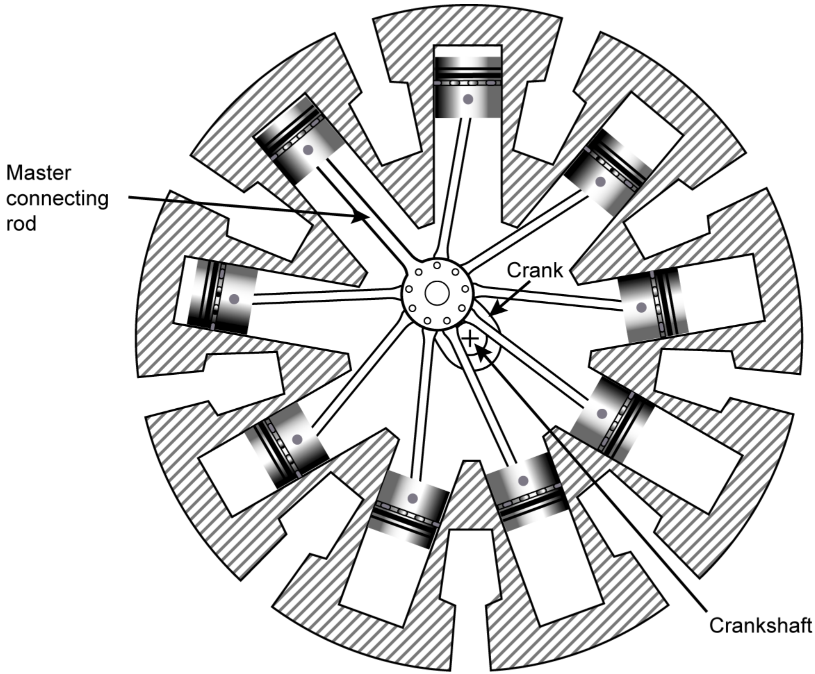
FIGURE 11-9
Sectioned End-View of V-Engine



Radial Engines. From the earliest powered aircraft both airframe and engine designers sought to improve the power-to-weight ratio by reducing the proportion of the crankshaft weight in the total weight of the engine. The rotary radial engine used in many World War One aircraft went as far as it is possible to go by having no crankshaft at all. The propeller was attached to the cylinder block in which, typically, seven cylinders were mounted radially in the same plane. The piston connecting rods were attached to a common point on the airframe, offset from the axis of the propeller. The entire engine and propeller rotated and as it did so the pistons reciprocated in their cylinders. The torque of this mass thrashing around can be imagined, causing the pilot control problems at take-off. Also lubrication presented unique difficulties. The basic principle was sound however, and led to the development of the static radial engine and the most powerful type of aircraft piston engine produced.

51. From the 1920's until superseded by the gas turbine, almost all large military and civil aircraft used radial piston engines. In this type an odd number of cylinders extend radially from the centreline of the crankshaft, the cylinders being spaced evenly in a common flat vertical plane. The piston in one of the cylinders is connected by a master connecting rod to the single crank throw. All the remaining pistons are connected to the master rod through articulated connecting rods. Figure 11-10 shows a partly sectioned end view of a nine-cylinder radial engine.

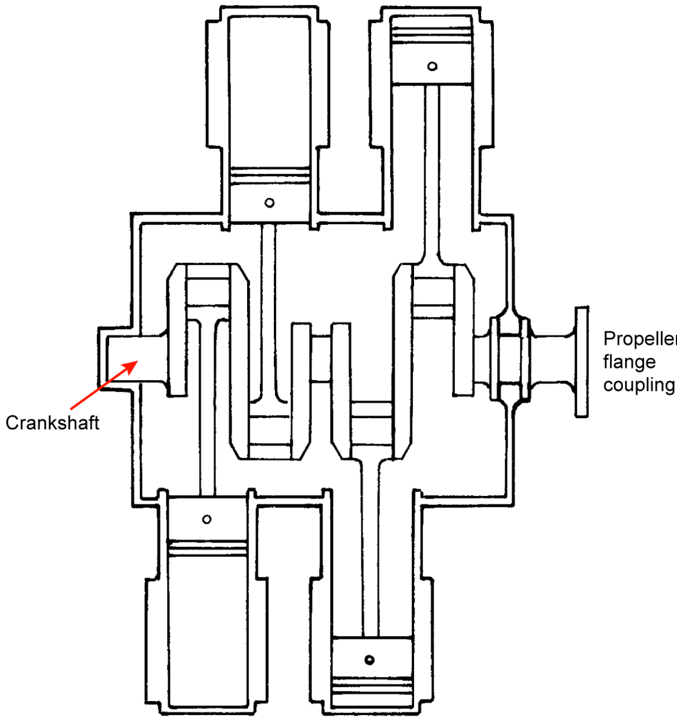
FIGURE 11-10
Partly Sectioned End View of Nine Cylinder Radial Engine



52. Radial engines most commonly used seven or nine cylinders, although three and five cylinder versions were built. An odd number of cylinders is necessary to achieve even distribution of power to the crankshaft. The engine operates on the four-stroke cycle, so it takes two complete revolutions of the crankshaft for all cylinders to fire. Greater power and improved power-to-weight ratio is achieved by adding a second row of cylinders, operating on a common crankshaft and 14 and 18 cylinder two-row radial engines had become common by the start of the Second World War. The most powerful aircraft piston engine used in production aircraft was a four-row, 28-cylinder radial engine, the Pratt and Whitney R-4360.

Horizontally Opposed Engines. By far the most popular aircraft piston engine in present day use is the horizontally opposed type, otherwise known as the Flat or O-Type engine. For fixed wing aircraft the engine is arranged with the cylinders lying horizontal on either side of the crankshaft, which is parallel to the longitudinal axis of the aircraft. The power-to-weight ratio is high because the crankshaft is kept short, about half the length of an equivalent power in-line engine. Horizontally opposed engines have been produced with two, four, six and eight cylinders, although four and six are the most common configuration. The design is light, because of the short crankshaft and because the short engine length lends itself to air cooling. Furthermore it has a small frontal area keeping drag to a minimum. Figure 11-11 shows a partly sectioned plan view of a four-cylinder opposed-piston engine.

FIGURE 11-11
Partly Sectioned View of Four-Cylinder Opposed Engine

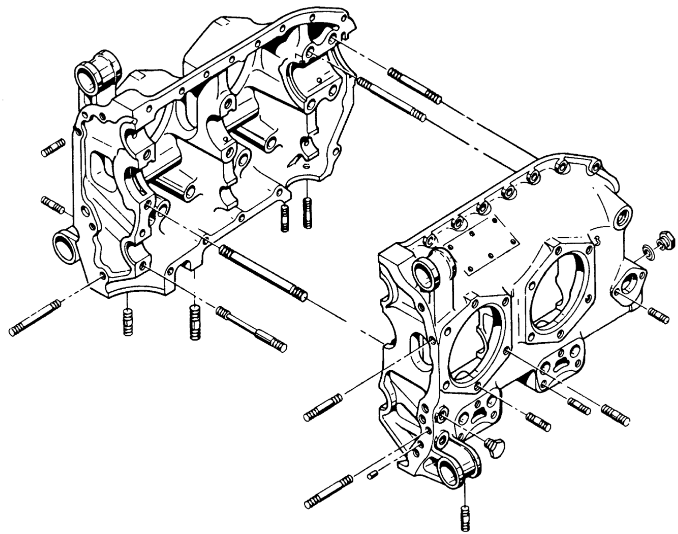


Mechanical Components

53. The principal mechanical components of the aircraft piston engine are the crankcase, the cylinders and valves, the valve operating mechanism, the pistons and connecting rods and the crankshaft and its bearings.

The Crankcase. Figure 11-12 illustrates the crankcase of a four cylinder opposed engine. It is an aluminium alloy casting in two parts, divided along the centreline of the engine and joined by threaded studs and nuts. Cast into the casing are the housings for the crankshaft bearings and valve camshaft bearings. At the sides of the crankcase are holes for the passage of the piston connecting rods, with machined facings to which the cylinders will be bolted. The crankcase casting also includes the attachment points by which the engine will be attached to the airframe in the engine nacelle of the aircraft.

FIGURE 11-12
Crankcase of Four-Cylinder Opposed Engine



The Cylinder. The cylinder of a piston engine must be strong enough to withstand the pressure caused by the expansion of the hot combustion gases and it must be able to dissipate the heat of combustion. The two major components of the cylinder are the cylinder barrel and the cylinder head.

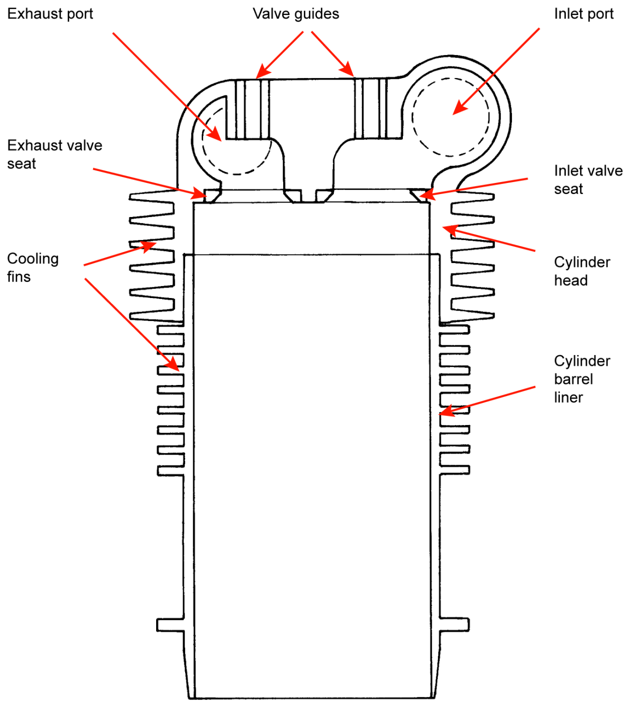
54. The cylinder barrel, within which the piston moves up and down, is usually made from chrome-molybdenum steel or chrome-nickel-molybdenum steel to provide the good bearing surfaces and high structural strength required. It is machined to a highly polished surface internally, and externally fins are machined to increase the surface area and assist with heat dissipation. The upper end of the barrel is threaded externally for attachment to the cylinder head.

55. The cylinder head forms the combustion chamber for the fuel-air mixture. It contains the machined guides for the inlet and exhaust valves, the mounting lugs (or bosses) for the valve rocker mechanism and the threaded holes into which the sparking plugs are inserted. Cylinder heads are usually cast in aluminium alloy and include extensive finning on the outside to dissipate the considerable heat of combustion.

56. The lower part of the combustion chamber in the head is threaded to receive the cylinder barrel and is deliberately machined so that its inside diameter is less than the outside diameter of the barrel. To fit the barrel into the cylinder the barrel is chilled, shrinking it, and the cylinder head is heated to expand it. The two are then threaded together and as the temperatures equalise the head shrinks to form a pressure-tight fit on the barrel. This process allows for the differential expansion rates of the two materials when the engine is operating.

57. A typical opposed engine cylinder is shown in sectional view at Figure 11-13.

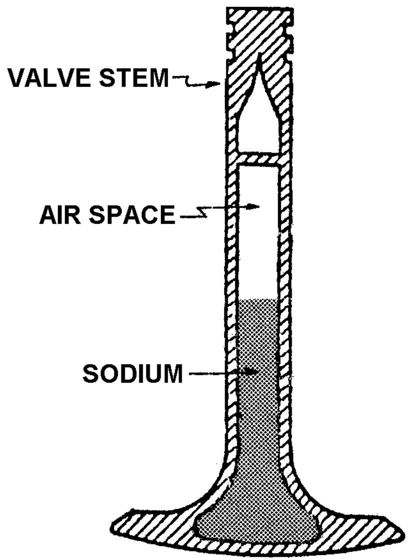
FIGURE 11-13
Cross-Section of a Cylinder



Inlet and Exhaust Valves. A valve is a movable mechanism which forms a pressure-tight seal when closed, but which can be opened to allow the passage of a fluid. The purpose of the inlet and exhaust valves in a piston engine is to open and close the inlet and exhaust ports as required during the operating cycle of the engine.

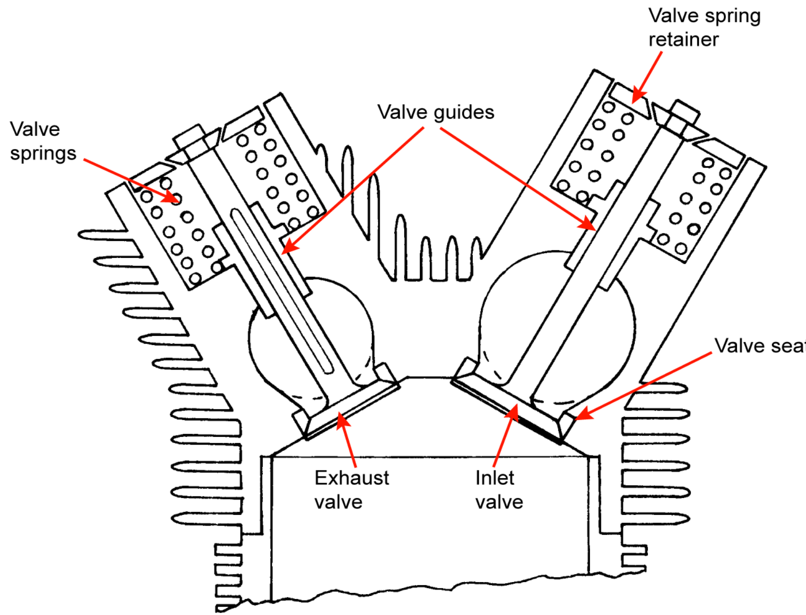
58. The type of valve almost universally employed for this purpose is the poppet valve, so-called because it pops open against the action of a spring and pops closed with the assistance of the spring. The valve comprises a parallel-sided stem, which slides in a valve guide, and a circular flat head set at right angles to the axis of the stem. The periphery of the head is machined to a smooth surface that will mate with the valve seat in the cylinder head to form a gas-tight seal. Valves are normally made of special steel alloys, and the exhaust valve may be sodium filled. The sodium becomes liquid at engine working temperatures and assists in conducting heat from the valve head, through the stem, to the valve guide where it is dissipated by the engine coolant. Such a valve is illustrated at Figure 11-14.

FIGURE 11-14
Sodium-Filled Valve



59. The inlet valves do not need cooling to the same extent as the exhaust valves since the incoming mixture assists in cooling. They are usually made with solid stems and flat or concave heads. Figure 11-15 shows a typical arrangement of valves, valve guides and closing springs in a cylinder head.

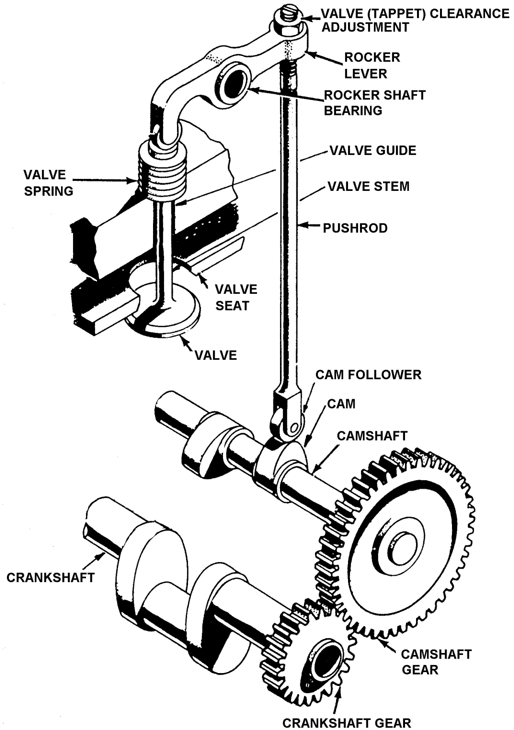
FIGURE 11-15
Valves, Guides and Springs in a Cylinder



60. The exhaust valve seat is typically made from steel alloy and the inlet valve seat from bronze and they are either shrunk or threaded into the cylinder head. Double valve springs are used, that is two helical-coil springs of different diameter, to prevent the valves from bouncing as they close due to the natural vibration frequency of a spring. The springs are of different gauge wire and different pitch to dampen out the natural frequency of each other.

Valve Operating Mechanisms. A valve operating mechanism is shown at Figure 11-16.

FIGURE 11-16
Valve Operating System

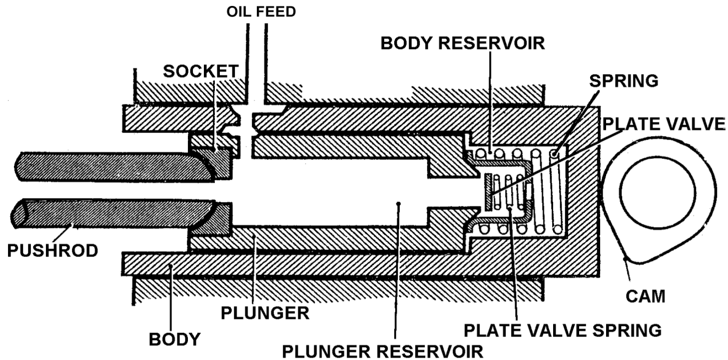


The camshaft is driven by the crankshaft through reduction gearing with a ratio of 1 to 2, so that the camshaft rotates at half crankshaft speed. The cam, forged as part of the camshaft, transmits linear motion through the cam follower to the push rod as the camshaft rotates. The linear movement of the push rod is transferred to the valve by means of the rocker lever, which is pivoted about a rocker shaft attached to the engine cylinder head. The movement of the rocker lever pushes the valve open against a strong spring. This spring serves to close the valve as the cam rotates further and linear movement of the push rod reverses.

61. In order to accommodate the differential expansion of the cylinder and the push rod as the engine heats up during operation, an adjustment to the effective length of the push rod is provided. This is carefully set so that, when the valve is shut, with the engine cold, a small clearance exists between the end of the valve stem and the rocker lever. This clearance, which will be just (and only just) absorbed by differential expansion rates, is known as valve clearance and is often referred to as tappet clearance. The tappet is an alternative name given to the push rod, or sometimes a part of the push rod.

62. Most opposed engines are fitted with hydraulic tappets (see Figure 11-17). This type of tappet consists basically of a body and plunger, with an internal spring and non-return valve, and a push rod socket. During operation, pressure oil supplied to the tappet is picked up by a groove around the body at the point when the tappet is near the outer end of its stroke. This oil lubricates the tappet bearing surface and enters the plunger reservoir through a port in the plunger wall. It then passes through the push rod socket and hollow push rod to lubricate the socket mechanism.

FIGURE 11-17
Hydraulic Tappet



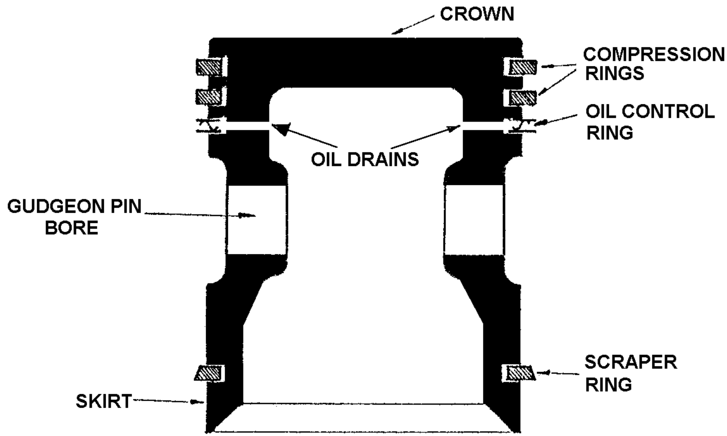
63. If clearance is present in the valve operating mechanism when the tappet is resting on the cam dwell, the spring in the tappet body pushes the plunger outwards to eliminate this clearance. At the same time the non-return valve will open to allow oil to pass into the body reservoir. As the cam lobe commences to push on the tappet, the non-return valve closes and a hydraulic lock is formed, thereby transmitting motion to the push rod. Thus clearance is eliminated from the mechanism. Valve closure will be unaffected, since the force applied by the tappet spring is much less than that of the valve springs.

The Piston and Connecting Rod. The piston forms a solid gas-tight plug within the cylinder, and is made of a forged aluminium alloy that is light but strong. Pistons are tapered slightly towards the crown, to prevent distortion at the high operating temperatures involved. The crown of the piston may be shaped to suit the cylinder head, giving the necessary valve clearances, and to add swirl to the inducted air/fuel mixture.

64. Piston compression rings are fitted into grooves in the side of the piston to maintain a gas seal between piston and cylinder. The rings are usually made of cast iron, which retains its springiness at high temperatures. The rings when new have a slight taper of the face which contacts the cylinder, giving a rapid initial wear, and therefore allowing the ring to wear to the bore of the cylinder during the first few hours of operation.

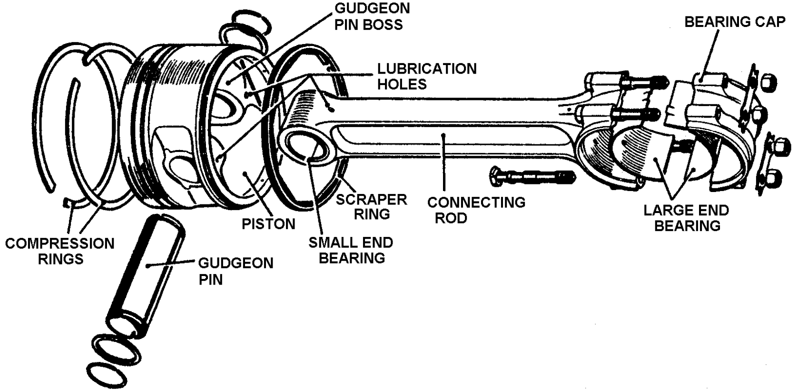
65. In addition to the compression rings, an oil control ring is fitted below the lowest compression ring to regulate the thickness of the oil film on the cylinder walls. This is to ensure that there is sufficient oil for lubrication of the compression rings, but not an excess of oil that would cause carbonisation problems in the combustion chamber. Some pistons also have an oil scraper ring fitted around the piston skirt for oil retention; otherwise too much oil might escape to the crankcase. The piston is attached to the connecting rod by a gudgeon pin, which forms a sliding fit in machined holes in the piston walls and is retained by means of circlips. A piston is illustrated at Figure 11-18.

FIGURE 11-18
Typical Piston



66. The connecting rod is normally made in H section out of alloy steel. The H section gives a greater inherent strength to the rod, and prevents the rod from bending under the considerable loads imposed upon it. A connecting rod and piston assembly is shown at Figure 11-19.

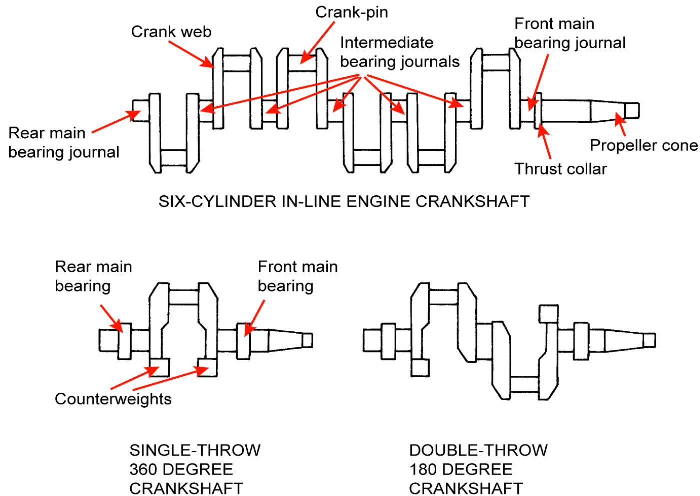
FIGURE 11-19
Connecting Rod & Piston Assembly



67. The crankshaft is a solid steel forging having a number of throws, or crank arms. In an in-line or opposed engine the number of throws is equal to the number of cylinders. The large end bearing of the connecting rod fits around the crank pin between the crank throws. At each end of the crankshaft a journal is machined to fit into a supporting bearing in the engine crankcase. Multi-throw crankshafts, as in a six-cylinder in-line engine for example, have further bearing journals between each pair of crank arms.

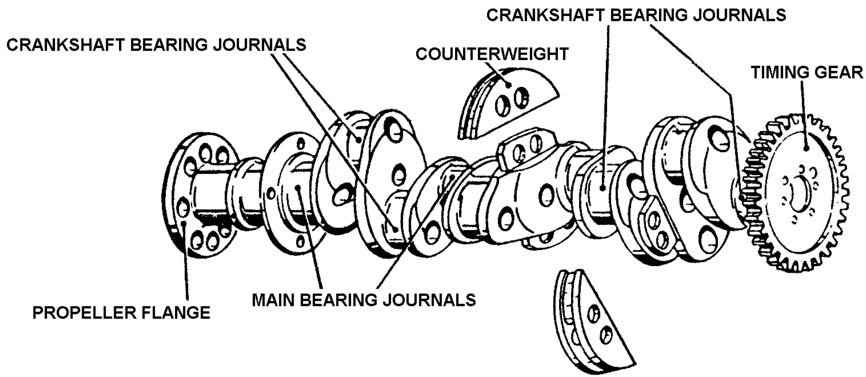
68. Single-throw, or 360° crankshafts are used for single row radial engines. In order to dynamically balance the crankshaft, counterweights must be fitted to the crank arms, or webs as they are sometimes called. Two row radial engines require a double-throw, or 180° crankshaft. In a six-cylinder in-line engine the cranks will be at 60° to each other, in other words a 60° crankshaft. These various types of crankshaft are illustrated at Figure 11-20.

FIGURE 11-20
Types of Crankshaft



69. The valve timing gear, which drives the camshaft, is usually mounted at the rear end of the crankshaft whilst the propeller flange is fitted to the opposite end. In some cases the propeller end of the crankshaft is splined to take the propeller. A four-throw (90°) crankshaft with counterweights, timing gear and propeller flange is illustrated at Figure 11-21.

FIGURE 11-21
Crankshaft with Counterweights & Timing Gear



Bearings. There are three types of bearing commonly used in aircraft piston engines, plain bearings, roller bearings and ball bearings.

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70. Plain bearings consist of two semi-circular shells that fit closely around a shaft journal and an example may be seen in the large end bearing shown at Figure 11-19. This type of bearing is best suited to radial loads in low-power engines. The plain bearing requires forced oil lubrication to prevent friction wear and overheating.

71. Roller bearings are very low friction bearings and are suitable for high radial and thrust (axial) loading. They consist of a ring of hardened steel rollers contained in a cage. In radial loaded bearings, such as a crankshaft support bearing, the rollers travel within an outer race, or guide, which is attached to the crankcase and roll around an inner race that is shrunk onto the crankshaft.

72. The construction of a ball bearing is similar to that of a roller bearing, with balls being used instead of rollers. Ball bearings have the lowest friction of any type of bearing. In many applications ball and roller bearings can be manufactured pre-lubricated and sealed for life.

Piston Engine Lubrication and Cooling

Piston Engine Lubrication and Cooling

Lubricants

1. The two principal functions of a lubricant are to reduce friction between moving surfaces, and to act as a coolant. Obviously two dry moving surfaces in contact with each other would soon wear, and would generate large amounts of heat. To prevent this, a thin film of lubricant is interposed between the surfaces, separating them, so that the friction is dramatically reduced.

2. The lubricants used in aircraft piston engines are mineral lubricants, the source of which is petroleum. Lubricants may be classified as solids, semisolids and fluids. Solid lubricants have little use in aircraft engines, except occasionally as additives to engine lubricating oil. Examples are mica and graphite. Semisolid lubricants, principally greases, may be used to lubricate ball and roller bearings but are unsuitable for circulating or continuous-operation lubrication systems as used in aircraft engines. Grease is a mixture of oil and soap.

3. Fluid lubricants (oils) are the principal lubricants in all types of internal combustion engines because they can be easily pumped around a circulating system, they absorb and dissipate heat readily and they provide a good ‘cushioning’ effect in bearings.

Properties of Lubricating Oil

4. The most important properties of an aircraft engine lubricating oil are its flash point and its viscosity.

5. Flash point is defined as the temperature to which oil must be heated in order to give off sufficient vapour to burn momentarily when brought into contact with a very small flame. Aircraft piston engines are air-cooled and so operate at high temperatures (compared to liquid-cooled engines). Consequently it is important that the lubricating oil used shall not vaporise too readily since, if the vaporised oil burns then, among other considerations, the engine will not be properly lubricated.

6. Viscosity is defined as the fluid friction of a liquid, in this case the oil. In simple terms it is a measure of the ease with which the oil will flow. The lower the viscosity the more freely the oil flows. In order to properly lubricate all the working parts of an engine the oil must have sufficient viscosity to give cushioning and "slipperiness", whilst still being capable of flowing freely through the lubricating oil system. The viscosity of oil tends to decrease with increasing temperature, although an ideal lubricating oil would maintain constant viscosity over the whole range of working temperatures of the engine, from cold, winter starting to hot, high temperature running.

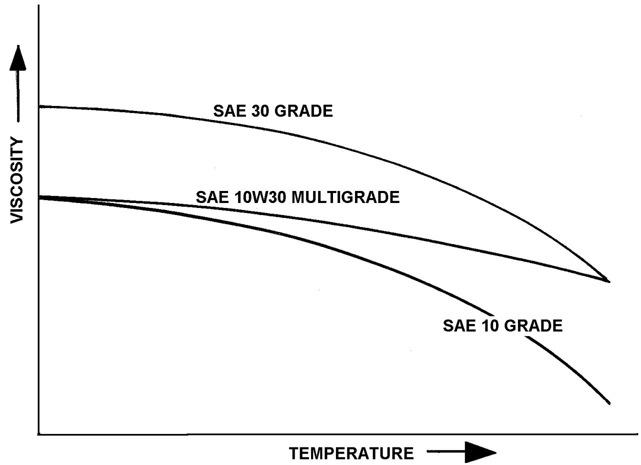
Oil Grades

7. Lubricating oils are graded, according to their viscosity, by a series of numbers allocated by the Society of Automotive Engineers (SAE). The SAE number is determined from the time taken for a given quantity of oil, at a specific temperature, to flow through an orifice of fixed size. Thus, the higher the SAE number, the higher the viscosity of the oil. In other words, SAE 10 grade oil flows more freely than SAE 50 oil.

8. An engine that is to be operated in cold climates will require less viscous oil than one that is to be operated in the tropics. However, if SAE 10 grade oil were selected for the former it may well prove too thin (insufficient viscosity) if the aircraft were taken to warmer climates. Similarly, if SAE 50 grade oil were selected for the warmer climates the oil would undoubtedly prove too viscous (thick) if the aircraft were taken to a cold climate. This would be especially problematical on start-up, when the thick oil would not circulate readily to all parts of the engine.

9. Because aircraft frequently operate in a range of climates and seasons these single-grade oils are unsuitable. Consequently, multi-grade or multi-viscosity oils are used which provide adequate lubrication over a much wider range of temperatures than a single-grade (or straight-grade) oil. Figure 12-1 shows, graphically, a comparison between two single-grade oils of SAE 10 and SAE 30 and multi-grade SAE 10W30 oil.

FIGURE 12-1
Oil Viscosities



Specific Gravity

10. In addition to its viscosity grade, the specific gravity (SG) of oil is usually given. If not, it is usual to assume a specific gravity of 0.9. Specific gravity is the weight of a substance compared with the weight of an equal volume of a standard substance (usually distilled water). It is necessary to know the specific gravity of a substance if its weight is to be calculated given its volume. One litre of distilled water weights one kilogram, one Imperial gallon weighs 10lbs. Hence, 10 litres of oil at SG 0.9 weighs 9 kilograms; one Imperial gallon of the same oil weighs 9 lbs.

Functions of Lubricating Oil

11. The lubricating oil in an aircraft piston engine serves a number of functions:

(a) Reduction of friction between moving parts, by interposing a film of oil.

(b) Cooling various parts of the engine by heat dissipation. Heat is transferred from hot metal engine parts (by convection) to the cooler lubricating oil.

(c) Sealing the combustion chamber by filling the small space between piston rings and cylinder walls, thus preventing the flow of combustion gases from combustion chamber to crankcase.

(d) Cleaning the engine by carrying sludge and residue (mainly products of combustion) from the moving parts and depositing them in the oil filter.

(e) Preventing corrosion by protecting metal parts of the engine from oxidising agents (oxygen & water).

(f) Providing a cushion between parts where impact loads occur. Large and small end bearings and crankshaft bearings are typical examples.

Lubrication Methods

12. The lubricating oil is distributed to the moving parts of a typical aircraft piston engine either by a pressurised system or by splash lubrication. In most cases a combination of the two is used.

13. In a pressure lubrication system an engine-driven mechanical pump supplies oil under pressure to the bearings. Oil is supplied to the pressure pump from the engine sump and the pump forces oil into an oil manifold, which distributes oil to the crankshaft bearings. A pressure-relief valve usually controls the oil delivery pressure. From the crankshaft, or main bearings oil passes through holes drilled in the crankshaft to the lower (big end) connecting rod bearings. Oil from the manifold also supplies the hollow camshaft to lubricate the camshaft bearings and cams. The overhead valve mechanism (rocker arm bearings and valve guides) is often supplied through hollow push rods, the oil having first been used to pressurise the hydraulic tappets.

14. The engine cylinder walls and the piston gudgeon pins are often lubricated by splash lubrication, the source of which is oil spraying from the large end bearings into the crankcase and thrown onto the pistons by the flailing cranks.

Pressure Lubrication Systems

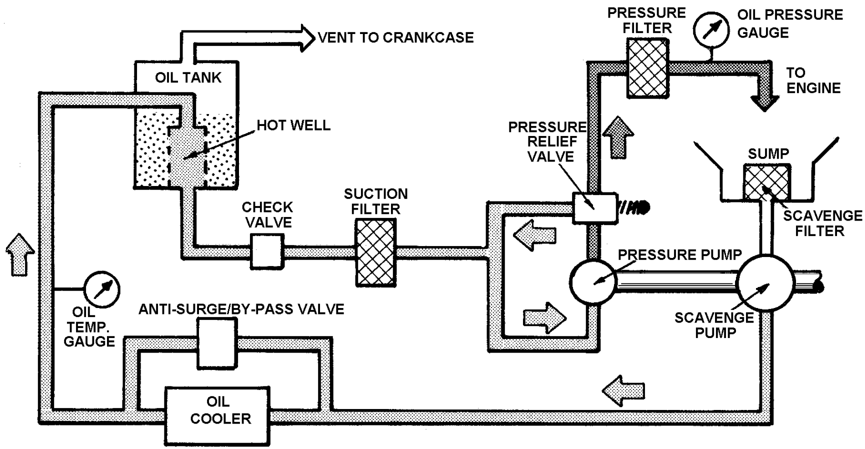
15. Aircraft piston engine pressure lubrication systems fall into two categories, the dry sump system in which the only oil contained within the engine when it is running is that which is lubricating the working parts, and the wet sump system in which all lubricating oil is contained within the engine.

The Dry Sump System

16. In a dry sump engine the oil not in circulation is stored in a tank rather than the engine sump. The sump is kept clear of oil, when the engine is running, by a scavenge pump, which passes the oil through a cooler and into the tank. The pressure pump draws its supply of oil, for engine lubrication, from the tank and delivers it to the engine bearings at a pressure controlled by the pressure relief valve.

17. The dry sump system is ideal for aerobatic aircraft, since the oil pump will not be starved of lubricant whilst the aircraft is inverted. A schematic diagram of a dry sump system is shown at Figure 12-2.

FIGURE 12-2
Dry Pump Oil System



18. Oil flows from the tank, through a suction filter to the inlet of the engine driven pump. This pump forces oil under pressure through a pressure filter to the shafts, bearings and other parts to be lubricated. In order to ensure that an adequate supply of oil is maintained at all times, the pump is designed to deliver more oil than the engine requires. The required oil pressure is then achieved by means of a pressure relief valve, which allows excess oil to be by-passed back to the suction side of the pump.

19. Cylinder walls can be lubricated either by the oil mist that forms when the oil forces its way out from between the bearings, or by jets directed to the walls. Pistons are cooled internally by oil jets directed under the crowns. Other lightly loaded components, for example camshafts, are lubricated by low-pressure oil, which is routed from the high-pressure supply line via a pressure-reducing valve.

20. The hot oil drains into the sump and is returned to the tank, normally via a cooler, by a scavenge pump. The capacity of the scavenge pump exceeds that of the oil delivery pump, and so there is no danger of oil accumulating in the engine sump.

21. The advantages of the dry sump system are summarised below;

(a) The system is suitable for inverted flight, as discussed.

(b) The volume of oil carried is not limited by the size of the engine sump.

(c) Higher engine rpm may be maintained without damage or overheating, since the oil is cleaned (filtered) and cooled following each journey through the engine.

(d) Oil is not permitted to accumulate in the sump, and so over-oiling cannot occur.

(e) The component parts of a dry sump system are now discussed.

(f) The oil tank normally contains baffles, which prevent a surge of oil during manoeuvres. The tank is normally separated from the engine by the firewall, and is mounted as high as possible so that gravity can aid the engine pump. An air space is always present above the maximum level of oil permitted in the tank. This air space allows for thermal expansion of the oil and frothing due to aeration. It also allows for displacement of oil from variable pitch propellers and the increased volume of oil delivered to the tank on start up, which may result from oil that has drained into the sump whilst the engine was shut down.

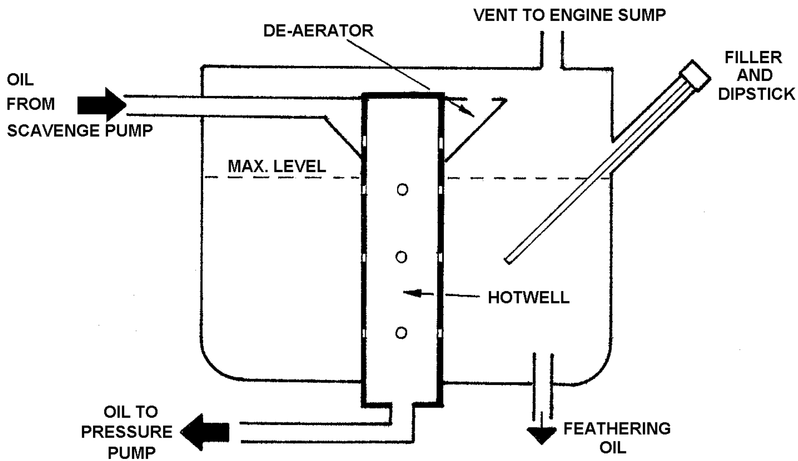
(g) Most dry sump oil tanks incorporate a chamber known as a hot well, or hot pot. The hot well ensures that only part of the oil supply circulates through the engine on start-up, so that optimum oil temperature and therefore viscosity is rapidly achieved.

(h) The hot well consists of a metal cylinder that covers the oil outlet to the engine. On start up the level of oil in the hot well drops, exposing a ring of small diameter ports that offer a high resistance to the high viscosity cold oil in the main tank. Consequently very little cold oil passes into the hot well. The circulated oil is returned to the hot well and is re-circulated. As the hot oil is returned, the hot well acts as a heat exchanger and progressively heats the main tank oil until a constant temperature is achieved throughout the tank.

(i) When feathering propellers are fitted, the lower ring of feed ports to the hot well are positioned above the bottom of the tank to provide an oil reserve for feathering in the event of main tank drainage due to a ruptured oil line, or excessive oil consumption in the engine.

22. The oil that is scavenged from the engine sump is routed over a de-aerator plate to the hot well. This plate separates the air from the oil to reduce frothing. The air gap in the oil tank is vented to the sump, and the sump is vented, via an oil breather to atmosphere. A typical dry sump oil tank is illustrated at Figure 12-3.

FIGURE 12-3
Dry Sump Oil Tank



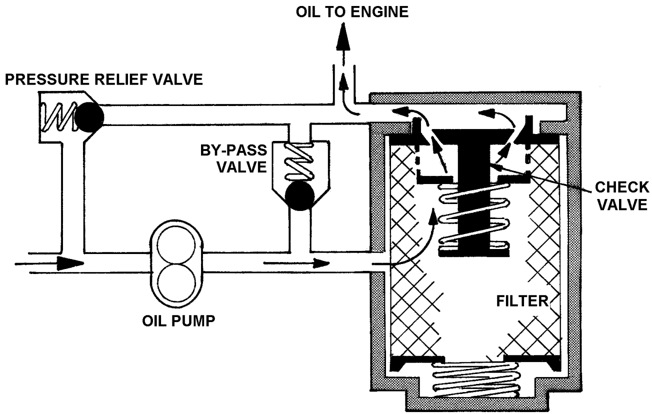
Oil Filters. Filters are incorporated in all oil systems, their purpose being to absorb foreign matter in the oil such as dirt and carbon. It is normal for a system to contain at least two filters, a suction filter and a pressure filter.

23. The suction filter is positioned upstream of the oil pump and is made of fairly coarse mesh to remove large particles of foreign matter. It thus protects the delivery pump itself.

24. The pressure filter is designed to remove very small particles of foreign matter before the oil is passed to the engine bearings. It is made of very fine mesh wire, or perhaps felt, reinforced with a wire gauze cover that prevents the element from collapsing. The filter is deeply corrugated to increase the filtration area, and is positioned over a filter spring.

25. Oil from the oil pump passes through the fine mesh of the filter and out to the engine by way of a check (non-return) valve. If the supply pressure to the engine exceeds a pre-set value the pressure relief valve opens, releasing excess pressure to the suction side of the oil pump. Clogging of the filter increases the pressure differential across it, and therefore across the by-pass valve. This will cause the by-pass valve to open, maintaining lubricating oil supply to the engine. This arrangement is illustrated at Figure 12-4.

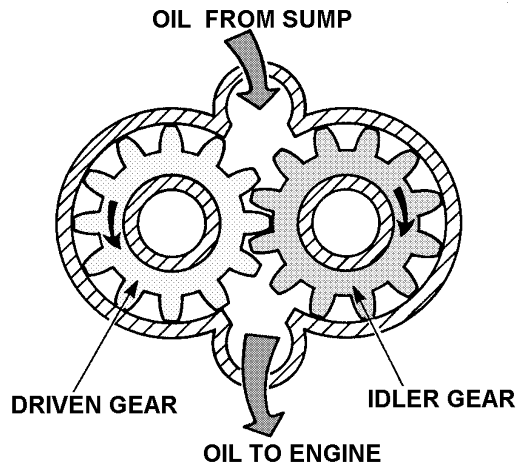
FIGURE 12-4
Oil Filter



26. Two important chemical changes take place in the oil during use. The first, oxidation, is due to contamination from corrosive lead salts produced during combustion. The second is the chemical effect of water vapour condensing inside the engine as the oil cools after engine shut down. No filter system will reverse these chemical changes, which will eventually destroy the lubricating qualities of the oil. It is therefore essential that the engine oil is drained and renewed regularly, as specified by the engine manufacturer.

Pressure Oil Pump. Oil pumps are mounted on or in the engine and are driven by the engine itself, hence the name engine driven pumps (EDP's). Frequently both the delivery (pressure) pump and the scavenge pump are driven by a common drive shaft. A geared type of oil pump (spur gear pump) is shown at Figure 12-5.

FIGURE 12-5
Geared Type Oil Pump



A geared oil pump consists of two deep-toothed gears, one of which is driven by the drive shaft, which are encased in a close fitting pump housing. Oil is carried around either side of the casing in the space between the teeth. The pressure of oil delivered by the pump will depend on the rpm of engine and pump, on the diameter of the pump outlet and on the viscosity of the oil. The capacity of the pump must ensure that, even at low rpm and with hot oil, the required minimum oil pressure is maintained. In order to prevent the oil pressure exceeding the maximum permitted value at high rpm and low oil temperature a pressure regulating relief valve is fitted as shown at Figure 12-2.

27. Scavenge Pumps are normally also of the geared type. The scavenge pump will be larger than its associated delivery pump, since it is required to have a larger capacity, in order to avoid an accumulation of oil in the sump.

28. Apart from the pressure relief valve already mentioned several other types of valves are commonly found in a dry sump system.

29. An anti-surge valve is fitted in order to prevent possible damage to the oil cooler on start up. This would occur when starting from cold with correspondingly high viscosity oil which has accumulated in the sump being cleared by the scavenge pump and forced, in a surge of high pressure, through the oil cooler. When the anti-surge valve senses that the oil pressure exceeds a safe value it simply bypasses the oil cooler until the pressure has stabilised.

30. A thermostatic valve will serve a similar function and is also located between scavenge pump and oil cooler. The thermostatic valve senses temperature rather than pressure, and again routes the oil through or around the cooler as required.

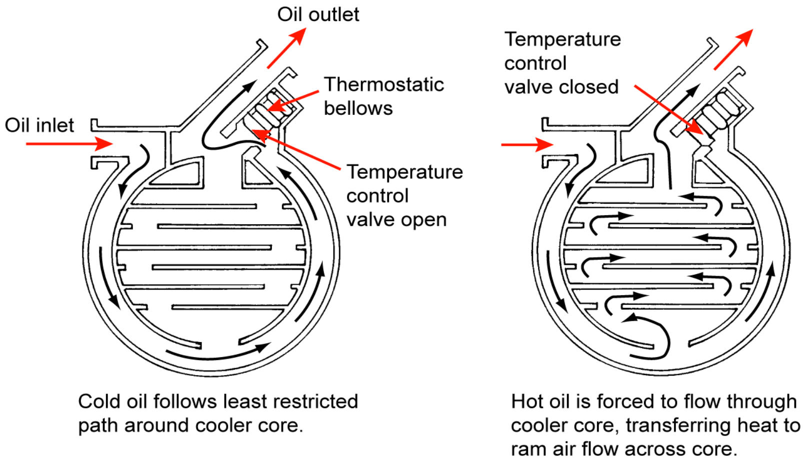
31. A check valve is fitted between oil tank and pump, as shown at Figure 12-2. It prevents seepage of hot oil through the stationary delivery pump and into the engine sump after engine shut down.

Oil Cooler. The oil cooler works on the same principle as a car radiator. A matrix is used with the intention of creating the maximum surface area for minimum volume of unit. The oil cooler unit may or may not have its own pressure relief by-pass valve, depending on the level of protection offered by anti-surge and/or thermostatic relief valves upstream of the cooler.

32. Some oil coolers have radiator shutters which are controlled by the pilot and which govern the amount of air passing over the cooler matrix. With such a system it is necessary to be aware of the possibility of coring occurring due to mishandling. Coring will result from rapid cooling of the oil in the cooler and causes the oil so cooled to congeal within the cooler matrix and to cause a blockage. The cooler will then be by-passed and a rapid rise in oil temperature, with no corresponding increase in cylinder head temperatures, will result. Despite this high oil temperature the correct action is now to close the shutters, in order to heat the congealed oil.

33. In most cases the flow of oil through the cooler is controlled automatically by a thermostatically operated temperature control valve as shown at Figure 12-6.

FIGURE 12-6
Thermostatic Oil Cooler Temperature Control



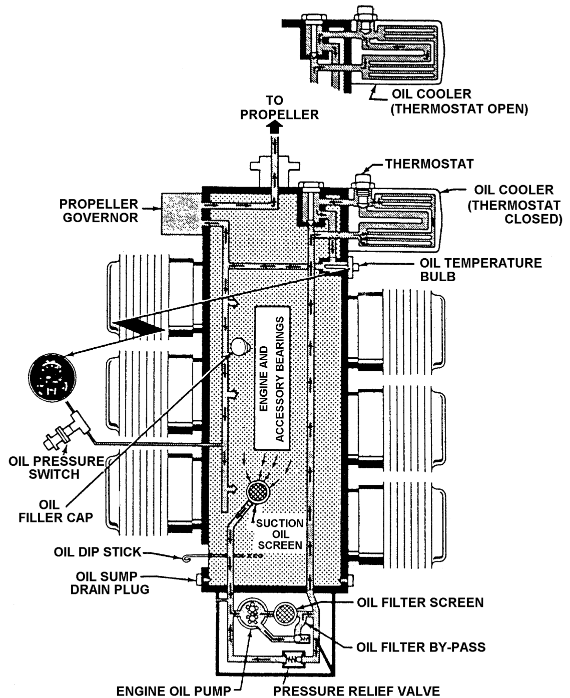
34. When the oil is cold the temperature control valve is open and oil is directed around the core of the cooler, so that it rapidly warms up. As the oil temperature rises the temperature control valve progressively closes, directing more oil through the core of the cooler where it transfers heat to the cooling air passing over the core matrix.

The Wet Sump System

35. In a wet sump system the oil is stored in the engine sump. The problems of feeding oil to many of the engine components are overcome, since the crankshaft rotation takes it through the oil reservoir (the sump), and this causes the oil to splash lubricate the piston walls. It is still necessary to use an oil pump to force oil into other parts of the engine.

36. A wet sump system for an opposed engine is illustrated at Figure 12-7.

FIGURE 12-7
Wet Sump Oil System



37. The wet sump system of lubrication suffers from two principal disadvantages. The first of these is that over-oiling tends to occur at high rpm. Only a very thin film of oil is required to prevent metal to metal contact. With splash lubrication at high rpm a much thicker film of oil is deposited in the cylinder walls, and energy is required to remove the surplus oil.

38. The second disadvantage is that the oil splashed up into the cylinders from the sump is hot and unfiltered.

39. Pressure oil supply to the remainder of the working parts of the engine is filtered, and cooled by passage through an oil cooler.

Engine Cooling

40. If an engine were perfectly efficient all the heat produced in it would be turned into useful work. This is clearly not the case, and in fact only approximately 30% of the heat energy released by combustion is converted into mechanical energy. Approximately 40% passes out with exhaust gases, whilst the remainder heats the engine itself. It is necessary to dissipate the heat that is passed into the engine block, otherwise mechanical damage, and perhaps total failure, will occur.

41. Not only must the mean temperatures of the engine be controlled, but also local hot spots must be avoided, so that distortion due to differential expansion does not occur. It is relatively easy to deal with the cylinders, which are in direct contact with the cooling medium. Pistons, valves and spark plugs require special attention however, since these can only be cooled indirectly. Pistons dissipate their heat by conduction to the cylinder walls, consequently they are made of a highly conductive metal. Inlet valves are cooled by the incoming mixture, and so pose no great problem. Exhaust valves are exposed to the burnt gases and operate at very high temperatures. Sodium filled exhaust valves help to dissipate the heat as previously described.

42. The majority of aircraft engines use an air cooling system. This gives reasonably efficient cooling, and avoids a large and heavy liquid cooling system, which is in any event prone to leakage and other malfunctions.

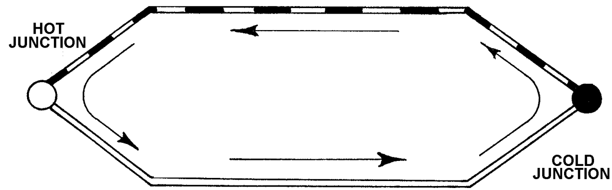
43. To improve the efficiency of the air cooling, heat dissipation is assisted by increasing the external surface area of the cylinder heads and barrels using deep fins. In order to avoid uneven cooling of the cylinders the airflow must be across the engine, rather than along it. In the latter event the cooling effect would be progressively reduced towards the rear of the engine. With in-line engines the cooling airflow is introduced to one side of the engine and directed across the cylinders by means of baffle plates. In opposed layouts the air is introduced into the cowling above the engine and baffles between the cylinders ensure that the airflow passes downward through the cylinder cooling fins to the lower part of the cowling.

44. Cowl flaps are often fitted in the lower cowling. These are opened manually by the pilot in low airflow situations, such as when the aircraft is on the ground, to encourage maximum cooling flow over the cylinders. In flight the ram air pressure is sufficient to ensure adequate cooling flow without the aid of cowl flaps, and they are closed to prevent over-cooling.

45. One of the principal reasons for cooling the cylinders is to avoid excessive cylinder head temperatures, which are a prime cause of detonation and serious engine damage. Monitoring of the effectiveness of air-cooling is therefore by means of the cylinder head temperature (CHT) gauge.

46. Piston engine cylinder head temperature is usually measured by a thermocouple instrument, which produces an electrical output proportional to temperature to operate a circular-scale indicator. Thermocouples convert heat energy into electrical energy at the source of heat. They rely upon what is known, as the Seebeck effect where, if two wires of dissimilar metal are joined at both ends (as shown, in Figure 12-8) an emf will be produced when there is a temperature difference between the two junctions. The greater the temperature difference, the greater the emf.

FIGURE 12-8
Thermocouple Principle



47. The hot junction represents the thermocouple probe, connected to a point on the cylinder head. The cold junction is formed at the temperature indicator, which measures current flow. The greater the temperature difference, the greater the emf produced and therefore the greater the current flow to deflect the CHT gauge pointer.

Piston Engine Ignition and Starting Systems

Starters for Piston Engines

Piston Engine Ignition and Starting Systems

1. The purpose of the engine ignition system is to produce a series of sparks of sufficient intensity and duration to ignite the fuel/air mixture. It is essential that the spark should occur at precisely the right time and that the cylinders are fired in the appropriate sequence.

2. As the piston approaches top dead centre on the compression stroke a high voltage electrical current is applied to the central electrode of a sparking plug situated in the upper part of the cylinder. The high voltage causes a spark to jump across the gap between the sparking plug electrodes to ignite the fuel-air mixture at precisely the right time in the four-stroke cycle. This process must be repeated in each cylinder at the same point in the cycle.

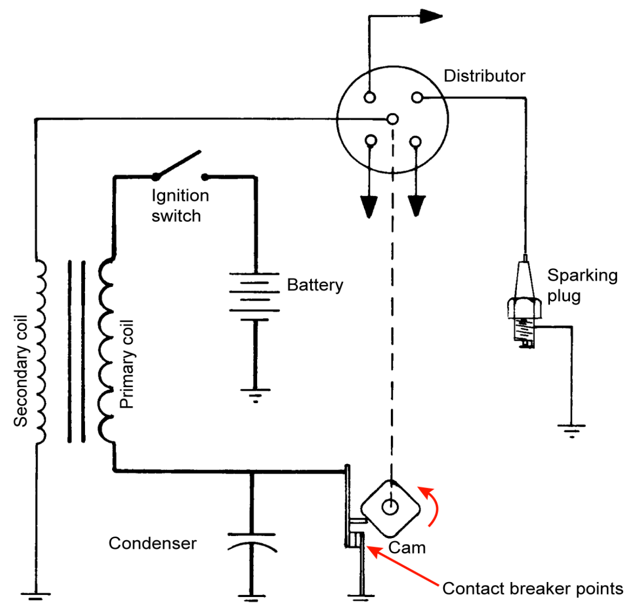
3. To provide a spark of sufficient intensity in the high-pressure environment of the cylinder during the compression stroke a high voltage is necessary, typically in excess of 20,000 volts. This is achieved by a system of electro-magnetic induction and a transformer.

4. To ensure that the spark occurs at precisely the right point in the cycle of operations a mechanically operated device distributes the electrical supply to the sparking plugs at the required instant and in the correct firing order of the engine cylinders. In a four-cylinder opposed engine this firing order is either 1-3-2-4 or 1-4-2-3, depending upon the engine manufacturer. In a six-cylinder opposed engine it is 1-4-5-2-3-6.

Battery Ignition System

5. Automobiles and a few aircraft piston engines use a battery-operated coil ignition system, as illustrated at Figure 13-1.

FIGURE 13-1
Battery Ignition System



6. The battery supplies a low voltage, high current electrical flow through the ignition switch to a primary coil and thence to earth via a cam-operated switch called a contact breaker. The cam is driven through gearing from the engine crankshaft. The high current flow through the low-resistance primary coil creates a strong magnetic field, which surrounds it and the secondary coil. The secondary coil is wound from thin wire and has many more turns than the primary, typically a ratio of at least 100 to 1.

7. As the cam rotates it separates the contact breaker points, thus breaking the primary circuit and instantaneously terminating the primary current flow. This causes the magnetic field surrounding the primary coil to rapidly collapse, inducing as it does so a high emf in the closely-wound turns of the secondary coil. The same rapid collapse of the field through the primary coil also induces a momentary high voltage there also, typically of about 240 volts. Given the ratio of the primary/secondary transformer it follows that the instantaneous voltage induced in the secondary coil will be of the order of 24,000 volts.

8. The secondary coil is connected to an engine-driven distributor, which is mechanically timed to ensure that, at the instant of induced high voltage, there is electrical connection to the sparking plug in the cylinder where the piston is just approaching TDC on the compression stroke.

9. The relatively high voltage induced in the primary circuit at the instant the contact breaker points separate would be liable to cause undesirable arcing at the contact breaker points. To absorb this voltage and prevent arcing a capacitive condenser is fitted in parallel with the contact breaker.

Magneto Ignition System

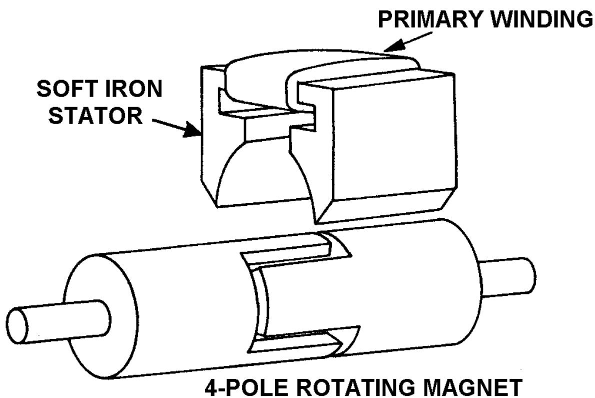
10. This principle of operation is common to most aircraft piston engine ignition systems, except that the source of primary current is an engine-driven a.c. generator called a magneto, rather than a battery. As well as a generator the magneto contains the primary and secondary transformer windings, the contact breaker and distributor, and the condenser.

11. The ignition system of an aero engine is completely duplicated. That is to say the complete system comprises two magnetos, two ignition wiring harnesses connecting the magneto to its sparking plugs, and two sparking plugs per cylinder. Each half of the system will work entirely independently of the other and both will function entirely independent of the other aircraft electrical systems.

12. Duplication of the ignition system has obvious safety advantages, but also improves engine efficiency. This is achieved because the two sparking plugs ignite the compressed fuel/air mixture at two separate points in the cylinder, giving more even, rapid and thorough combustion of the mixture, raising the power developed in the engine.

13. The basic construction of a rotating magnet magneto is shown at Figure 13-2.

FIGURE 13-2
Basic Rotating Magnet Magneto



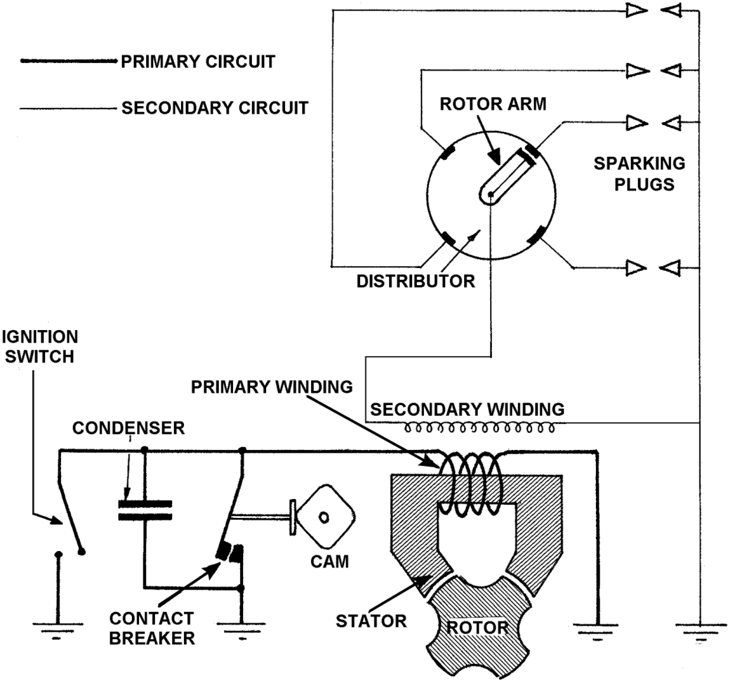
14. The four-pole permanent magnet is engine-driven via an ancillary drive and rotates within the jaws of the soft iron 'U' shaped stator. Wound upon the stator is the primary coil, formed from relatively thick, low-resistance copper wire. As the fields of the rotor magnets cut through the primary coil windings an emf is induced in them by electro-magnetic induction. If the primary coil is now connected to a suitably low-resistance circuit the induced emf will cause a relatively high current to flow through the circuit.

15. Thus, the rotating magnet magneto has replaced the battery in the ignition system previously described. The advantage of the magneto for aircraft use is that it requires no external source of power and is therefore reliable even in the event of battery or electrical system failure.

16. Surrounding the primary windings are the secondary windings (not shown at Figure 13-2). As before, the secondary windings contain many more turns of wire than the primary windings, and therefore the combination of the two acts as a transformer. The function of this transformer is to convert the low voltage/high amperage current flow in the primary circuit into high voltage/low amperage current in the secondary circuit.

17. A magneto ignition system is illustrated at Figure 13-3. The current induced in the primary coil creates a magnetic field that saturates the conductors of the secondary coil. With the ignition switch ON (that is to say open, see Figure 13-3) the only path to earth for the current in the primary circuit is through the points of the contact breaker when they are closed.

FIGURE 13-3
Magneto Ignition System



18. The contact breaker is a mechanically operated switch, opened by an engine-driven rotating cam and closed by spring force. When the contact breaker points open the primary circuit is broken and primary current immediately falls to zero. Consequently, the primary magnetic field collapses, virtually instantaneously, through the secondary winding.

19. This rapid movement of a magnetic flux across the secondary winding induces a very high voltage in the secondary circuit. It is this high voltage which is supplied, by way of the distributor rotor/stator contacts, to the appropriate spark plug.

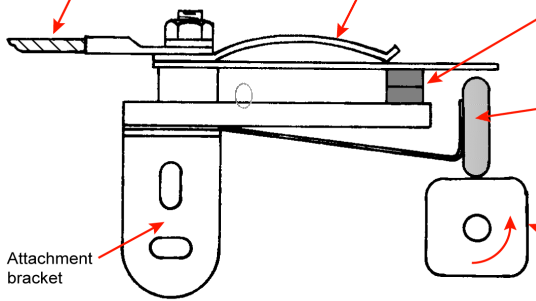
20. The collapsing primary magnetic field also passes across the primary winding, where it induces a significant voltage, sufficient to cause arcing (flashover) at the contact breaker points. To prevent this undesirable arcing, a primary capacitor (condenser) is connected across the contact breaker. The capacitor stores the temporary high primary circuit voltage, preventing the potential difference across the points from becoming high enough to cause arcing. The capacitor discharges its stored voltage once the contact breaker points have fully separated. As this condenser ages, it is likely that its capacitance will drop below its rated value. Consequently it may be unable to absorb the high voltage, arcing will occur, and the working faces of the points will become pitted or eroded.

21. Figure 13-3 shows a simplified piston engine ignition system, or rather half of it, since the system shown would be totally duplicated as previously discussed. Note at Figure 13-3 that the means of switching off a magneto is to earth the live side of the primary winding with the ignition switch. Should the ignition switch not earth the primary circuit on shut down, the magneto will oblige with a perfectly healthy spark should the propeller be turned during ground handling. If a combustible charge has been left (or drawn into) the cylinder, the engine will now quite happily start running, with disastrous consequences for whoever was leaning on the propeller.

22. Contained within the same casing as the magneto is the cam-driven contact breaker and the distributor. The contact breaker cam and the distributor rotor arm are both driven through spur gearing from the magneto rotor shaft.

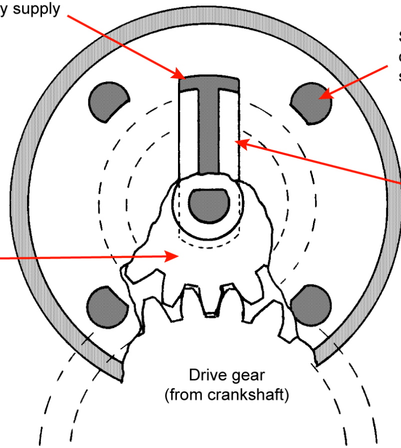
23. A contact breaker is illustrated at Figure 13-4. The upper contact point is supplied with primary current through a conducting flat metal arm. It is held in contact with the lower point by a leaf spring when the cam is in the dwell position as shown, completing the primary circuit to earth through the mounting bracket. A cam follower is held lightly against the rotating cam by a spring so that the cam lobes will force it up against the upper contact point arm and separate the points, breaking the primary circuit. The cam and follower are lubricated by an oil-soaked felt strip, to minimise cam wear. The upper contact point arm is separated from the mounting bracket by an insulator. The contact breaker points are made of heat- and wear-resistant material, such as platinum-iridium alloy. The cam drive gearing is arranged so that the contact breaker points open at the time when the magnetic flux from the rotor magnets, through the primary coil windings, is at a maximum.

FIGURE 13-4
Contact Breaker



24. The distributor consists of a rotating arm, gear-driven from the magneto rotor shaft at a speed that is half engine crankshaft speed. Spaced equidistantly around the stator, or casing, of the distributor are a number of electrical contacts, one for each sparking plug. At the end of the rotor arm is an electrical contact, supplied with voltage from the secondary coil of the magneto transformer. As the end of the rotor arm passes a stator contact the gap between the two is small enough to allow high-voltage current flow from secondary coil to sparking plug. A distributor for a four-cylinder engine is illustrated at Figure 13-5.

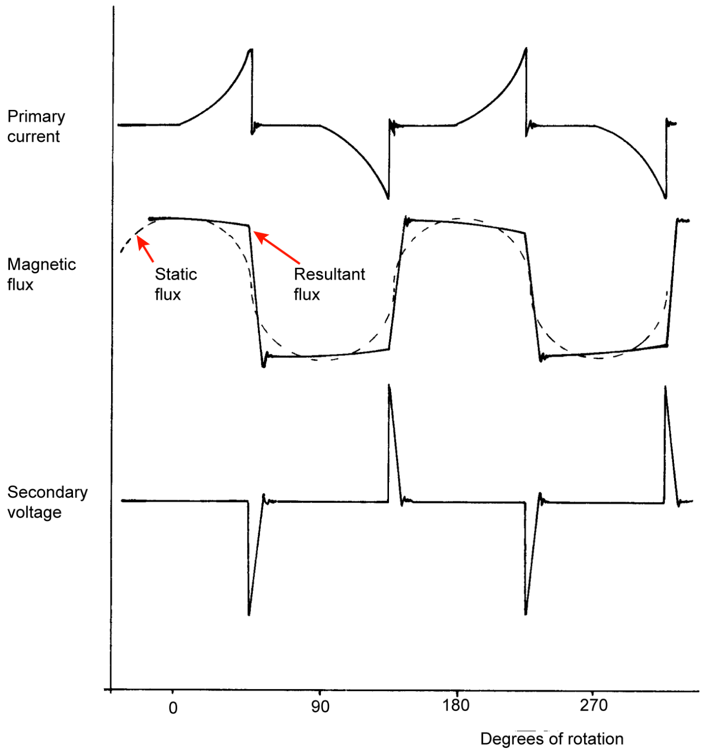
FIGURE 13-5
Distributer for a Four-Cylinder Engine



25. The gear drive between the magneto rotor and the distributor is arranged to ensure that the contacts are coincident when secondary voltage is at peak value. This, and the timing of the contact breaker point opening, is known as the internal timing of the magneto. Ignition timing, that is ensuring that the spark occurs at the right instant in the engine operating cycle, is achieved by adjustment of the geared drive between crankshaft and magneto.

26. The correct internal timing of the magneto is essential in order to ensure the maximum possible induced voltage in the secondary coil. The internal timing is illustrated graphically at Figure 13-6.

FIGURE 13-6
Graphical Internal
Magneto Timing



27. As the rotor magnet of the magneto turns the magnetic flux concentrated in the soft iron core of the primary coil increases to a maximum of one polarity, then decreases to zero before increasing again to a maximum, but of the opposite polarity. This moving magnetic flux field induces an emf in the primary coil. Whilst the magnetic flux is decreasing from maximum value, the contact breaker points are closed and consequently a primary current is flowing to earth. The current flow produces a magnetic field that opposes the changing flux of the field produced by the rotating magnet, thus sustaining high flux strength in the primary coil core, known as the resultant flux.

28. It is arranged that the contact breaker points open during this period when the high core flux strength is sustained by the primary current flow. The immediate cessation of current flow, shown in the upper graph at Figure 13-6, then ensures a rapid change of flux from high of one polarity to high of the opposite polarity. This is illustrated in the second graph at Figure 13-6.

29. The rapid change of flux through the windings of the secondary coil induces the high voltage needed for the ignition spark, as shown in the third graph at Figure 13-6.

30. The number of degrees of magneto rotation between the point when the flux in the core due to the rotating magnet (the static flux) is at a steady maximum, and the opening of the breaker points, is known as the E-gap.

31. Magnetos required to operate at high altitude are pressurised with air to prevent arcing, or flashover within the distributor. At altitude atmospheric air is less dense and this allows high voltage electricity to jump across the gap between rotor arm and electrode more easily than in denser air. Pressurised magnetos are used on many turbo-charged engines and are usually recognisable because they are painted grey or dark blue (as opposed to black).

32. The self-sufficiency of the magneto ignition system, requiring no external electrical supply to keep the engine operating, is the main reason for its use with aircraft piston engines. It suffers, however, from a serious disadvantage when it comes to starting the engine.

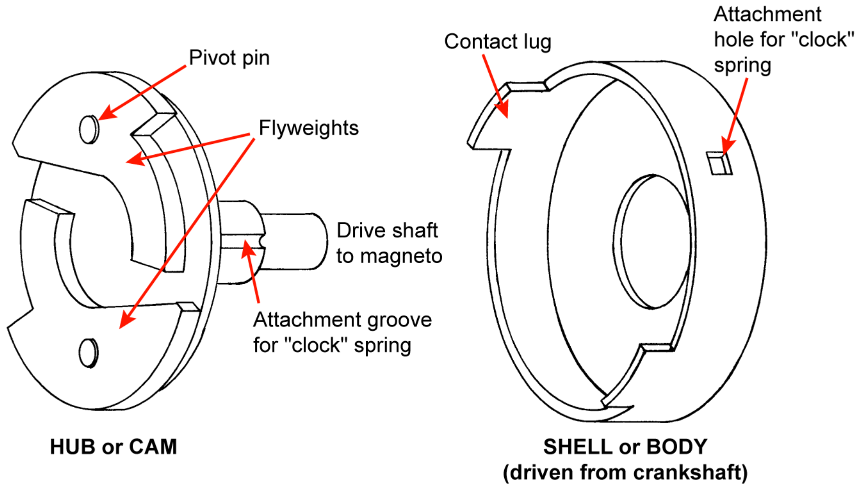
33. The magneto of a four-cylinder engine rotates at half crankshaft speed. At the very low crankshaft rpm when the starting motor is turning the engine, the magneto is turning too slowly to generate sufficient current in the primary circuit, consequently the primary flux field is very weak. Thus, when the contact breaker points open the collapse of this weak field cannot induce a sufficiently high voltage in the secondary coil to cause a spark at the sparking plugs. With no spark there is no ignition and so the engine will not start. Clearly it is necessary to introduce some device that will boost the magneto output until such time as the engine starts and magneto rotary speed is adequate.

The Impulse Coupling

34. A spring-loaded clutch is located between the drive spindle and the magneto shaft. As the engine is turned slowly by the starter motor the spring-loaded clutch is wound up and then released once every half-revolution. During the release phase the magneto rotor is briefly turning very rapidly and releasing a sufficiently energetic spark to ignite the mixture. Once the engine has started, the spring-loaded clutch is disengaged by centrifugal force and drive is direct from engine drive shaft to magneto rotor shaft.

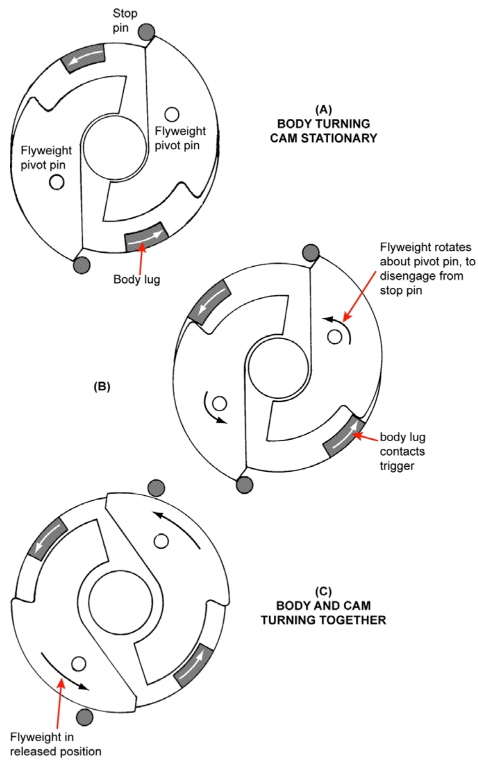
35. Figure 13-7 is a diagram of an impulse coupling that has been separated to show the shell, or body, which is driven from the engine drive shaft. This is in turn connected to the hub, or cam, which is connected to the magneto rotor, by a strong spring.

FIGURE 13-7
Impulse Coupling



36. Figure 13-8 shows the action of the flyweights mounted on the hub of the impulse coupling. In Figure 13-8a the engine has started to rotate under the influence of the starter motor. The body of the coupling is turning, but the hub and therefore the magneto is held stationary by a stop pin on the casing of the magneto. Consequently the spring between the two is being wound up, storing mechanical energy.

FIGURE 13-8
Impulse Magneto Operation



37. In Figure 13-8b the body has rotated to a point where a lug attached to it contacts a trigger ramp on one of the flyweights. This causes the flyweight to rotate about its pivot point, disengaging it from the stop pin. The wound spring now rotates the coupling hub, and magneto rotor at high enough speed to generate sufficient secondary voltage for a spark at the plug. This procedure will be repeated until the engine fires and accelerates.

38. Once the engine is running under its own power the centrifugal action on the heavy flyweight tails will rotate the flyweights about their pivot points so that they are clear of the stop pins and drive from the impulse coupling body to the hub is unimpeded. This is illustrated at Figure 13-8c.

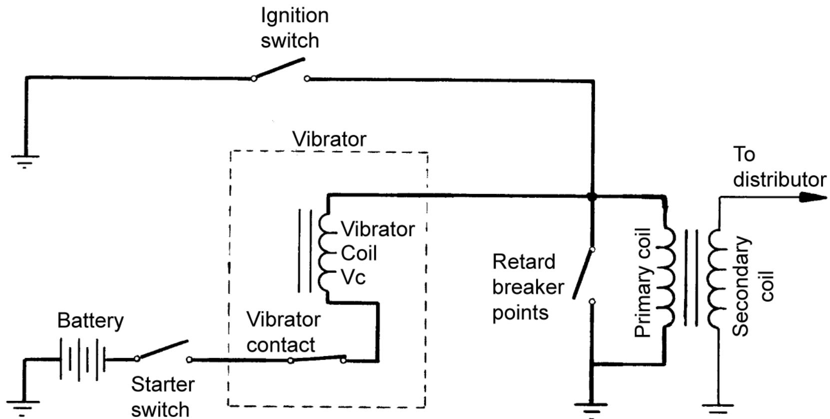
39. It will be appreciated that, during engine start, the engine crankshaft turns through a number of degrees whilst the magneto is held stationary by the impulse coupling. This has the effect of delaying the spark beyond its normal point in the cycle and thus retarding the ignition. This has the desirable effect of ensuring that when the engine does fire it will immediately begin turning in the correct direction of rotation. Once the engine starts and drive to the magneto is uninterrupted the ignition timing returns to its normal advanced position.

40. Usually only one of the two magnetos is fitted with an impulse coupling.

Induction Vibrator

41. Some magneto ignition systems employ an induction vibrator instead of an impulse coupling to ensure a sufficiently high secondary voltage for starting. The device supplies pulsating direct current to the magneto primary coil whilst the starter switch is closed. A diagram of an induction vibrator circuit is shown at Figure 13-9.

FIGURE 13-9
Induction Vibrator Circuit



42. When the starter switch is closed battery current is supplied to the vibrator coil VC through the vibrator contact points and through the retard breaker points to earth. The retard breaker points are located in one of the two magnetos. As the vibrator coil is energised the electro-magnetic field opens the vibrator contact points and the vibrator coil is de-energised. The vibrator contact points close again under spring force and the process is repeated many times per second, sending a pulsating current through the primary winding of the magneto.

43. The system thus behaves in the same way as a battery ignition system, inducing high voltage impulses in the magneto secondary circuit for the sparking plugs. The retard breaker points when closed short circuit the supply from vibrator coil to magneto primary coil. They are timed to open at a point later than normal in the engine cycle, thus retarding ignition during starting. Once the engine has started the starter switch will be opened, thus de-energising the induction vibrator.

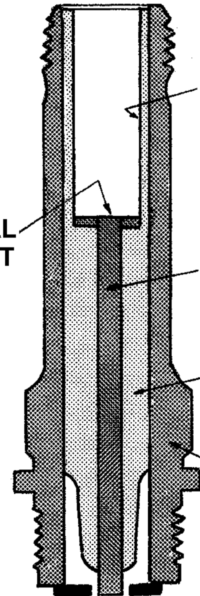
Sparking Plugs

44. A sparking plug comprises three main parts, the metal shell that screws into the cylinder head, a central electrode to which the magneto secondary high voltage is supplied and a ceramic insulator separating the two. Attached to the lower end of the steel shell are one or more electrodes set a small distance from the central electrode, forming the gap over which the spark jumps.

45. Aircraft sparking plugs usually incorporate a metal barrel around the upper part of the plug to provide a shield for the electro-magnetic field they produce, and which would otherwise cause serious radio interference.

46. A typical aircraft sparking plug is illustrated at Figure 13-10.

FIGURE 13-10
Sparking Plug



Magneto Drop Check

47. The correct functioning of the duplicated magneto ignition system is checked by means of a magneto drop check. This is a check that each magneto ignition system functions correctly when its ignition switch is open (in the ON position) and that it ceases to function when its ignition switch is closed (in the OFF position). With the engine running at the rpm recommended in the operating manual, the magnetos are individually switched off and the drop in rpm with the cylinders firing on one plug only (each) is noted. This is a check both of the health of the individual magneto ignition systems and that the ignition switches are not earthed. The rpm drop observed on a magneto drop check should typically be between 50 and 125 rpm.

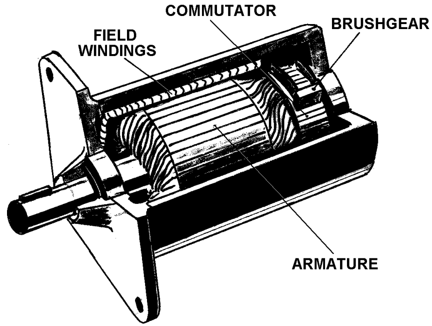
Starters for Piston Engines

48. Aircraft piston engine starters normally employ a direct drive (direct-cranking) electric motor. This drives the engine through a starter pinion gear that engages with a large diameter starter gear ring attached to one end of the engine crankshaft. The starter motor is only engaged with the engine during starting. When battery power is disconnected from the starter motor it disengages from the engine.

49. Engagement of the starter drive is usually automatic once battery power is applied to the motor (by operating the starter switch), although in some light aircraft it may be engaged manually. As soon as the engine has started the starter drive is automatically disengaged to prevent the engine driving the starter motor, which would lead to overheating of the motor windings.

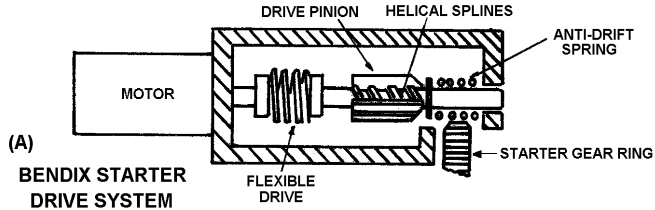
50. Figure 13-11 shows a typical direct-cranking starter motor.

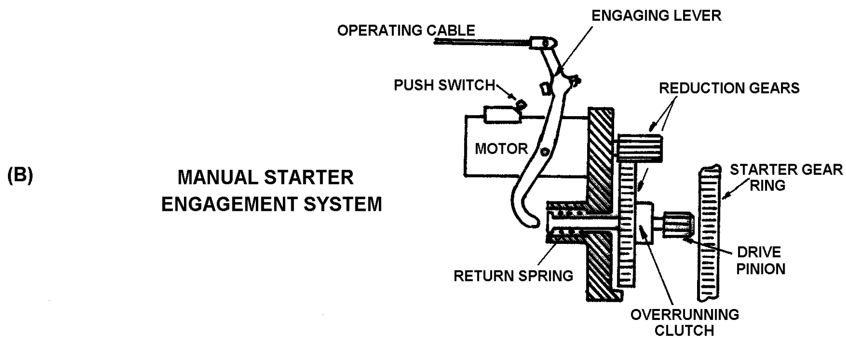
FIGURE 13-11
Direct-Cranking Starter Motor



51. Probably the commonest method of automatic starter engagement is the Bendix drive, illustrated at Figure 13-12. The motor armature shaft is keyed or splined to a flexible drive, which is in turn keyed to the drive shaft of the engaging gear. The drive pinion is threaded onto helical splines on the drive shaft, and is normally held out of engagement with the starter gear ring by the anti-drift spring.

FIGURE 13-12
Bendix Dive Starter





52. When battery power is applied to the starter motor the armature rotates the drive shaft. Because of its inertia, the drive pinion tends to rotate more slowly and so it moves axially along the helical splines on the drive shaft (like a nut on a bolt) to engage with the starter gear ring and turn the engine.

53. When the engine starts, the starter gear ring turns the pinion faster than the starter drive shaft. This forces the pinion back along the helical splines and out of engagement with the starter ring, assisted by the anti-drift spring. When battery power is cut off, by releasing the starter switch, the anti-drift spring holds the pinion disengaged from the starter gear ring.

54. In service the engagement of the rotating pinion with the stationary gear ring, each time the engine is started, leads to wear and damage of the gear ring teeth. This, in turn, can lead to failure of the pinion to disengage on start-up. When this occurs an amber warning lamp remains lit on the control panel and the engine must be shut down to avoid damage to the starter motor.

Piston Engine Fuel Supply

Piston Engine Fuel Supply

Introduction

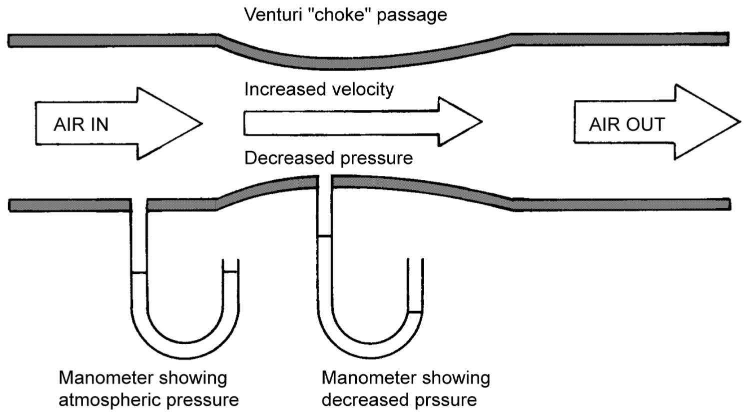
1. The fuel supplied to the engine must be mixed with air so that there is sufficient oxygen to ensure efficient and complete combustion of the fuel for the maximum release of heat energy. The quantity (mass) of fuel supplied must also be exactly that required for the engine power or rpm demanded by the pilot. The devices that achieve this exact metering of the fuel supply are the carburettor or the fuel injection system.

The Carburettor

2. The function of the carburettor is to meter a fuel/air mixture to the engine at the correct mixture ratio and in the quantity required for the specific engine power condition. This process is known as carburetion.

3. In order to achieve the correct fuel/air mixture ratio the carburettor employs a venturi tube, the operation of which is based upon Bernoulli's theorem. The venturi tube is placed in the inlet air passage to the engine and forms a convergent/divergent nozzle, or throat, in the air passage as illustrated at Figure 14-1.

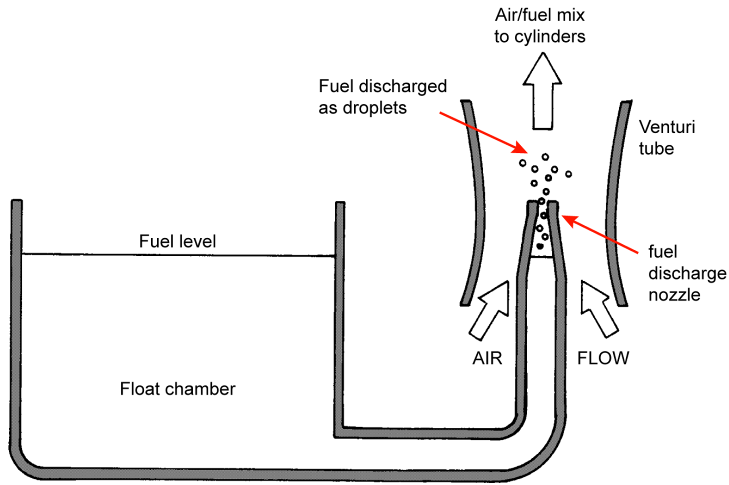
FIGURE 14-1
Venturi Principle



4. The quantity or mass of air flowing into the tube is the same as that flowing out of it since the entry and exit passages are the same diameter. Therefore, the velocity of the air as it passes through the venturi must increase in order to maintain constant mass flow. Bernoulli's theorem states that the total energy of a fluid in motion is constant at all points, so an increase in velocity must be matched by a decrease in pressure.

5. The pressure drop at the throat of the venturi is proportional to the air mass flow through it and this pressure drop is used in the carburettor to meter a proportionate mass of fuel to mix with the air at the correct fuel/air ratio. The principle is illustrated at Figure 14-2.

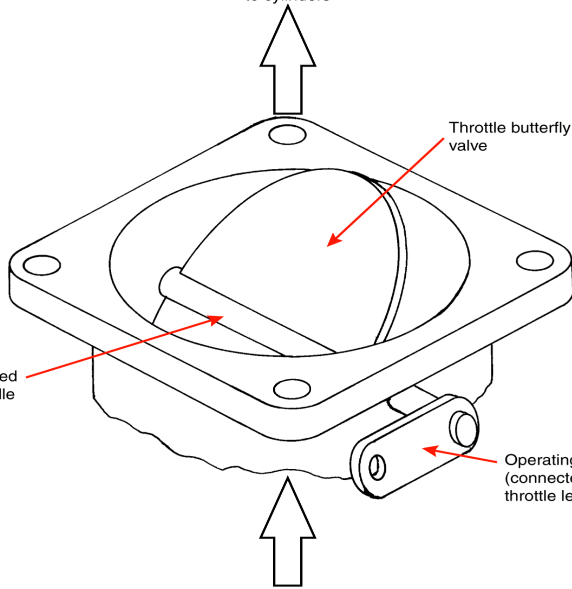
FIGURE 14-2
Carburretor
Venturi



6. A fuel chamber is connected to a nozzle placed in the airflow at the venturi throat, where there is a pressure reduction, or depression. The fuel in the chamber is subject to atmospheric pressure and thus there is a positive pressure difference between chamber and nozzle. This will force fuel to flow from the chamber, through the nozzle and into the airflow in the venturi to mix with the air. The quantity or mass of fuel that flows will be in proportion to the pressure difference between chamber and nozzle, which is in turn proportional to the mass of airflow through the venturi.

7. Air is induced to flow through the carburettor venturi into the engine cylinders by the action of the pistons during their induction stroke, as previously described. The amount of air drawn in is controlled by a throttle valve placed between the venturi and the cylinders, in the inlet manifold. The type of valve used is usually a butterfly valve as shown at Figure 14-3.

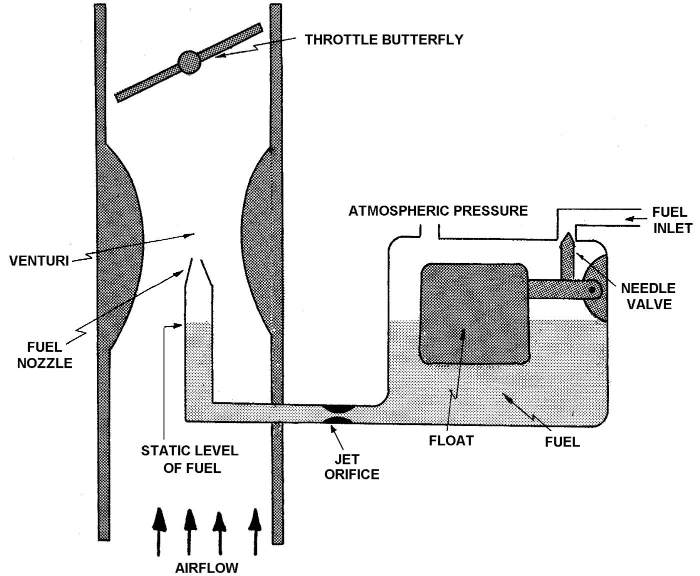
FIGURE 14-3
Carburretor
Butterfly Valve



8. The butterfly valve is an oval metal disc placed in the inlet manifold and pivoted on a central spindle that is perpendicular to the axis of the manifold. The periphery of the disc is shaped to form a close fit with the walls of the manifold when the disc is rotated to an angle of about 70° to the axis of the manifold. This is the closed position of the valve. When the disc is rotated to a position parallel to the axis of the manifold bore it offers little restriction to airflow, because of its thinness. This is the fully open position of the valve.

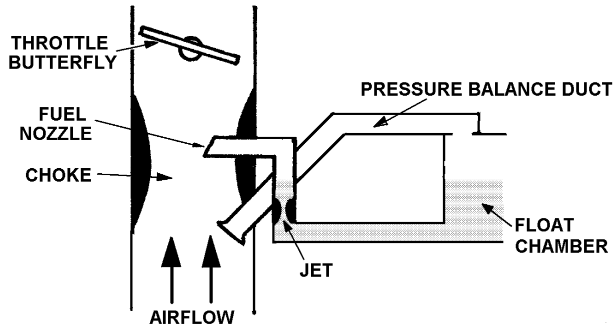
9. Figure 14-4 shows a basic carburettor system. The fuel chamber contains a float-operated inlet valve, which controls fuel flow into the chamber proportional to fuel flow out through the nozzle. Airflow to the engine is controlled by a throttle butterfly valve. The fuel chamber is connected to atmospheric pressure and a fixed orifice, or jet, is placed between fuel chamber and nozzle to limit the maximum fuel flow rate. The fuel flow from the nozzle will be proportional to the pressure difference across the jet orifice, which we have already seen is proportional to the airflow through the venturi. This is known as a float-chamber carburettor and is the basis of many aircraft engine carburettors.

FIGURE 14-4
Float Chamber Carburretor



10. In order to prevent changes in the rate of fuel discharge from the nozzle due to changes in atmospheric pressure at the air intake, air pressure is frequently introduced to the float chamber via a duct from the intake itself. In this way any change in pressure at the air intake will equally affect the float chamber, and the mixture ratio should not change due to intake pressure fluctuations. A carburettor fitted with a pressure balance duct is illustrated at Figure 14-5.

FIGURE 14-5
Pressure Balance Duct



11. The system described is simple, but has some serious limitations. The main problem is the inability of the system to deliver a fuel/air mixture at the required ratio under all conditions of engine speed. An increase in the engine rpm will cause an increase in airflow through the venturi. This will increase the pressure differential between float chamber and venturi, and will cause more fuel to enter the airflow as described in the foregoing paragraphs.

12. However, the flow of a fluid through an orifice is not directly proportional to the pressure drop across the orifice, but to the square root of the pressure drop. Consequently as airflow through the venturi increases with increased engine rpm the pressure drop across the jet orifice increases and fuel flow to the engine increases as the square of that pressure drop. Hence the fuel/air mixture would become unacceptably rich at high rpm and/or unacceptably lean at low rpm.

Mixture Ratio Control

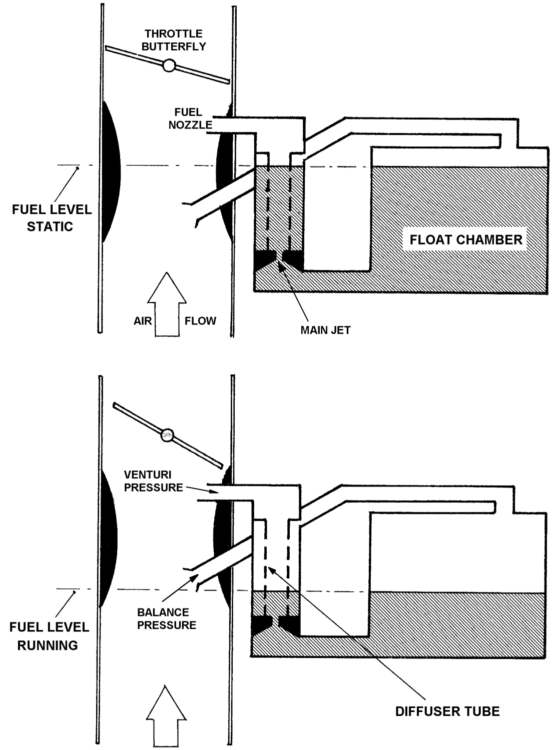
13. To ensure that the fuel mass flow increases linearly with the air mass flow, maintaining a constant fuel/air ratio over the whole speed range of the engine, some form of fuel flow adjustment is necessary. One such device is the diffuser tube, fitted in many float-chamber carburettors.

14. The diffuser tube relies for its operation upon the effect of the float-operated fuel inlet valve in the float chamber. When the engine is stopped and no fuel is discharging from the fuel nozzle the fuel level in the float chamber will rise until the inlet needle valve is closed. This is known as the static fuel level. When the engine is running the pressure differential created across the jet orifice, or main jet, will ensure that fuel flows from the nozzle. The fuel level in the float chamber will fall, as fuel is drawn off, and the float will fall with it, opening the needle valve. When the needle valve is open just sufficiently so that the fuel flow in equals the fuel flow out the float chamber fuel level will stabilise, but at a lower level in the chamber than when the engine was stationary.

15. The higher the engine rpm the greater the pressure difference across the main jet and the greater the fuel flow from the float chamber. Hence, the further the float must fall before the needle valve is open sufficiently to stabilise the fuel level in the float chamber. From this it can be seen that the float chamber fuel level is proportional to engine speed.

16. The diffuser tube is fitted immediately downstream of the main fuel metering jet as shown at Figure 14-6.

FIGURE 14-6
Diffuser Tube



17. Air from the pressure balance duct is bled through the diffuser, which is basically a perforated tube within the main fuel passage. The perforations occur below the static fuel level, so that they are completely covered at low rpm. As the rpm increases the fuel level in the diffuser drops, uncovering the upper perforations in the diffuser tube. This allows a bleed of air at intake pressure to enter the tube and raise the pressure to a value slightly above venturi depression pressure. Thus, the pressure differential across the main jet is decreased slightly, reducing the fuel flow through the jet slightly.

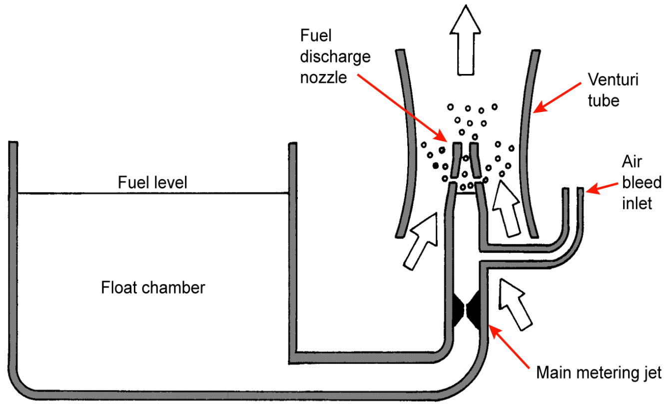
18. The higher the engine rpm the more float chamber fuel level falls, uncovering more of the diffuser tube perforations and allowing more air at intake pressure into the tube. The increase of pressure differential across the main jet is thereby limited with increasing rpm.

19. The sizes of the perforations in the diffuser tube are carefully graduated to ensure that the pressure drop across the main jet, and therefore the fuel flow through it, is matched to the airflow through the venturi at a constant ratio over the whole speed range of the engine. To put it another way, the diffuser tube adjusts the pressure difference across the main jet as airflow through the venturi increases, to ensure that the fuel flow increase is in direct proportion to the airflow increase, thus maintaining a constant fuel/air ratio at all engine speeds.

20. The diffuser has a second advantage in that it aids in the atomisation of the fuel. Because the diffuser, by its action, introduces air into the fuel, the fuel will break up into tiny droplets - an emulsion of air and fuel. Since such a mixture or emulsion is less dense than liquid fuel, it can be drawn to the lip of the discharge nozzle that much more readily. Also, the emulsion will have a larger surface area than liquid fuel, so it will evaporate more readily.

21. In carburettors where a diffuser tube is not used an air bleed into the fuel feed from main jet to nozzle is introduced to create such an emulsion. This is illustrated at Figure 14-7.

FIGURE 14-7
Air Bleed
Emulsifier

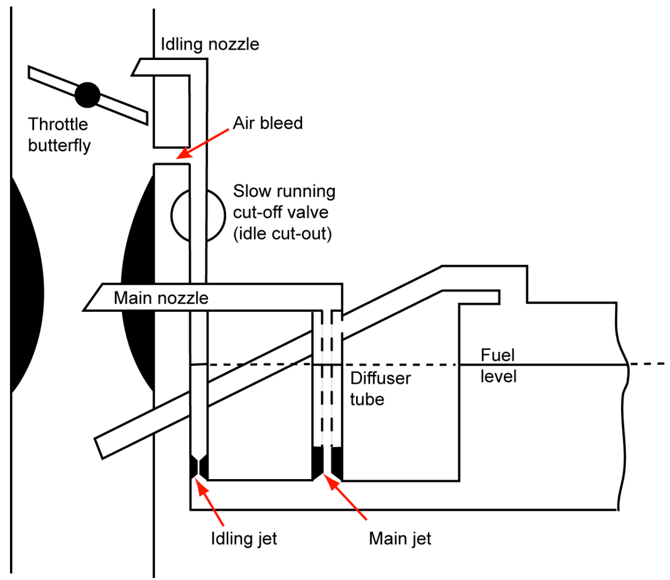


Slow Running Control

22. A slow running, or idling jet is necessary to ensure smooth running at low rpm. Despite the best efforts of the diffuser to overcome the problem of weak mixtures at low rpm, the problem still exists to some extent. This is due to the fact that, with the throttle butterfly valve almost totally closed, the pressure drop through the venturi is too small to draw sufficient fuel through the main jet and the engine is liable to stop or run unevenly. To remedy this situation a slow-running jet is positioned as shown at Figure 14-8.

23. The slow running system works because, with the throttle butterfly valve almost closed, a considerable venturi effect occurs around the butterfly valve itself. Fuel from the slow-running jet is therefore introduced just downstream of the throttle valve. The air that is bled from just upstream of the butterfly valve helps to atomise the fuel.

FIGURE 14-8
Slow Running Jet



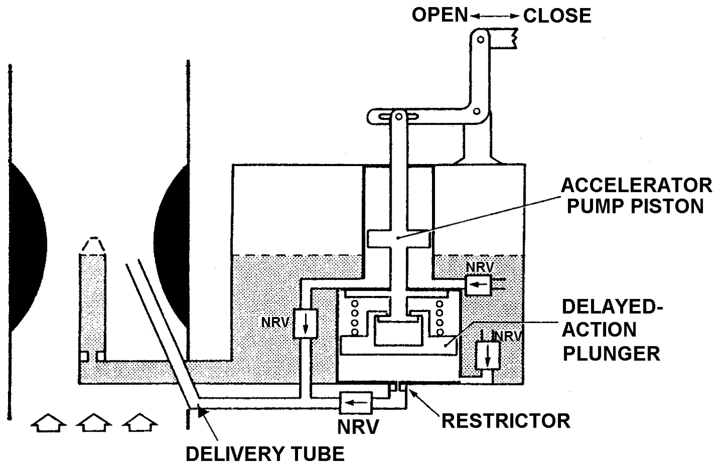
24. It will be noted that a valve is positioned in the fuel passage to the slow running nozzle. This is the slow running, or idle cut-out which is used to shut down the engine, normally by selecting the idle cut-off position on the mixture control lever. By doing this rather than simply switching off the ignition (earthing the magnetos), the possibility of engine over-running is avoided. This would occur if any carbon hot spots within the cylinders caused the mixture to continue igniting even with the spark plugs inactive.

Accelerator Pump

25. It is clearly desirable that the rate at which the throttle valve is opened should produce a similar response, in terms of rate of increase in rpm, from the engine. In other words, slowly opening the throttle results in a slow increase in rpm, whilst rapid throttle opening produces a rapid increase in engine rpm. In the simple carburettor discussed thus far this would not be the case. Opening the throttle valve rapidly would immediately produce a small increase in airflow and subsequently a small increase in fuel flow, which would cause the engine rpm to begin accelerating. This would increase the airflow, and thus the fuel flow, further increasing acceleration. As can be seen, after an initial delay (called a flat spot) the process would be progressive rather than immediate.

26. In order to ensure immediate response to a throttle demand for more power or rpm it is necessary to ensure that there is an immediate increase in fuel flow as well as airflow. This is achieved by the accelerator pump, which is illustrated at Figure 14-9.

FIGURE 14-9
Accelerator Pump



27. The accelerator pump piston operates in a cylinder immersed in the float chamber of the carburettor and is linked directly to the throttle valve operating mechanism. When the throttle lever is moved to open the throttle valve the accelerator pump piston is forced down within its cylinder, discharging fuel through a delivery tube to an accelerator nozzle positioned adjacent to the main fuel nozzle, enabling the engine to accelerate in immediate response to the pilot's demand.

28. The downward movement of the piston also compresses the spring of a delayed action plunger. As the engine responds to the initial action of the accelerator pump piston the compressed spring expands, forcing the delayed action plunger down to deliver a further controlled discharge of fuel to maintain acceleration. The additional fuel supplied during acceleration is thus directly proportional to both the rate and extent of throttle lever movement.

Mixture Control

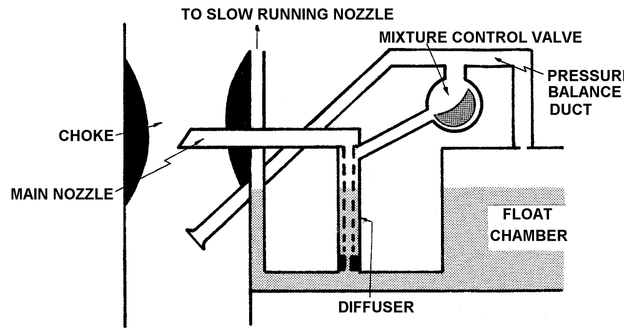
29. Because of the changes in air density with altitude it is necessary to be able to vary the fuel/air mixture ratio for an aircraft piston engine. As the air pressure decreases with altitude its density also decreases and so the normally aspirated (un-supercharged) engine receives less mass of air at each induction stroke and the mixture ratio becomes progressively more rich. Consequently, to maintain the correct mixture ratio the amount of fuel supplied by the carburettor must be reduced as altitude increases. Similarly, the fuel supplied must be increased as air density increases, whether due to increasing pressure in the descent or decreasing temperature at constant pressure, in order to maintain the correct fuel/air ratio.

30. When operating at high power settings it is essential to avoid excessive cylinder head temperatures and the accompanying danger of detonation. The fuel burn temperature can be constrained by selecting a higher-than-normal (i.e. rich) fuel/air mixture.

31. Mixture control is also required so that fuel economy can be attained at low power settings, where high cylinder temperature and detonation is not a hazard, by selecting a slightly lean mixture.

32. In all cases mixture control is by adjustment of the quantity of fuel passing through the main jet whilst the airflow remains unchanged. This may be achieved by altering the pressure differential across the main jet or by restricting the fuel flow to the jet. Figure 14-10 is a diagram of a mixture control system in which the fuel flow through the main jet is varied by altering the pressure above the jet and thus the pressure difference across it.

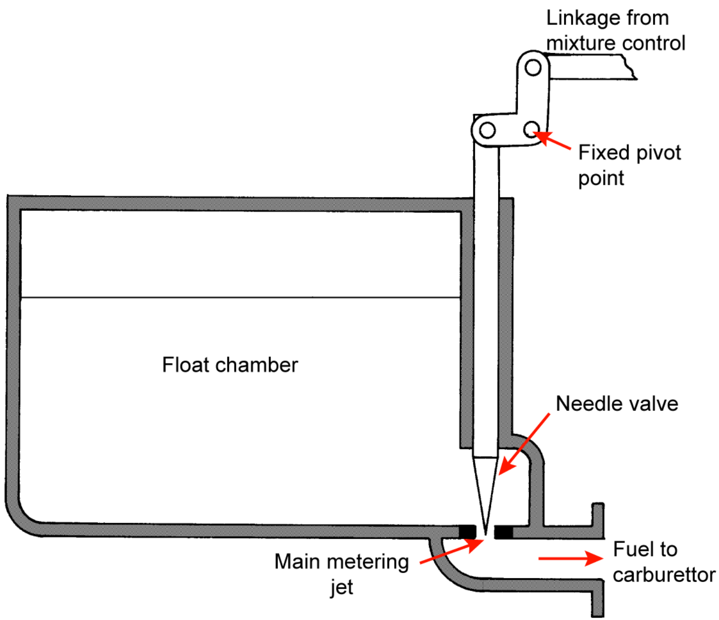
FIGURE 14-10
Manual Mixture Control



33. If the mixture control valve in Figure 14-10 is closed the pressure drop across the main jet is the difference between venturi depression pressure and air intake pressure. This is the fully rich setting where fuel flow will be maximum for a given airflow through the venturi. Moving the mixture control valve toward the open position will progressively reduce the pressure difference across the main jet, reducing fuel flow through it, by leaking some intake pressure into the diffuser tube above the main jet. The fully lean condition is achieved with the mixture control valve fully open.

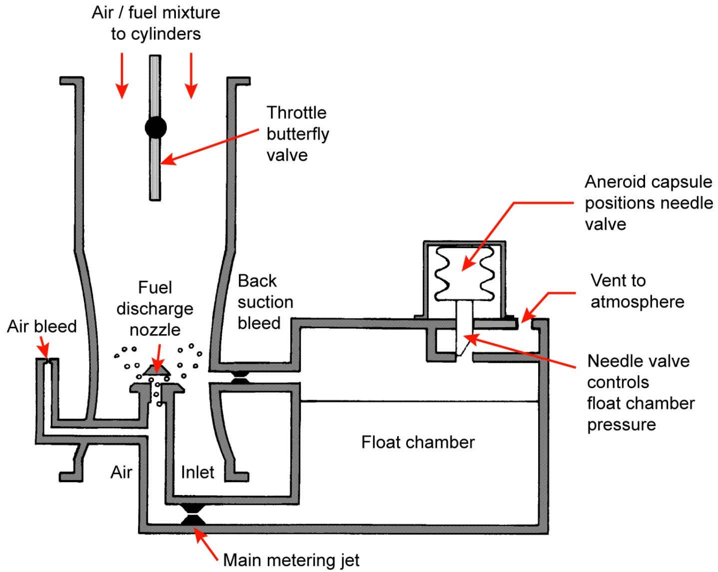
34. An alternative type of mixture control valve uses a needle valve that directly controls the size of the main jet orifice. The needle valve is actuated by a cam, which is controlled from the pilots' mixture lever. Such an arrangement is illustrated at Figure 14-11.

FIGURE 14-11
Needle Valve
Mixture Control



35. Some sophisticated carburettors use an automatic mixture control device that adjusts the fuel/air ratio automatically as ambient atmospheric pressure changes, alleviating the pilot of the need to adjust mixture ratio during climb or descent. Such a system is illustrated at Figure 14-12. A bleed from the venturi is connected to the carburettor float chamber and an aneroid capsule is connected to a valve situated in the float chamber vent to atmosphere.

FIGURE 14-12
Automatic Mixture Control



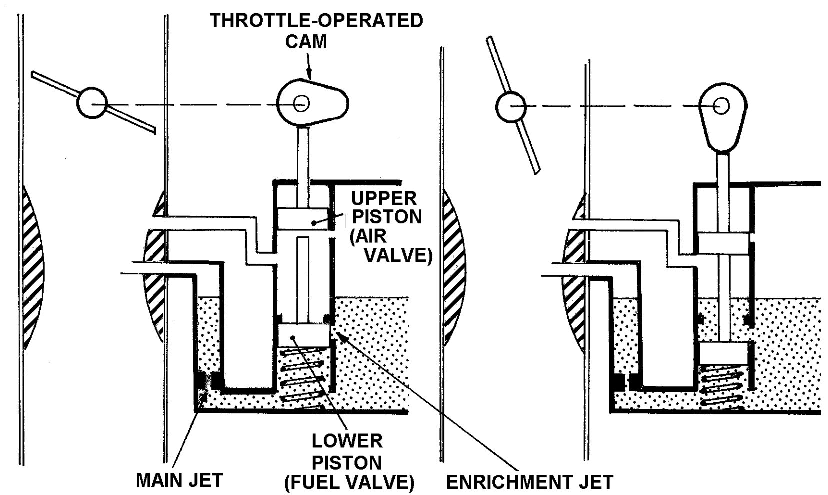
36. At sea level the aneroid capsule is fully compressed and the atmospheric vent valve is fully open, so pressure difference across the main jet is maximum for any given airflow through the venturi. This is the rich mixture setting. As altitude increases and atmospheric pressure decreases the aneroid capsule expands, moving the vent valve towards closed. Meanwhile the suction bleed from the venturi to the float chamber is causing a pressure reduction in the chamber, which the restricted vent cannot equalise. The pressure difference across the main jet, and the fuel flow through it, is reduced thus reducing (leaning) the fuel/air mixture ratio.

Economiser

37. Economisers, or power enrichment systems are often fitted to deliberately enrich the mixture whilst the engine is operating towards the top of its permissible power range. The valve, or valves, are cam-operated and may come into operation when the throttle is advanced beyond a pre-set point, or when the manifold air pressure control mechanism is set to deliver take-off power. Because these systems ensure that enrichment does not occur in the cruise regime, they are sometimes referred to as economiser systems.

38. The enrichment system shown at Figure 14-13 feeds fuel directly into the venturi, adding to the fuel already being discharged from the main nozzle, when the throttle setting approaches the full throttle position. A cam attached to the throttle operating linkage rotates sufficiently at full throttle selection to force down the upper piston, closing off an air connection to the economiser nozzle. As the upper piston moves down further it contacts the shaft of the lower piston and forces that down, uncovering a jet orifice connected to the float chamber. The differential pressure across the jet then forces fuel to the economiser nozzle. Thus the normal nozzle supply to the engine is supplemented, or enriched, at high power settings.

FIGURE 14-13
Economiser



Carburettor Icing

39. Carburettor ice can form with outside air temperatures as high as +30°C and in clear air, given a relative humidity of 30% or more, depending upon air temperature. The most critical free air temperature is thought to be +13°C or so. Icing is more likely to occur at low power settings, such as descent power, when the throttle butterfly is almost closed and creating a significant pressure/temperature drop.

40. It is necessary to protect the carburettor from ice formation by means of some form of carburettor heat control. Ice will form within the carburettor, especially in the region of the throttle butterfly valve and the venturi tube, and will eventually be sufficient to alter the shape of the venturi to such an extent that the zone of low pressure is moved away from the fuel nozzle. In this event the engine stops and probably will not restart until the ice formation is melted.

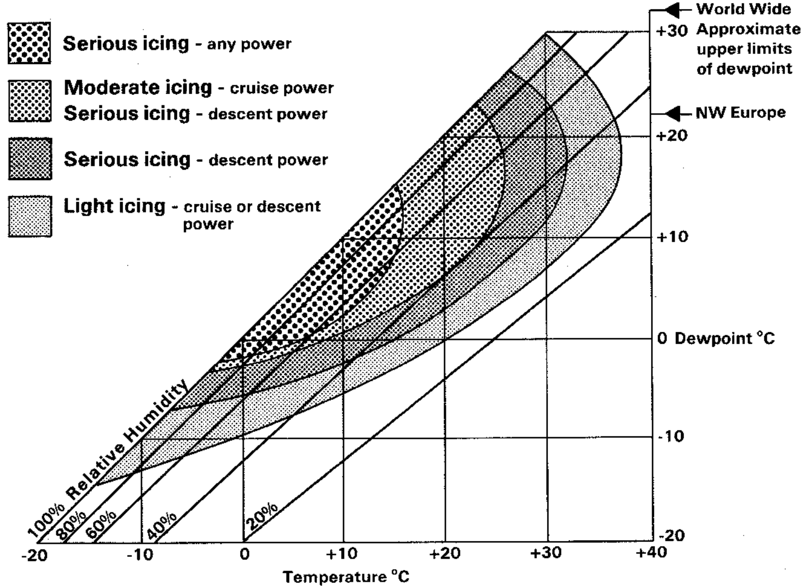
41. The causes of ice formation are adiabatic cooling as the air accelerates through the venturi and around the butterfly valve, with the consequent drop in pressure and cooling as the evaporating fuel absorbs latent heat from both the air and the body of the carburettor.

42. Carburettor heating normally takes the form of an alternative air source, which is routed through a muffler around the exhaust pipe. It is normal for the source of this heating air to be within the engine compartment and for this air to be unfiltered. Both these facts will help to preclude the risk of impact icing affecting the alternate (heated) air supply during use at temperatures of 0°C or below. Impact icing is simply airframe icing forming on the engine air intake ducts and/or filters.

43. Carburettor and/or intake icing symptoms are reduced rpm with fixed pitch propeller systems or reduced manifold pressure with constant speed, variable pitch propeller systems.

44. It should be appreciated that use of hot air will lower the density of the air intake and therefore reduce engine power. Furthermore, use of hot air at ambient temperatures of 0°C and below may in fact compound rather than cure the problem of carburettor icing, by raising the temperature of the incoming air to a critical level of around +13°

FIGURE 14-14
Carburretor Icing Graph

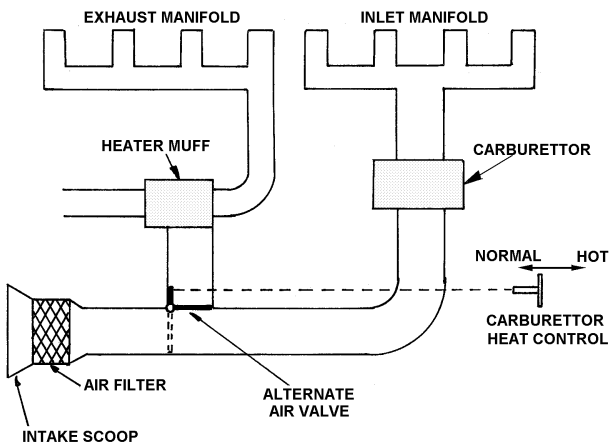


45. The chart shown at Figure 14-14, reproduced from AIC 133/1992 (Pink 68), indicates the wide range of ambient conditions conducive to the formation of induction system icing for a typical light aircraft piston engine. Particular note should be taken of the much greater risk of serious icing with descent power. The closer together the temperature and dew point readings the greater the relative humidity.

Alternate Air

46. The principal elements of a carburettor induction system are shown at Figure 14-15.

FIGURE 14-15
Carburretor Induction System



47. An air filter is situated near the air scoop in order to prevent dust, sand or large foreign objects entering the engine by way of the induction system. The air filter may be of the dry paper type, polyurethane foam or wetted mesh. Dry paper filters are made of pleated layers of porous paper and are similar to those found on most automotive power units. Polyurethane foam filters are a relatively new innovation, using wetted foam as the filtration element. Wetted mesh filters comprise a mat of wire mesh elements wetted with oil, through which the intake air must pass.

48. In some engine induction systems unfiltered ram air is led directly from the intake scoop to the carburettor. When the aircraft is operating on the ground or in dusty/sandy conditions an alternate, filtered air supply is selected by the pilot. In flight the carburettor receives an uninterrupted air supply.

49. For flight in known or forecast icing conditions the pilot is able to select an alternate, heated air supply by operating the alternate air valve, shown in Figure 14-15, to shut off the main air duct, opening the air supply from the exhaust-heated air muff pipe. The alternate air valve is often sprung to the 'normal' position, thus requiring positive action by the pilot to select alternate heated air. This is because the use of heated intake air will reduce charge density/mass and therefore limit maximum available power.

50. The exhaust heater muff is an open-ended pipe surrounding the engine exhaust pipe. Air from inside the engine compartment enters this pipe and, when the alternate air valve is set to HOT, flows around the hot exhaust before passing through the alternate air duct to the main air duct.

Air Intake Blockage

51. Clearly, if the air supply to the carburettor is restricted the mass of the induced charge will be reduced and the engine will be incapable of achieving full power. The more the intake is blocked the less power available until eventually the engine would stop altogether.

52. The effect of a partial blockage of the air intake system will be to reduce the manifold air pressure. The indication to the pilot that this has occurred will depend upon the power plant arrangement of the aircraft.

53. A normally aspirated engine driving a fixed pitch propeller is not usually fitted with a manifold air pressure (MAP) gauge. The loss of engine power associated with the partial blockage will result in a reduction of rpm and this is likely to be the first indication that the pilot has.

54. A normally aspirated or a supercharged engine driving a constant speed propeller must be equipped with a MAP gauge. This is because the first indication to the pilot of intake system blockage will be a reduction of manifold air pressure. Engine/propeller rpm will not change since the propeller governor will adjust propeller pitch to maintain constant rpm.

55. Blockage of the induction system of a stationary piston engine will almost certainly prevent it from starting, since there will be insufficient airflow through the carburettor to initiate fuel flow.

Carburettor Disadvantages

56. The venturi, or choke carburettor has a number of disadvantages, which were appreciated early in the development of aircraft piston engines. In the main, these disadvantages affect the maximum power available from the engine. In summary they are:

- The fuel in the float chamber of the carburettor is subject to gravity and inertia, resulting in variations of the mixture strength during manoeuvres and an inability to function in inverted flight.
- The choke, or venturi tube, has a tendency to icing in certain conditions. This must be countered with heating, which reduces engine power by reducing inlet air density.

- The choke and the protruding fuel nozzles from the carburettor jets impede the flow of air through the intake and therefore reduce the volumetric efficiency of the engine.

57. Consequently, the float chamber type of carburettor has been largely superseded, particularly in high-powered engines, by systems in which fuel is injected under pressure directly into the inlet manifold. In some cases the fuel is directed into the eye of the supercharger impeller, but more often individual nozzles positioned just upstream of each inlet valve inject it into the induction system. In this latter system an even distribution of fuel to each cylinder is achieved.

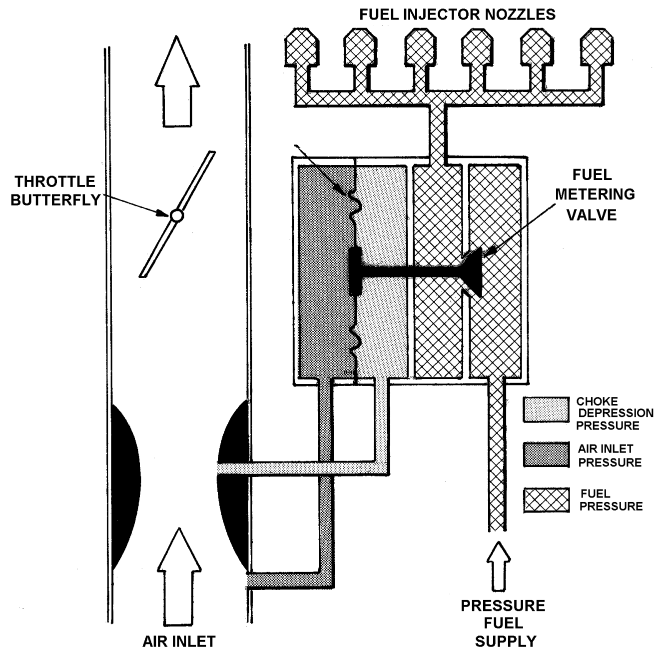
58. Fuel injection may be by means of an injection carburettor, in which the choke is retained as the method of measuring airflow. More commonly however, it is by means of a fuel injection pump in which the maintenance of the correct air/fuel ratio over the whole speed/power range of the engine is achieved without the use of a venturi.

Fuel Injection Systems

Injection Carburettor

59. An injection carburettor retains a choke passage, or venturi in the inlet manifold. However, the depression it creates is used purely as a means of measuring airflow. It is not used to suck fuel from a float chamber through jets, in fact there is no float chamber. The basic principle of operation of an injection carburettor regulator is shown at Figure 14-16.

FIGURE 14-16
Injection
Carburretor



60. The differential pressure between ambient atmospheric pressure and venturi depression pressure flexes a diaphragm to position the fuel metering valve. Fuel is supplied to the valve under pressure from an engine-driven pump, and delivered to injector nozzles positioned at the cylinder inlet valves.

61. Opening the throttle butterfly valve cause the airflow through the venturi to increase, decreasing the venturi depression pressure. The pressure differential across the diaphragm is therefore increased, flexing the diaphragm to open the fuel metering valve. The response to throttle opening is thus the immediate supply of more fuel to the injector nozzles in direct proportion to the increased airflow.

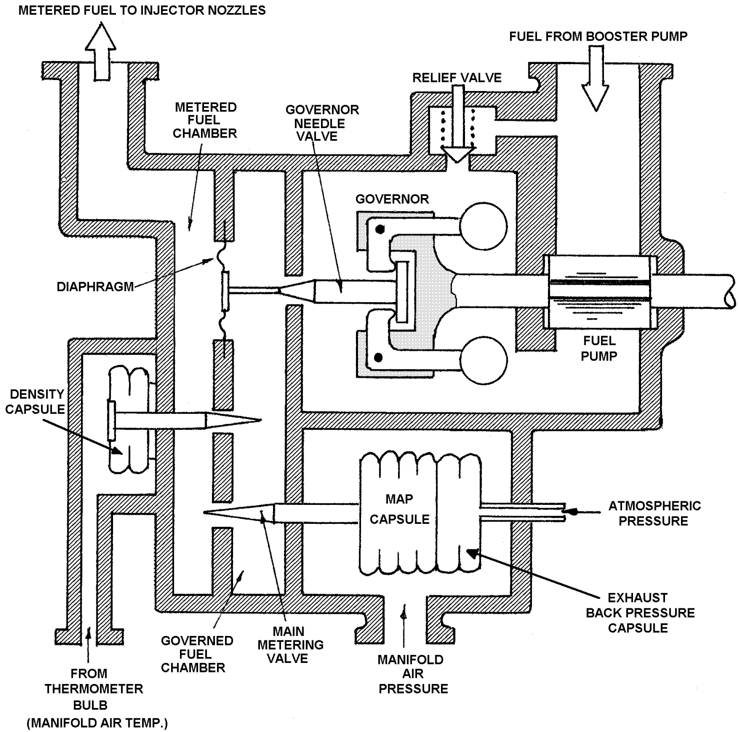
62. From the metering valve fuel flows to injector nozzles, which spray fuel directly into the manifold at the inlet valves. Control of airflow in the inlet manifold is by means of a conventional throttle butterfly valve. These injection carburettors are much more complex than the simplified diagram at Figure 14-16 would suggest. The quantity of fuel supplied, and the mixture strength, is regulated to match manifold air pressure (MAP), altitude and temperature (air density) and power settings.

63. Mixture control is manually adjusted by means of a needle valve, which controls a port between the air inlet pressure chamber and the choke depression pressure chamber. Adjustment of the valve adjusts the pressure differential across the diaphragm and therefore the opening of the fuel-metering valve. Opening the needle valve reduces the pressure differential, leaning the fuel/air mixture, closing it has the opposite effect.

Fuel Injection Pumps

64. The fundamental difference between a fuel injector system and a carburettor system (whether float-chamber or fuel injection carburettor) is that the intake airflow is not measured by a venturi. Instead, fuel is supplied to the cylinders by an engine-driven pump and the quantity is metered to suit engine requirements. Figure 14-17 is a diagram of a fuel injection pump that will meter fuel to match MAP and engine speed conditions.

FIGURE 14-17
Fuel Injection Pump



65. Engine air supply is controlled in the usual manner, by means of a throttle butterfly valve in the intake. The fuel injector pump unit controls fuel supply.

66. A vane type fuel pump pressurises the governor chamber, pressure in this chamber is controlled by the pressure relief valve. The governor consists of flyweights, mounted on the same shaft as the engine-driven pump, which move outwards under the centrifugal force of rotation. In so doing they open a needle valve allowing fuel to flow into the governed fuel chamber and thence, via variable jets, to the metered fuel chamber and the engine.

67. A diaphragm attached to the governor needle valve separates the governed and metered fuel chambers. Thus, the differential pressure between the two chambers acts in opposition to the action of the centrifugal governor and this has the effect of matching fuel flow to engine rpm. The thrust exerted by a centrifugal governor is proportional to the square of the governor rpm and the diaphragm balances that thrust. Consequently the pressure difference across the diaphragm is also proportional to the square of the rpm. The pressure difference across a jet is proportional to the square of the fuel flow through it, so fuel flow past the governor needle valve is matched directly to engine rpm.

68. The flow of fuel through the main jet is controlled by the main metering needle, the position of which compensates for changes in MAP and exhaust back pressure.

69. If MAP increases, the evacuated capsules will be compressed, causing the main metering needle valve to open and increase fuel flow to match the increased MAP. The profile of the needle valve is shaped to ensure that the mixture strength is correctly maintained over the whole range of manifold air pressures from idling to maximum power.

70. As ambient air pressure decreases with altitude the exhaust back pressure decreases and scavenging of the cylinders improves. Consequently volumetric efficiency increases. This means that more of the air available in the manifold is drawn into the cylinders, which tends to weaken the mixture. In a float-chamber carburettor the extra airflow through the venturi compensates for this by drawing more fuel through the jets, but there is no choke with a fuel injector.

71. To overcome this problem the back pressure capsule, attached to the evacuated MAP capsule, is connected internally to atmospheric pressure. Thus, as altitude increases the back pressure capsule is progressively compressed (by the progressively greater difference between MAP and atmospheric pressure) causing it to progressively open the main metering needle valve, maintaining the mixture strength.

72. To compensate for the reduction in density of the inlet charge, which results from an increase in charge temperature, a thermometer bulb in the inlet manifold is used to control the position of a second capsule-controlled metering needle valve. The capsule chamber is connected to the thermometer bulb by a liquid-filled capillary tube. An increased manifold air temperature causes the liquid to expand, compressing the capsule and closing the needle valve to reduce fuel flow, matching the reduced density of the charge air.

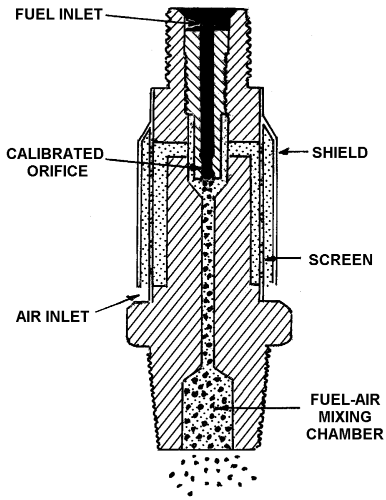
Manual Mixture Control

73. In addition to the automatic mixture control described above, fuel injection pumps normally incorporate some form of manual mixture control to enable the pilot to lean the fuel/air ratio for economic cruising and to enrich it for fuel cooling at high engine power settings. In many cases this manual mixture adjustment takes the form of an adjustment to the size of the main metering orifice, either by adjustment of the main metering valve position or by means of a rotary valve in the orifice. Fuel enrichment may also be by means of an enrichment jet offering a second fuel path to the metered fuel chamber.

Injector Nozzle

74. The fuel injector nozzles are located in the cylinder head such that the fuel discharge is directed into the intake port just upstream of the inlet valve. A typical fuel discharge nozzle is illustrated at Figure 14-18. Fuel is supplied from the injection pump, via a fuel distribution manifold, to a central passage in the nozzle, through a calibrated orifice. At the upper end of the central passage the fuel sprays into a counterbore, which induces a supply of external air through radial holes. Air is carried along the central passage, with the fuel, to mix with the fuel in an outlet chamber before it is sprayed into the cylinder inlet port. This air-fuel mixing within the nozzle assists in the atomisation of the fuel, which in turn assists with vaporisation for better combustion

FIGURE 14-18
Injector Nozzle



Self Assessed Exercise No. 3

QUESTIONS:

QUESTION 1.

The firing order of a four cylinder piston engine is 1-3-4-2. If number one piston is on its compression stroke which stroke is number 3 piston?

QUESTION 2.

During cruising flight the engine oil temperature starts to rise rapidly with all other indications remaining normal. What is the most probable cause and what action should be taken to correct it?

QUESTION 3.

What is the purpose of the finning on air cooled piston engine cylinders?

QUESTION 4.

After engine start up how quickly should normal oil pressure indications appear and what action is required if this is not achieved?

QUESTION 5.

Is an oil of SAE grade 50 more or less viscous than one of SAE grade 20?

QUESTION 6.

What is the difference between a wet sump and a dry sump lubrication system?

QUESTION 7.

Where are oil temperature and pressure usually measured in a piston engine lubrication system?

QUESTION 8.

What is the purpose of the air space at the top of the oil tank in a dry sump lubrication system?

QUESTION 9.

What is the most probable cause of blue smoke issuing from the exhaust of a piston engine?

QUESTION 10.

What is the most probable cause of wildly fluctuating oil pressure in a piston engine?

QUESTION 11.

What is the purpose of the scavenge pumps in a dry sump lubrication system and why are they of larger capacity than the pressure pumps?

QUESTION 12.

Define the term coring as applied to aero-engine oil coolers?

QUESTION 13.

Describe the form of an opposed piston engine?

QUESTION 14.

What is the primary reason for compressing the mixture during the compression stroke of a piston engine?

QUESTION 15.

What is the equation for the brake horsepower (BHP) of a piston engine?

QUESTION 16.

Define the term Mechanical Efficiency as applied to piston engines?

QUESTION 17.

What is the relationship between crank throw and piston stroke?

QUESTION 18.

Define the term volumetric efficiency as used in relation to piston engines?

QUESTION 19.

What causes the inlet and exhaust valves to close in a piston engine?

QUESTION 20.

Define the compression ratio of a piston engine in terms of clearance volume, swept volume and total volume?

QUESTION 21.

How and why will increasing cylinder head temperature affect volumetric efficiency?

QUESTION 22.

In what ways do the pressure and temperature of the gas in a piston engine cylinder vary during the latter half of the power stroke?

QUESTION 23.

How does increasing engine RPM affect volumetric efficiency in a piston engine?

QUESTION 24.

Define the term valve overlap as used in a piston engine?

QUESTION 25.

How is the impulse coupling in a piston engine magneto ignition system de-activated after engine starting?

QUESTION 26.

How is the degree of valve overlap in a piston engine expressed?

QUESTION 27.

Where would you expect to find the oil control rings in a piston engine?

QUESTION 28.

How will the temperature, pressure, volume and mass of the mixture in the cylinder change during the compression stroke?

QUESTION 29.

An engine has compression ratio of 5:1. If the total volume is 1500cc, what will the swept volume be?

QUESTION 30.

Define the term specific fuel consumption as applied to a piston engine?

QUESTION 31.

What term is used to refer to the fuel:air ratio at which a piston engine produces maximum power for a given RPM?

QUESTION 32.

What is hydraulicing in a piston engine and what is the most common cause of it?

QUESTION 33.

How is the drive from the starter motor to the engine engaged and disengaged in the Bendix starter system?

QUESTION 34.

In a piston engine ignition system is a spark which occurs at 20 degrees before TDC said to be more or less advanced than one occurring at 5 degrees before TDC?

QUESTION 35.

What effect does engine RPM have upon the timing and degree of valve overlap?

QUESTION 36.

In the practical four stroke cycle why is it necessary to open the inlet valves before TDC and close them after BDC and to open the exhaust valves before BDC and close them after TDC?

QUESTION 37.

What aspect of the motion of the piston engine permits valve lead and lag without significant backflow of inlet and exhaust gasses?

QUESTION 38.

What is the principal difference between the two stroke and four stroke piston engine working cycles?

QUESTION 39.

What effect does increasing engine RPM have on the point in the rotation of the engine at which maximum cylinder pressure occurs in a four stroke engine cylinder?

QUESTION 40.

How is cylinder head temperature measured in a modern piston aero-engine and what source of electrical power is used?

QUESTION 41.

What source of energy is employed to power the magneto ignition system in a piston aero-engine, what problem does this cause during engine starting and how is this problem overcome?

QUESTION 42.

Why does engine RPM reduce when conducting a magneto drop check?

QUESTION 43.

What is indicated if the engine stops when one ignition system is switched off during the magneto drop check?

QUESTION 44.

What is the purpose of the induction vibrator circuit in a magnetic ignition system?

QUESTION 45.

Describe two methods that may be employed to provide manual mixture control in float chamber carburettors?

QUESTION 46.

Why are some magnetos pressurised and how is this indicated?

QUESTION 47.

Where in the carburettor should the lowest air pressure occur when the engine is running at idle RPM? What problem does this cause and how is this overcome?

ANSWERS:

ANSWER 1.

Number 3 piston is on its induction stoke

ANSWER 2.

The most probable cause is coring of the oil cooler. The correct action to take is to close the cooler cowls for a short period before reopening them slowly.

ANSWER 3.

The purpose of the finning is to increase the surface area in contact with air flow in order to maximise cooling .

ANSWER 4.

Normal pressure indications should appear within 30 seconds of start-up. The engine must be shut down immediately if this is not so.

ANSWER 5.

An SAE grade 50 oils more viscous than an SAE 20 oil

ANSWER 6.

In wet sump systems the oil is stored in the sump. In the dry sump system it is stored in a separate oil tank leaving the sump substantially dry.

ANSWER 7.

Oil temperature is usually measured downstream of the oil cooler. Oil pressure is usually measured downstream of the oil pressure filter immediately before the areas to be lubricated.

ANSWER 8.

The purpose of the air space is to allow for thermal expansion of the oil.

ANSWER 9.

Blue smoke indicates that oil is being burned in the engine. The most common causes are leaking piston rings or valve oil seals.

ANSWER 10.

The most probable cause of wildly fluctuating oil pressure in a piston engine is insufficient oil contents. Under such circumstances the pressure fluctuations are the result of the pump periodically running dry because of air bubbles circulating in the system.

ANSWER 11.

Scavenge pumps draw oil from the sump and transfer it back to the oil tank. They are of greater capacity than pressure pumps to allow for the greater volume due to air bubbles within the scavenged oil.

ANSWER 12.

The term coring refers to a condition in which excessive cooling causes a highly viscous layer of oil to form on the inner walls of the oil cooler matrix. This reduces oil flow through the cooler causing a rapid increase in oil temperature.

ANSWER 13.

An opposed piston engine is one in which the cylinders are at opposite sides of a common crankshaft.

ANSWER 14.

To increase its temperature.

ANSWER 15.

Brake Horsepower = Indicated Horsepower – Friction Horsepower

ANSWER 16.

The Mechanical Efficiency of a piston engine is the ratio of its Brake Horsepower to its Indicated Horsepower.

ANSWER 17.

Crank throw is half the piston stroke.

ANSWER 18.

The volumetric efficiency of a piston engine is the ratio of the mass of mixture drawn into the cylinder during the induction stroke, to the mass of the same volume (the swept volume) of mixture at ambient pressure. Alternatively this may be stated as the ration of the induced volume at ambient pressure, to the swept volume.

ANSWER 19.

The valve springs cause the valves to close.

ANSWER 20.

The compression ratio of a piston engine is the ratio of its total volume to its clearance volume.

Compression Ration = Total Volume/Clearance Volume

Or, because Total Volume = Swept Volume + Clearance Volume;

Compression ratio = (Swept Volume + Clearance Volume)/Clearance Volume

ANSWER 21.

Increasing cylinder head temperature will increase incoming mixture temperature causing it to expand. This will reduce the mixture density thereby reducing the mass drawn into the cylinder and hence volumetric efficiency.

ANSWER 22.

Pressure and temperature both reduce as energy is extracted by the piston.

ANSWER 23.

Increasing engine RPM reduce volumetric efficiency due to increasing friction losses in the inlet manifold.

ANSWER 24.

Valve overlap is the period of crankshaft rotation during the latter part of the exhaust stroke and first part of the induction stroke during which both the exhaust and inlet valves are open simultaneously.

ANSWER 25.

As engine RPM increases after starting centrifugal force causes the weights in the impulse coupling to move such that they no longer contact the stop pins in the outer casing. The impulse coupling then acts as a normal drive shaft.

ANSWER 26.

Valve overlap is expressed in degrees of crankshaft rotation.

ANSWER 27.

The oil control rings will be located between the oil scarper rings and the compression rings around the skirt of the piston.

ANSWER 28.

The temperature and pressure of the mixture in the cylinder will increase, its volume will decrease, and its mass will remain unchanged during the compression stroke.

ANSWER 29.

The compression ration (CR) = total volume (TV) divided by the clearance volume (CV). The total volume (SV) is the swept volume plus the clearance volume. In this case total volume is 1500cc.

This means that CR = 1500/CV = 5, so CV = 1500/5 = 300cc

But TV = (SV + CV) = 1500cc, and CV = 300

So SV = 1500 – 300 = 1200cc

ANSWER 30.

The specific fuel consumption of a piston engine is defined as the weight of fuel required to produce a unit of power per unit of time. In imperial units this would be specified in 1bf/Brake Horsepower/Hour.

ANSWER 31.

The fuel:air ratio at which a piston engine produces the most power at a given RPM is known as the best power mixture.

ANSWER 32.

Hydraulicing is the situation when liquid trapped in one or more of the cylinders, prevents engine rotation. The most common cause is oil leakage past the piston rings in inverted or radial engines.

ANSWER 33.

Initial rotation of the starter causes the drive pinion to move along a helical spline to engage with the engine starter ring. When engine speed exceeds starter speed the pinion moves back along the spline out of engagement, assisted by a spring.

ANSWER 34.

A spark which occurs at 20 degrees before TDC is more advanced than one that occurs at 5 degrees before TDC.

ANSWER 35.

Engine RPM has no effect on the timing of the degree of valve overlap angle because these are fixed functions of engine design and construction.

ANSWER 36.

Because of the effects of gas inertia and the restriction to flow caused by the valves and the manifolds, insufficient gas would by induced and exhausted if the valves opened and closed at TDC and BDC. Opening the valves early and closing them late extends the period of induction and exhaust thereby improving volumetric efficiency and power output.

ANSWER 37.

Because of the phenomenon of ineffective crank angle piston motion is very small over a finite angular range close to TDC and BDC. This means that there is little or no tendency for backflow in the inlet and exhaust manifolds provided the degree of the valve lead and lag is well within the effective crank angle range.

ANSWER 38.

In the two stroke system the induction, compression, power and exhaust processes for each cylinder are achieved in a single revolution (two strokes) of the engine. In the four system the same processes are carried out over two revolutions (four strokes).

ANSWER 39.

Because of their relatively low rotation rates piston aero-engines do not normally employ automatic ignition advance and retard and hence ignition is commenced at the same point in rotation regardless of engine RPM. Also the flame rate and hence time required to complete combustion is approximately constant at all speeds. The above factors mean that the timing of combustion and hence maximum pressure does not vary with RPM. Increasing RPM does however mean that the piston will have moved further down the cylinder by the time combustion completion and hence maximum cylinder pressure occurs.

ANSWER 40.

Cylinder head temperature in a modern piston aero-engine is normally measured using surface mounted thermocouples. The electrical power used is produced by the seebeck effect on the hot and cold junctions.

ANSWER 41.

The power source in a piston aero-engine magneto system is mechanical energy provided through the engine drive shaft. During engine starting the rotation rate of the magneto drive shaft is too low to provide sufficient energy to generate the high voltages required at the spark plugs. This problem is usually overcome by means of HT or LT booster coils or impulse couplings.

ANSWER 42.

During normal operation with both ignition systems operating combustion is initiated at two points and conducted on two fronts. It is therefore completed in approximately half the time that would be taken with a single ignition point. During the magneto drop check each ignition system is tested in isolation from the other, thereby causing ignition to occur at a single point. Because of the longer time required for combustion on a single flame front, lower cylinder pressures are achieved reducing power output and RPM.

ANSWER 43.

If the engine stops when one ignition system is switched off this indicates that the other system is defective in that it is producing no spark or insufficiently strong sparks to initiate combustion.

ANSWER 44.

The purpose of the indication vibrator circuit is to provide a boosted low tension current to the magneto primary circuit during engine starting when the induced primary current is too low due to the low magneto rotation rate.

ANSWER 45.

By varying the air supply from the float chamber to the diffuser tube the pressure drop across the main fuel jet can be modified to adjust fuel mixture. By directly controlling the fuel flow to the main jet effectively altering the size of the jet.

ANSWER 46.

At high altitude the low air pressures mean that it is easier for high voltages to flashover or jump across gaps in circuits. Magnetos that have been pressurised to prevent this are painted blue or grey.

ANSWER 47.

With the engine running at idle RPM the lowest air pressure in the carburettor will occur where the air is caused to accelerate around the closed throttle butterfly valve. Under these circumstances the pressure drop in the choke tube will be insufficient to draw an adequate supply of fuel through the main jets to keeps the engine running. This problem is normally overcome by positioning an additional slow running or idling jet close to the butterfly valve to maintain fuel flow at low RPM.

Piston Engine Power Augmentation and Performance

Piston Engine Power Augmentation and Performance

Introduction

1. The internal combustion engine is a heat engine. That is to say, it converts heat energy created by burning fuel in its cylinders into mechanical energy by means of the moving pistons and crankshaft. The power delivered by the engine is the function of the work done by these moving parts over a period of time.

2. Thus, the power output of the engine is directly proportional to the quantity of fuel burned in the cylinders. The fuel will only burn if it is mixed with air in the correct ratio, which for efficient combustion must be 15 parts of air to every one part of fuel by weight. This mixture of fuel and air, when supplied to the engine cylinders is known as the fuel/air charge. Consequently, the statement at the beginning of this paragraph can be expanded to say that the power output of the engine is directly proportional to the weight of the fuel/air charge.

Charge Induction

3. In a normally aspirated engine, for air at ambient atmospheric pressure to enter the cylinder, the pressure in the cylinder must be less than the ambient atmospheric pressure. This reduced pressure is created in the cylinder by the movement of the piston during the induction stroke. Hence, the charge in the cylinder at the end of the induction stroke is always at a pressure lower than ambient atmospheric pressure, even when operating with the throttle valve wide open.

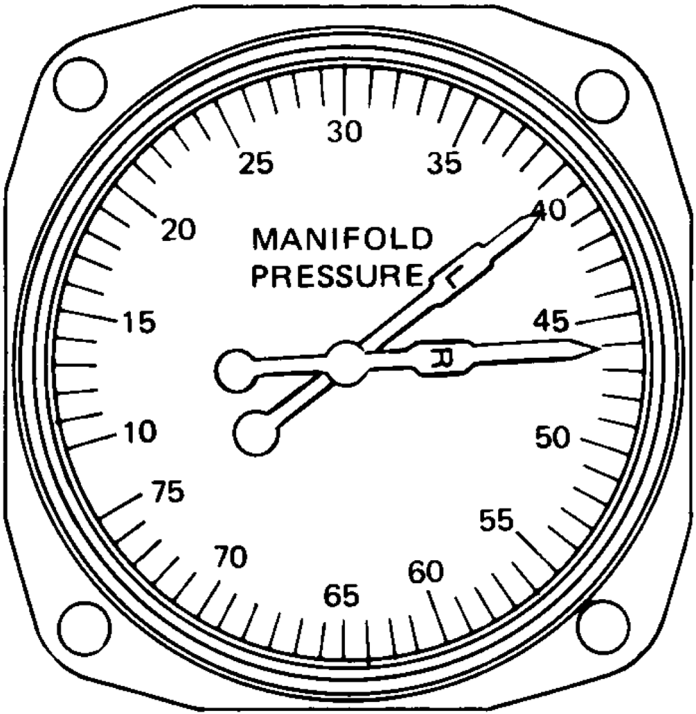
4. It follows that, if the charge pressure is less than atmospheric pressure, then the charge density must be less than ambient atmospheric density. Since density is defined as mass per unit volume, the mass (or weight) of the induced charge must be less than the same volume of air at ambient atmospheric pressure.

5. Let us now examine this. The weight of the charge is directly proportional to its density, which is in turn directly proportional to its pressure. Therefore, developing the conclusion of the second paragraph on this page, the power output of a piston engine is directly proportional to the charge pressure. The charge pressure is, by definition, the pressure in the induction system during the induction stroke. The induction system is otherwise known as the inlet air manifold and so the induction system pressure is more commonly referred to as Manifold Air Pressure (MAP).

6. From this it can be concluded that the power output of a piston engine is directly proportional to the MAP. MAP is measured at a point in the induction system between the throttle valve and the cylinder inlet valves.

7. A typical manifold pressure gauge presentation for a twin-engine aircraft is shown at Figure 15-1. Strictly speaking, MAP stands for manifold absolute pressure, which means that the gauge is calibrated to read pressure above absolute zero. Thus when the engine is stopped the gauge shows the ambient atmospheric pressure. The gauge scale in Figure 15-1 is calibrated in inches of mercury (ins. Hg), which is usually the case.

FIGURE 15-1
Manifold Pressure Gauge Presentation



8. The manifold pressure gauge typically contains an evacuated chamber sealed by a diaphragm, or containing a bellows unit connected to the inlet manifold. As pressure increases the diaphragm or bellows flexes proportionately to operate the gauge pointer.

9. The manifold pressure gauge is of particular importance in supercharged engines. We have already established that MAP is directly proportional to engine power, since it has direct effect upon the mean effective pressure in the cylinders. This, in turn, directly affects the cylinder operating temperature and high cylinder temperature leads to pre-ignition and detonation.

10. With a fixed-pitch propeller installation the power output of the engine is indicated by the engine rpm. However, when an engine is driving a constant speed (variable-pitch) propeller the engine rpm reamains constant and power increases or decreases according to the propeller pitch setting. It is therefore important that the pilot is aware of the manifold pressure, and thus the power output of the engine, to ensure that the engine is not producing excessive torque.

Factors Affecting Induction

11. When the throttle valve is wide open it offers virtually no obstruction to the air passage to the cylinders. Under these conditions the MAP is at its highest possible value but, as explained above, in a normally aspirated engine it is still less than ambient atmospheric pressure.

12. If the throttle valve is set at anything less than wide open it offers a restriction to airflow to the cylinders. The pistons are trying to draw air past this restriction, and must create greater suction in order to do so. Increased suction equals reduced MAP and therefore less weight of charge. Closing the throttle reduces engine power!

13. If the induction passages have sharp bends or rough internal surfaces the flow of air to the cylinders will be impeded. The result will be that MAP is lower than it could be and therefore charge weight is less than it could be. Poor intake design results in lost power.

14. The higher the air temperature, the lower the air density and reduced air density means reduced charge weight, which means reduced power. When operating in conditions of high ambient air temperature the power available for any given throttle setting is reduced. This includes full throttle, so the maximum power available is reduced. Any increase to the induction air temperature will decrease the charge weight and, therefore, the engine power output.

15. If the pressure of a volume of gas is decreased its density is decreased proportionally. When a piston engine is operated in conditions of low atmospheric pressure, as at altitude, the decreased density of the induced air results in lower charge weight. Thus, the higher the operating altitude the lower the power output of the engine.

Volumetric Efficiency

16. The astute student will by now have spotted that all the factors which affect the weight of the induced charge are also those which affect the volumetric efficiency of the engine. Volumetric efficiency is the ratio of the volume of the induced charge at atmospheric pressure to the volume displaced by the piston. In a normally aspirated engine the induced charge pressure is always at less than ambient atmospheric pressure. Hence, if that mass of reduced-pressure gas were at ambient atmospheric pressure its volume would be less than the volume displaced by the piston. Consequently, a normally aspirated piston engine is always operating with a volumetric efficiency of less than 100%.

17. Suppose now an engine with a volumetric efficiency of 80% is operating at full throttle and let us also suppose that it is producing 80 BHP. If it were possible to increase the volumetric efficiency, by increasing the weight of the charge, to 100% then the power output of the engine would increase in direct proportion (in theory at least) to 100 BHP.

Super Charging

18. Increasing the weight of the charge beyond that possible by normal aspiration is known as super charging. It is achieved by forcing air into the induction system with some sort of air pump, or compressor, instead of sucking it in with the pistons. The more air forced in, the higher the MAP and therefore the greater the power output of the engine. By this means it is possible to increase the power developed by a piston engine so that the same power is available at altitude as at sea level, significantly improving both the climb performance and the operational ceiling of the aircraft. This is known as an altitude-boosted engine. Further, the power available at sea level can be increased, giving improved take-off performance. This is known as a ground-boosted engine.

19. The supercharger is a blower, or compressor, that draws in atmospheric air and discharges it to the engine induction system thereby increasing the pressure, density and consequently the mass of air in the induction system. Maintaining the MAP and density (mass) of the air-fuel charge at a constant value during the climb will, of itself, maintain engine power constant. Because of the decrease in ambient atmospheric pressure, scavenging of the exhaust gases improves. A greater proportion of the charge is able to enter the cylinders and the engine need do less pumping work in expelling the exhaust gas. The result is that, if constant MAP is maintained in the climb, there is a power increase at the rate of about one per cent per 1000 feet of altitude gained.

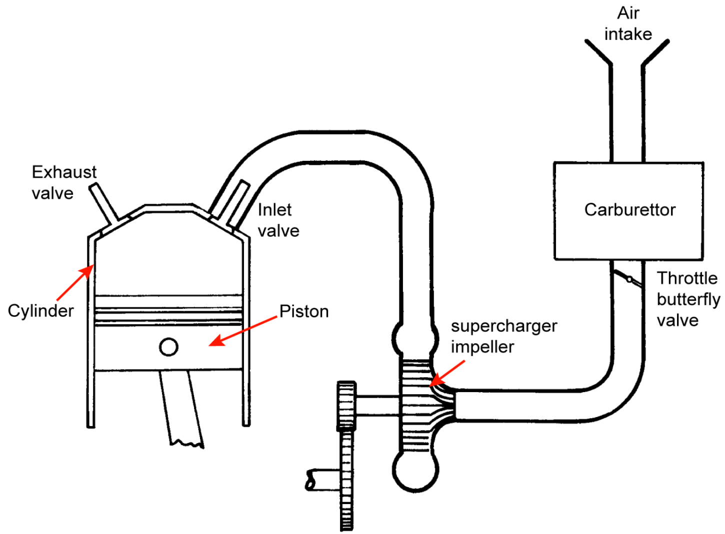
20. With an altitude-boosted system, the throttle will have been less than fully open on take-off, in order to avoid over-boosting. In the climb, as ambient air pressure decreases, the throttle is gradually opened to maintain the required power (either manually or automatically), and consequently throttling losses are reduced. Throttling losses occur when the throttle butterfly valve is other than fully open, since the air has to accelerate around this obstruction and suffers a reduction in pressure (and density) in consequence. As the aircraft continues climbing it will eventually reach an altitude at which, in order for the supercharger to maintain MAP constant, the throttle valve will have to be wide open. This is known as the full throttle altitude for that MAP/power. Any further increase in altitude must result in a fall in MAP and power unless the supercharger can be made to pump more air (by increasing its speed).

21. When an aircraft is climbing at rated power (the maximum continuous power for the engine) and it reaches full throttle altitude this is also known as the rated altitude. Rated altitude is the greatest altitude at which a supercharged engine can maintain rated power. In a turbo-charged engine this same condition is usually referred to as the critical altitude, but more of that later.

22. Supercharging or turbocharging of a piston engine provides a means of achieving a greater power output from a given engine without an undue increase in the weight and size of the power unit concerned.

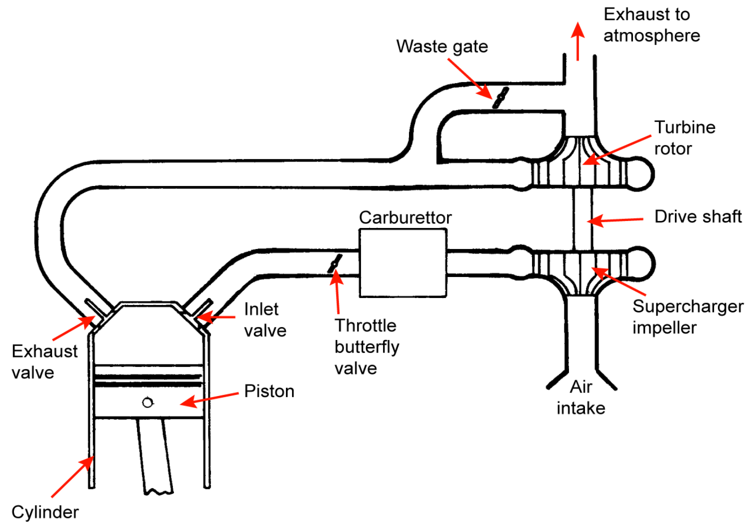
23. The basic difference between a supercharger and a turbocharger is that the former is driven directly by the engine, with an ancillary drive much like that driving the fuel pump, and the latter is driven by the exhaust gases. The basic layout of a mechanically driven, or internal-type supercharger system is shown at Figure 15-2.

FIGURE 15-2
Internal
Supercharger



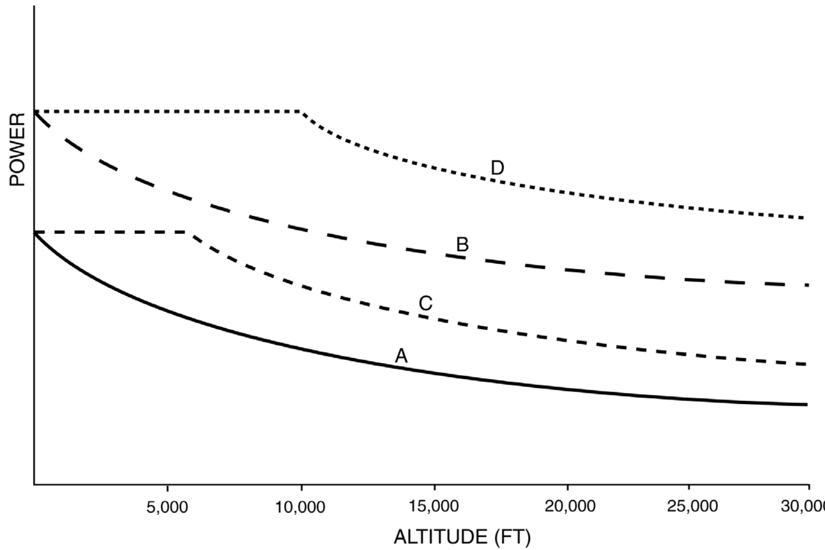
24. The basic layout of an exhaust driven turbocharger, or external-type supercharger is shown at Figure 15-3.

FIGURE 15-3
External Supercharger (Turbocharger)



25. Since the power output of a piston engine is dependent upon the weight of the induced charge, and this is in turn dependent upon the density of the air, it follows that the power output of a normally aspirated engine must fall as altitude increases. This is shown in curve (A) at Figure 15-3. A ground-boosted, internally supercharged, engine will produce greater power at any given altitude than a similar unsupercharged engine, but will exhibit loss of power with altitude in the same manner, as shown in curve (B). An altitude-boosted engine is able to maintain sea level full power to a given altitude, but thereafter power will decline with increasing altitude as shown at curve (C). A turocharger is capable not only of increasing power at sea level, but also of maintaining that power to a significant altitude as shown in curve (D) of Figure 15-3.

FIGURE 15-4
Curves of Maximum Power vs. Altitude



Turbochargers or External Superchargers

The Turbocharger

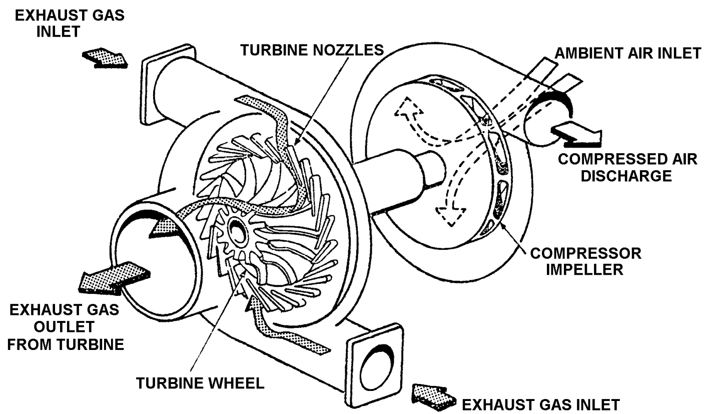
26. When the impeller of the supercharger is driven by exhaust gas flow, rather than by direct drive from the engine, it is known either as a turbocharger or as an external-type supercharger.

27. As with an internal-type supercharger the ratio of compressor outlet pressure to compressor inlet pressure is the supercharger/turbocharger ratio and, for a unit of given dimensions, this ratio will depend upon the speed of the compressor.

28. The output of a turbocharger remains essentially constant in the climb, since its speed is increased to match decreasing atmospheric pressure. Hence, scavenging improves as the exhaust back pressure decreases, so the volumetric efficiency of the engine increases with altitude up to the critical altitude (discussed shortly).

29. A turbocharger arrangement is illustrated at Figure 15-5.

FIGURE 15-5
Turbocharger gas Flow

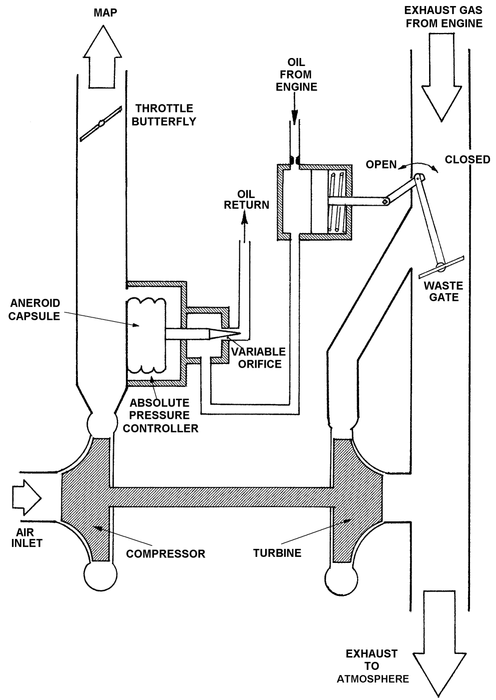


30. If all of the exhaust gases were used to drive the turbocharger its speed of rotation, and the consequent compressor output pressure, would be too high, as would the back pressure in the exhaust manifold.

31. It is therefore necessary to control the speed of rotation by directing the exhaust gases through a Y junction, whereby one arm of the plumbing diverts the exhaust gases to the turbocharger turbine, whilst the other arm directs the gases to atmosphere through a valve known as a waste gate. This principle is illustrated at Figure 15-6.

Turbocharger Control

FIGURE 15-6
Turbocharger Control



32. From Figure 15-6 it can be seen that the position of the waste gate governs the amount of exhaust gas available to drive the turbocharger turbine, and therefore governs the manifold air pressure downstream of the turbocharger compressor. The range of waste gate movement is limited, to ensure that some gas is always routed through the turbine, to prevent stalling.

33. The waste gate position is set by a hydraulic actuator, within which engine oil pressure opposes a spring to operate the waste gate linkage. Oil is supplied to the actuator through a fixed orifice, and is allowed to return to the engine sump through a variable orifice valve, operated by the absolute pressure controller.

34. The absolute pressure controller consists of an aneroid capsule connected to the variable orifice needle valve. The capsule is mounted in a chamber connected to the induction manifold on the outlet side of the turbocharger compressor, and is therefore sensitive to compressor discharge pressure.

35. The function of the absolute pressure controller is to maintain a constant absolute pressure (i.e. pressure above absolute zero) between compressor and throttle valve. This is known as the compressor discharge pressure. Manifold air pressure (MAP), between throttle valve and engine inlet valves, is controlled by the throttle valve setting.

36. Let us assume that the throttle lever has been advanced, increasing the throttle butterfly valve opening. As the valve opens, admitting more charge air to the inlet manifold, MAP will increase. However, the greater demand on the compressor causes compressor discharge pressure to fall and this will cause the aneroid capsule in the absolute pressure controller to expand, partially closing the variable orifice (oil outlet) needle valve.

37. Oil flow through the fixed inlet restriction will now exceed outflow through the variable orifice needle valve, and oil pressure on the waste gate actuator piston will increase. Overcoming spring pressure, the increased oil pressure will move the actuator piston to the right (in Figure 15-6), closing the waste gate to direct more gas through the turbine.

38. Turbine, and therefore compressor speed will increase, increasing compressor discharge pressure to maintain the higher MAP required by the original throttle advance. As compressor discharge pressure rises the absolute pressure controller capsule will be progressively compressed, opening the variable orifice valve until oil outflow just matches oil inflow through the fixed restriction. At this point a new steady-state condition is reached, with the throttle butterfly valve open wider, waste gate further closed, turbocharger rotating faster to maintain constant compressor discharge pressure and MAP higher to match the new power setting.

39. A decrease in MAP, whether due to closing the throttle valve or underspeeding of the turbocharger, will have the reverse effect to that described above.

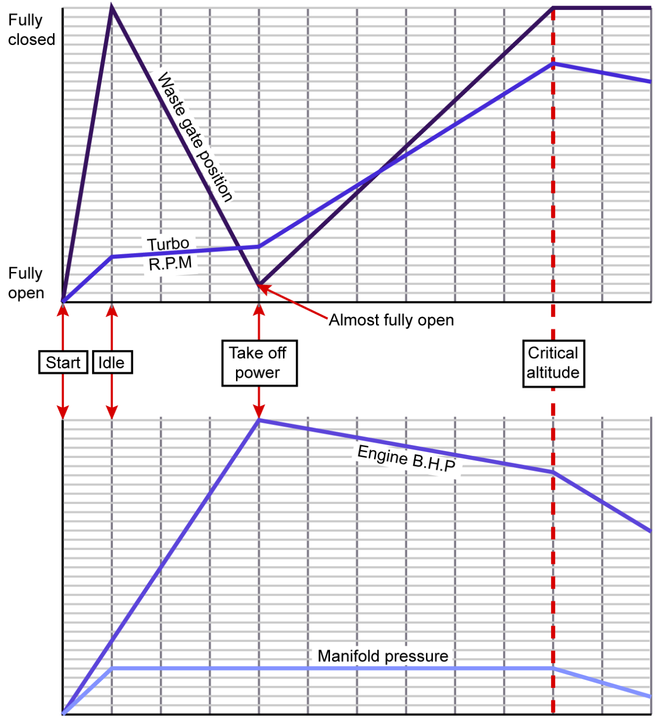
40. When the engine is stopped there is no engine oil pressure acting on the waste gate actuator piston, so the spring will move the actuator to the waste gate full open position. The main exhaust gas path is therefore direct to atmosphere to minimise exhaust back pressure during starting.

41. Once the engine is idling, or at low power, the manifold air pressure will be low and the absolute pressure controller capsule will only be partially compressed. Consequently, the variable orifice needle valve will be almost closed and oil pressure will build up in the waste gate actuator, moving the waste gate towards the closed position. Because, at low rpm, the oil pressure will also be low the actuator spring will not be fully compressed and the waste gate will probably not reach the fully closed position. However, most of the available exhaust gas will be directed to the turbine, maintaining an adequate flow of induction air through the compressor.

42. For every possible power setting selected by the pilot there is an altitude at which the waste gate will have to be fully closed in order to maintain the MAP required for that power. Above that altitude, power will fall since the turbocharger can no longer increase speed to match the falling atmospheric pressure. When a higher power setting cannot be selected (the throttle is fully open), the turbocharger's critical altitude has been reached.

Piston Engine Power Augmentation and Performance

FIGURE 15-7



The Relationship of Waste Gate Position to Engine Power, Manifold Pressure and Turbo charger R.P.M. throught flight

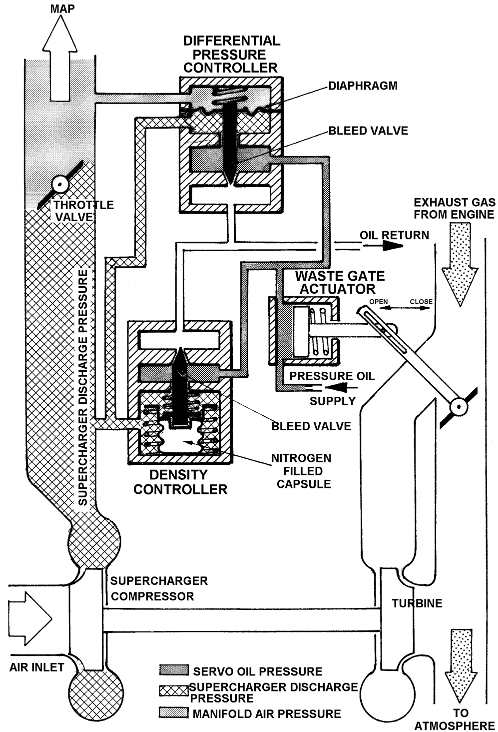
Manifold Air Pressure (MAP) Control

43. Turbochargers may be fitted with a fixed datum controller, to ensure that MAP does not exceed a pre-set maximum value. Alternatively a variable datum controller may be fitted, whereby any MAP exceeding a value set by the pilot will be prevented. On some older engines the waste gate may be controlled manually, in which case the pilot is required to progressively close the waste gate in the climb to maintain the boost pressure, and to progressively open it in the descent to prevent over-boosting.

44. The increased speed of rotation of the turbocharger compressor, which is necessary at altitude in order to maintain MAP, will increase the temperature of the charge going to the cylinders. This will result in a small loss of power (which can be overcome by boosting the MAP slightly) and also an increase in cylinder head temperature.

45. Turbochargers are prone to a condition known as bootstrapping. This is a hunting condition that arises because of the inertia of the rotating assembly (turbine and compressor). The assembly takes time to accelerate or decelerate following an alteration in exhaust gas flow through the turbine, and this is known as 'turbo lag'. Suppose for example, the waste gate is opened to increase exhaust gas flow. After a delay due to turbo lag the turbocharger speed increases, which causes an increase in MAP, which increases the power output of the engine, which increases exhaust flow further.......and so on. In an attempt to overcome this situation the absolute pressure controller actuates the waste gate, which decreases exhaust gas flow to the turbine. After a delay (due to turbo lag) the turbine begins to decelerate, which decreases compressor speed, which decreases MAP, which decreases power, which decreases exhaust gas flow.......and so on. The problem is not usually as bad as it sounds, but to overcome it additional controller units may be fitted. A dual-unit turbocharger control system is illustrated schematically at Figure 15-8.

FIGURE 15-8
Dual Unit Supercharger Control System



46. The single-unit turbocharger control system illustrated at Figure 15-6 is designed to maintain a constant supercharger discharge pressure, regardless of the manifold air pressure (MAP) required. Hence, at low power, where a low MAP is required the throttle valve will be only partly open, creating a large pressure drop across the valve. This is an inefficient way of operating, since it means that the turbocharger is being driven faster than necessary for the manifold air pressure required.

47. By introducing a second controller (as shown at Figure 15-8), which is sensitive to the differential pressure across the throttle valve, the waste gate is adjusted to reduce turbocharger discharge pressure at partial throttle openings. Because the Differential Pressure Controller is sensitive to the pressure drop across the throttle valve it smooths out any tendency towards hunting of the manifold pressure, or bootstrapping.

48. A chamber in the Differential Pressure Controller is divided by a diaphragm which has supercharger discharge pressure acting on one side and manifold air pressure plus a spring acting on the other. When the throttle valve is fully open the pressure drop across the valve is least and the diaphragm operated bleed valve in the Differential Pressure Controller is held closed by the spring. The more the throttle valve is closed the greater the differential pressure across it, and therefore across the diaphragm. Hence, as the throttle valve is closed the differential between supercharger discharge pressure and manifold air pressure will increase until sufficient to overcome the spring force. Beyond this the diaphragm will flex to progressively open the bleed valve. This will reduce the waste gate actuator servo pressure, causing the waste gate to open and reduce turbocharger rpm to match the reduced power selected.

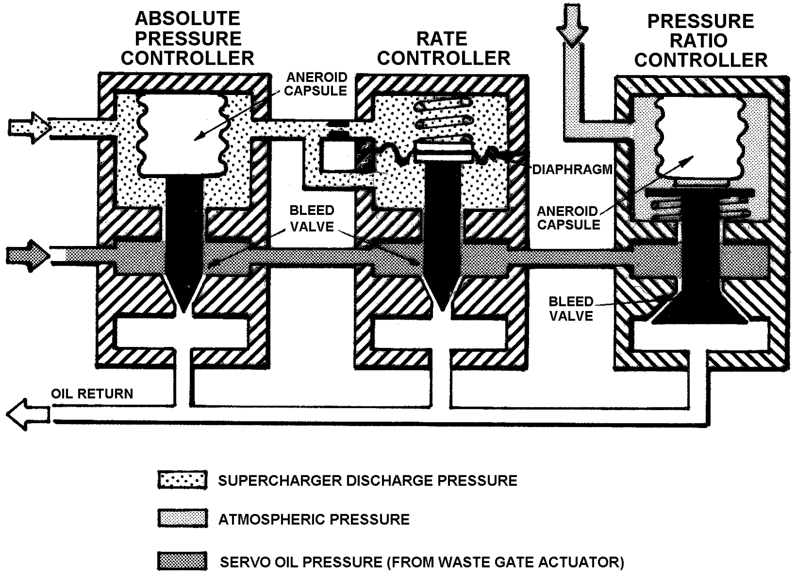
49. The Density Controller prevents the supercharger discharge pressure from exceeding its limiting value. It controls the waste gate actuator servo oil pressure only when the throttle valve is fully open and up to the turbocharger's critical altitude (above critical altitude the waste gate will be fully closed). The closed capsule is filled with dry nitrogen and is consequently sensitive to air temperature changes as well as pressure changes.

50. If supercharger discharge pressure rises above a pre-set value the capsule will be compressed, opening the bleed valve and reducing waste gate actuator servo pressure. The actuator spring will move the waste gate towards OPEN, reducing turbocharger rpm and thereby reducing supercharger discharge pressure.

51. If the discharge temperature rises the capsule will expand, slightly closing the bleed valve and increasing the waste gate actuator servo pressure, moving the waste gate towards CLOSE and diverting more exhaust gas to drive the turbine. This will increase turbocharger rpm, and thus supercharger discharge pressure, to maintain the density of the charge air.

52. Some ground-boosted turbochargers use a triple-unit control system in which two controllers control the waste gate actuator servo pressure up to critical altitude and a third takes over from these above critical altitude. Such a control system is illustrated at Figure 15-9. The waste gate, turbocharger and throttle valve are omitted from the diagram for simplicity. Their layout is as shown at Figure 15-8.

FIGURE 15-9
Triple Unit Turbocharger Control System



Absolute Pressure Controller. This controls the supercharger discharge pressure below critical altitude in exactly the same way as the absolute pressure controller at Figure 15-6.

Rate Controller. The rate controller controls the rate at which supercharger discharge pressure can increase, in order to prevent a surge of boost pressure, which might cause interim overboosting of the engine when the throttle is opened.

53. Supercharger discharge pressure is led to both sides of the bleed valve diaphragm, but supply to one side is restricted. Thus, if the throttle valve is opened supercharger discharge pressure will fall and the restricted orifice will ensure a temporary differential pressure across the diaphragm, closing the bleed valve, increasing servo pressure and moving the waste gate towards CLOSE until pressure on either side of the diaphragm has equalised.

54. This will increase turbocharger speed to restore supercharger discharge pressure. However, if discharge pressure rises too rapidly the restrictor will ensure that pressure rises more on one side of the diaphragm than the other. The diaphragm will therefore flex, opening the bleed valve and moving the waste gate temporarily towards OPEN to limit the rate of increase of supercharger discharge pressure.

55. Once pressures on either side of the diaphragm have equalised, the spring will return the bleed valve to the closed position, returning primary control to the Absolute Pressure Controller.

Pressure Ratio Controller. As aircraft altitude increases, and ambient atmospheric pressure decreases, the supercharger has to rotate faster and compress more air in order to maintain constant manifold air pressure, and therefore engine power. The additional work done by the supercharger compressor increases the temperature of the air delivered to the engine. This could, if continued to sufficient altitude, lead to detonation.

56. In order to prevent this, a limit is placed upon maximum manifold air pressure above a specified altitude (often 16,000 feet). The pressure ratio controller automatically limits supercharger discharge pressure above the specified altitude by progressively opening a bleed valve and reducing waste gate actuator servo pressure, thereby progressively reducing turbocharger speed. This results in a controlled progressive reduction of supercharger discharge pressure.

57. Ambient atmospheric pressure is supplied to a chamber containing an aneroid capsule. At the specified altitude the capsule will have expanded sufficiently to contact the bleed valve stem. If altitude is increased further the capsule expands further, opening the bleed valve to reduce servo pressure and adjust the waste gate so as to allow more exhaust gas to atmosphere and less to drive the turbocharger turbine. The Pressure Ratio Controller is set to reduce the waste gate actuator servo pressure so as to maintain a ratio between supercharger discharge pressure and atmospheric pressure of, typically, 2.2: 1.

58. Finally, a pressure relief device, in the form of a simple poppet valve, is normally fitted to the inlet manifold to bleed off excess pressure (overboosting) in the event of failure of the boost pressure limiting devices, or failure of the waste gate.

59. Overboosting would cause excess power and could lead to serious structural damage to the engine.

Internal Superchargers

60. Mechanically driven, or internal-type superchargers demand considerable power to drive them. Whilst this was acceptable with the high powered engines of the 40's and 50's, it is not so with the relatively low powered piston engines of modern general aviation aircraft. Hence, the supercharger has been almost universally superseded by the turbocharger.

61. Those few engines still in use which are fitted with internally-driven superchargers generally employ a single impeller compressor driven at a fixed speed ratio to the crankshaft (typically between six and twelve times crankshaft speed). This type of supercharger is usually capable of maintaining sea level manifold pressure up to an altitude of between 5,000 and 10,000 feet, depending upon the gear ratio, at rated power. In high-powered piston engines of the past, two-speed superchargers or double-impeller compressors have been used to increase the rated altitude of the engine.

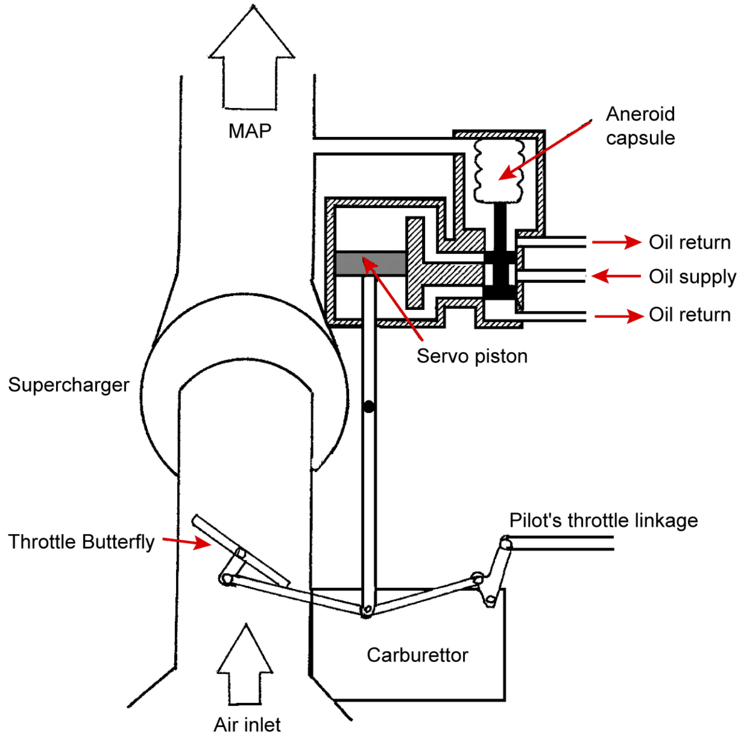
62. Since the supercharger speed is geared to engine crankshaft speed, at full power rpm both are rotating at maximum speed. The function of the supercharger is to maintain sea level MAP at altitude, where the air is less dense. In the relatively dense air at sea level it follows that the supercharger impeller is rotating faster than necessary. However, maximum engine rpm must be maintained for full power, so MAP can only be limited by partly closing the throttle valve.

63. As the aircraft climbs it will be necessary to progressively open the throttle valve in order to maintain constant MAP. When an altitude is reached where it is necessary for the throttle valve to be fully open to maintain MAP, this is known as full throttle altitude, as previously discussed. For each power setting there is a different engine rpm/impeller speed and so for each power setting there is a different full throttle altitude. The lower the power, the lower the full throttle altitude.

64. From the foregoing it will be seen that in conditions of high atmospheric pressure (low altitude) the supercharger is doing more work in compressing air than necessary, which tends to overheat the induction air. Excess induction air temperature may not only cause detonation, but is also partly defeating the object of supercharging (high temperature means decreased density). For this reason superchargers are always fitted downstream of the carburettor and throttle valve, where evaporation of the fuel in the charge can exert a cooling effect.

65. Control of manifold air pressure (MAP) is achieved by adjusting the position of the throttle valve. The pilot may adjust the throttle valve in the normal way, but to relieve the pilot of the need to progressively open the throttle valve for a constant-power climb an automatic device is often provided for this purpose. Such a device, known as an Automatic Boost Controller, is illustrated at Figure 15-10.

FIGURE 15-10
Supercharger Automatic Boost Controller



66. An aneroid capsule, sensitive to MAP, controls the position of a piston valve supplied with oil under pressure (from the engine's lubricating oil system). Suppose the pilot has set the throttle valve for the required MAP at the start of the climb. As the aircraft climbs air intake pressure falls, whilst supercharger rpm remains constant. This will cause the supercharger discharge (manifold air) pressure to tend to fall. The drop in MAP is sensed by the aneroid capsule which expands, positioning the piston valve so that oil pressure is directed to the underside of a servo piston whilst connecting the upper side to a return system. The servo piston moves upwards as a result and, through linkages, opens the throttle valve to admit more air to the supercharger, restoring MAP.

Piston Engine Performance

Piston Engine Performance

Introduction

1. The factors affecting the power output of an aircraft piston engine are manifold air pressure (MAP) and engine speed (rpm). It has been shown that increasing the MAP increases the power output in direct proportion. In the section dealing with power (3.9), it was shown that calculation of indicated horsepower includes the number of working strokes per minute as one of the factors directly related to power. The higher the engine rpm the greater the number of working strokes per minute and the greater the power developed. To achieve a given power output from a piston engine requires that a particular MAP and rpm must be selected.

2. With an engine driving a fixed pitch propeller the two factors are inextricably linked. The only way in which the power of the engine can be increased or decreased is by adjustment of the engine/propeller rpm. This is achieved by operation of the throttle, which affects the manifold air pressure, of course, but this is a secondary consideration. A given airspeed or rate of acceleration requires a particular propeller thrust which, in turn, requires a particular engine/propeller rpm. With such a power plant there is no MAP gauge, the pilot's 'power' control is the throttle and the only 'power' reference is the rpm gauge.

MAP versus rpm

3. Where a constant speed, variable pitch propeller is driven by a normally aspirated or super/turbo-charged engine the pilot now has two controls which, together, not only control the power, but also the performance of the engine. Engine/propeller rpm is controlled by the propeller governor, which is set by the pilot using the rpm (propeller) lever. Positioning the propeller lever selects a particular rpm and the governor adjusts the propeller blade pitch to maintain that rpm constant. When a higher rpm is selected the governor moves the propeller blades to a finer pitch, which offers less resistance to rotation and so the engine is able to rotate the propeller faster and engine/propeller rpm increase. Selection of a lower rpm causes the governor to move the propeller into coarser pitch, with opposite results.

4. The second pilot control is the throttle, which is used to control engine power. The more the throttle valve is opened the higher the MAP and the greater the power developed. If the pilot were to open the throttle, increasing MAP, engine power output would increase. The propeller rpm would not change however, since the governor would coarsen the propeller to maintain constant rpm. The engine is working harder to maintain the same rpm and the propeller blade angle is now incorrect for the airspeed.

5. Similarly, if the propeller (rpm) lever is advanced the engine/propeller rpm will increase but engine power remains the same. The engine is now delivering insufficient power for the higher airspeed associated with higher rpm.

6. In order to obtain the optimum performance from an engine driving a constant speed propeller it is necessary to select a MAP and rpm which gives maximum propeller thrust at optimum power for a given airspeed. Consequently, throttle and propeller lever are never operated in isolation, but in conjunction with each other to produce the optimum power (in the climb for example) for optimum fuel economy, and airspeed for range and endurance. The manufacturer determines the optimum MAP and rpm settings for each situation during initial aircraft type testing. These are then published in chart or tabular form for the pilot.

7. It is particularly important to avoid setting too high a MAP for the selected rpm. This will result in the engine delivering high power at low rpm to a propeller in coarse pitch. The result is likely to be overheating of the engine (excessive cylinder head temperatures with the risk of detonation) and over-torquing of the propeller shaft due to the high propeller load.

Effect of Ambient Conditions

Pressure and Density Altitude

8. Pressure altitude is defined as an atmospheric pressure expressed in terms of altitude which corresponds to that pressure in the International Standard Atmosphere (ISA).

9. Density altitude is defined as that height (above or below mean sea level) in the ISA to which the actual density at any particular point corresponds. (In standard conditions of temperature and pressure, density altitude is the same as pressure altitude.)

10. Density altitude can be converted to pressure altitude using the formula:

$$\text{pressure altitude} = \text{density altitude} \pm (120 \times \text{temperature deviation})$$

(if temperature deviation is minus, the sign used is plus)

11. Pressure reduces in the atmosphere more rapidly than temperature, indicating a decrease in density with height at all levels. The decrease at lower levels (up to about 20,000 ft), is given by subtracting 3% of the value for any given level to obtain the value 1000 ft higher. In the ISA at 20,000 ft density is approximately half its sea level value, and a quarter at 40,000 ft.

Altitude and Temperature

12. At a given engine rpm and MAP its power output is dependent upon air density, which in turn is dependent upon pressure, temperature and humidity. These factors must be taken into account when determining the performance of an engine. In order to obtain the actual power of an engine power charts are produced by the manufacturer which show the sea level performance in terms of BHP for any given MAP and rpm and the altitude performance achieved with the same settings at various density altitudes. (Density altitude is pressure altitude corrected for ambient temperature conditions).

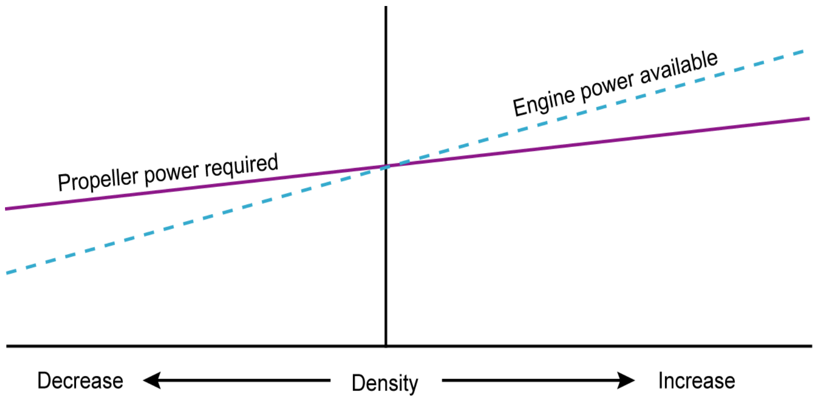
13. So long as MAP can be maintained for a given rpm, power output (performance) increases with altitude. However, when density falls (due to increased altitude or temperature) to a point where MAP cannot be maintained, power output (performance) begins to fall off.

14. The effect of density on propeller performance must also be considered. As air density decreases engine power available to drive the propeller will decrease (especially in a normally aspirated engine). However, the less dense air offers less resistance to the propeller, which consequently requires less power to maintain constant rpm

Engine/Propeller Relationship

15. The relationship between engine power available and propeller power required, in conditions of changing air density, is illustrated at Figure 16-1.

FIGURE 16-1
Engine Power Versus Propeller Power



16. When density is higher than Standard (lower than ISA temperature or higher than ISA pressure) an increased power output from the engine will be achieved for a given throttle setting. However, the power required by the propeller will be greater, since it has to be turned in a denser environment where there is increased resistance to rotation. In conditions of decreased density (higher than ISA temperature or lower than ISA pressure), the reverse will apply.

17. However, the ratio of increased/decreased engine power to increased/decreased propeller torque requirement is not constant. When density is above ISA there will be an excess of engine power available compared with the propeller power required. Consequently less power is needed to maintain the same propeller rpm or higher rpm can be achieved for the same throttle setting.

18. When density is lower than ISA the engine power produced at a given throttle setting will be reduced and there will be insufficient engine power available compared with the propeller power required. Consequently, a higher throttle setting will be necessary to maintain a given power. However, less power will be necessary to maintain a given propeller rpm in the less dense air.

Effects of Fuel-Air Ratio

19. The performance of a piston engine is affected by the fuel-air mixture ratio and it is necessary to consider the two cases of prime interest to the aircraft operator. These are the mixture for best power and the mixture for best economy.

20. The best power mixture is the fuel-air ratio under which the engine produces maximum power for a given rpm and is of importance during climb-out, for example.

21. The best economy mixture is the fuel-air ratio that gives the least fuel consumption for a given power and is of importance when flying for range.

22. At take-off power the mixture setting is FULL RICH, at which the fuel/air mixture ratio is designed to permit full power with adequate additional fuel for exhaust cooling. As an aeroplane climbs it is necessary to progressively adjust the mixture control toward the LEAN setting. This is because, with increasing altitude, the mass of the air charge induced into the cylinders decreases with the decreasing air density, whilst the mass of fuel induced remains essentially the same. Consequently, if no adjustment were made, the mixture would become progressively richer until the engine ceases to run smoothly and fuel consumption is unnecessarily high.

23. With the engine at cruise power the mixture is leaned in order to both obtain smooth operation and to economise in the use of fuel. The procedure used to determine the optimum mixture setting is usually described in the pilot's handbook or aircraft operating manual and it will vary according to whether the engine is fitted with a float carburettor or fuel injection.

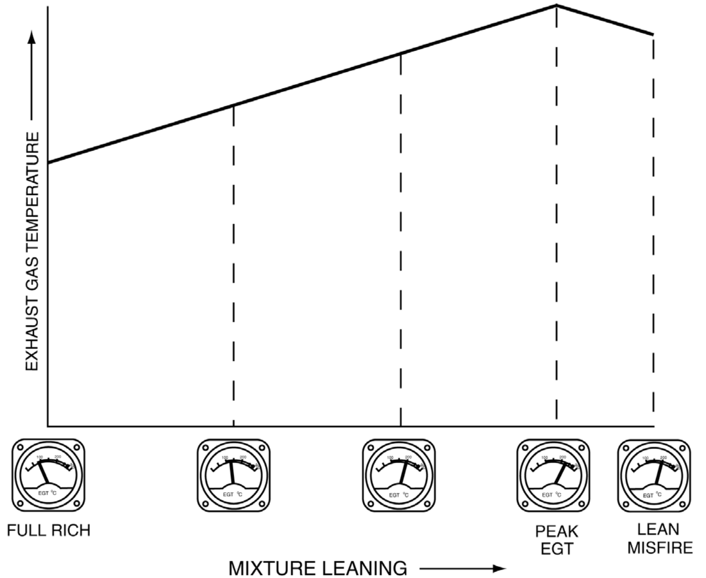
24. With a float carburettor and a fixed-pitch propeller it is usual to adjust the mixture control toward lean until the maximum engine rpm or airspeed is obtained or until engine rough running is encountered. This roughness is due to the leanest cylinder ceasing to fire. The mixture is then slightly enriched.

25. With a float carburettor and a constant speed propeller the mixture is leaned until the engine just begins to run roughly and then enriched slightly to regain smooth running.

26. With a fuel injected engine that is equipped with an exhaust gas temperature (EGT) gauge it is the usual practice to use the EGT gauge reading to determine the optimum mixture setting. At the FULL RICH setting and cruise power there is a significant amount of unburned fuel, which cools the exhaust gas and results in a low reading on the EGT gauge. Leaning the mixture reduces the excess fuel and causes the EGT to rise. When the mixture setting is such that there is no excess fuel and there is complete combustion a peak EGT is reached. If the mixture is leaned beyond this point the EGT will begin to fall because there is now cooling due to the excess air in the mixture. Any significant leaning beyond peak EGT will result in misfiring. The principle is illustrated graphically at Figure 16-2.

27. The procedure generally followed when setting the mixture at cruise power is to adjust the mixture control from FULL RICH toward LEAN until peak EGT is obtained. At this setting the best specific fuel consumption (fuel used per BHP produced per unit time) is achieved and the aircraft range is increased by approximately 15%. Alternatively, if maximum cruise power is required the mixture is leaned to the peak EGT point, and then enriched until the EGT is about 100°F (55°C) below peak value.

FIGURE 16-2


EXHAUST GAS TEMPERATURE
FULL RICH
PEAK
EGT
LEAN
MISFIRE
MIXTURE LEANING

Other Factors Affecting Performance

28. Carburettor Air Temperature affects the density, and therefore mass of air entering the induction system. The higher the temperature the lower the density and therefore the lower the power developed. Performance decreases with increased carburettor or manifold air temperature at the rate of approximately one HP for each 6°C rise. It must also be remembered that too high an induction temperature will inevitably lead to detonation.

29. Exhaust Back-Pressure, as we have seen before, has a decided effect upon the volumetric efficiency of a piston engine. The higher the exhaust back-pressure the lower the volumetric efficiency and the worse the engine performance.

30. The pressure in the exhaust system (back-pressure) above ambient atmospheric pressure is largely dependent upon the design of the system itself. The more convoluted the passage of the exhaust gases, through the exhaust ports, silencers and pipes, the higher the back pressure in the system and the more adverse its effect upon performance.

Ram Air Pressure. Some benefit in terms of engine performance is gained from the ram air pressure, due to forward velocity of the aircraft, at the air intake scoop. This has a slight supercharging effect so that the engine volumetric efficiency, and hence power developed, is greater than under static conditions.

Fuel for Piston Engines

31. The aeroplane piston engine burns gasoline because this is a fuel with a high calorific value and which will evaporate readily when brought into contact with air at ordinary atmospheric temperatures. The important characteristics of an aviation piston engine fuel are the calorific value of the fuel, its volatility and its anti-knock qualities.

32. The calorific value of the fuel is a measure of the heat energy available in it. Those fuels with a high hydrogen content will produce the most calories of heat energy per unit volume.

33. The volatility of a fuel is its readiness to evaporate. A volatile fuel makes for easy starting, but fuel evaporation can increase the risk of carburettor icing, and can lead to the formation of vapour locks in the fuel supply system. An ideal aviation fuel must therefore be sufficiently volatile to make engine starting easy, but must not evaporate at such a rate that it will vaporise in the fuel system.

34. The anti-knock value of a fuel is a measure of its resistance to detonation. The anti-knock quality of the fuel will determine the pressure to which the fuel-air mixture can be raised before ignition, without detonation occurring. It is therefore one of the factors which determines the maximum power output of the engine.

Types of Fuel

35. The types of fuel ideal for use in spark ignition internal combustion engines are gasoline, alcohol and the aromatic fuels. The latter two, however, have a number of disadvantages that render them less than ideal for use in aircraft engines. Alcohol is incompatible with many of the seal materials used in aircraft engines, causing them to deteriorate rapidly. The expense of using seal materials resistant to alcohol attack does not justify the use of gasoline/alcohol mixtures, even though these may be cheaper than pure gasoline.

36. The aromatic fuels are so called because their molecular and atomic structure is identical with that of perfumes. They are hydrocarbon compounds obtained either from coal or oil. They are present to a limited extent in most gasoline distillations. The best known of the aromatic fuels is Benzol, which has the advantage of being resistant to detonation. Its disadvantages are its ability to dissolve rubber and its very slow burning rate. As a result its inclusion in aviation gasoline is limited to 5 per cent by volume.

37. Other important aromatics are toluene, xylene and cumene. All are rubber solvents, though to a lesser degree than Benzol. Of the three, toluene is the most useful because of its low freezing point and good volatility and it is suitable for blending in aviation gasoline up to 15 per cent by volume.

Avgas

38. Avgas (aviation gasoline) consists of approximately 85% carbon and 15% hydrogen. The air with which it is mixed to form a combustible fluid consists of approximately 78% nitrogen and 21% oxygen. During the burn the hydrogen, carbon and oxygen form carbon dioxide and water vapour, and the nitrogen, which is inert, acts as a buffer in what would otherwise be an explosion rather than a controlled combustion. The result, hopefully, is a rapid but steady burn of the mixture, giving a uniform rise of pressure on the piston. In this controlled situation the flame rate, the rate at which the flame spreads through the mixture, will be in the order of 60 to 80 feet per second.

39. If the burn is not controlled, and the mixture effectively explodes, then detonation is said to occur. This is evidenced by a characteristic knocking of the engine, and causes physical damage to the unit in very short order.

40. Detonation occurs because, although combustion begins normally, the temperature of the unburned part of the mixture becomes so high that it ignites spontaneously, giving a flame rate of 1000 ft/sec or so.

41. The piston cannot absorb the rapid rise in pressure resulting from detonation, and so the energy is wasted as heat, which causes an overall rise in engine temperature, and the formation of local hot spots. Prolonged detonation will burn the piston crown, which in due course will collapse altogether. Detonation will also rapidly ruin the gas tight seals around the valves, and can result in distortion of the cylinder head, the connecting rods and the pistons.

42. Detonation is caused when the temperature and pressure of the charge during the compression stroke reaches a level at which the fuel-air mixture will burn instantaneously (explode). Because of adiabatic heating an increase in pressure will always result in an increase in temperature. The density of the mixture will influence both pressure and temperature, and the ratio of fuel to air will also be an important factor. In practical terms, therefore, any or all of the following may cause detonation:

(a) Too weak (lean) a mixture.

(b) The ignition too far advanced.

(c) The engine (torque) loading too high for rpm selected.

(d) A high charge temperature, due to inappropriate use of carburettor heat, or of alternate hot air in a fuel-injected system, or of excessive supercharging.

(e) The selection of high power at low rpm, with a constant speed propeller.

(f) Too low a fuel octane rating.

43. Detonation should not be confused with pre-ignition, although detonation may follow as a result of pre-ignition. If an engine is allowed to become overheated, or if the combustion chamber is excessively carbonised, the temperature of some projection in the combustion chamber, such as the sparking plug points or a piece of carbon deposit on the cylinder head, may rise so much that it glows red hot. This is known as a 'hot spot' and is liable to ignite the mixture before the spark occurs during the compression stroke. In this condition the engine may continue to fire with the ignition system switched off, although probably not on all cylinders.

44. With pre-ignition there is a loss of power, rough running, and further overheating. The higher the engine speed the worse pre-ignition will become, whereas detonation almost always diminishes with an increase in rpm. Pre-ignition is avoided by use of the correct fuel and by operation within the correct limits of MAP and cylinder head temperature.

Octane Rating

45. The anti-knock qualities of a fuel are broadly expressed in the octane rating. The octane system of classifying fuel is based on the ratio at which two pure hydrocarbon spirits, normally found in gasoline, are mixed together to give a certain temperature at which detonation will occur. The two spirits are isooctane, which has a very high resistance to detonation, and heptane, which has a very low resistance to detonation. Thus, a fuel with an 85 octane rating is one which shows similar anti-knock characteristics to a mixture comprising 85% isooctane and 15% heptane.

46. The current trend is towards low lead fuels, and this presents a problem to engine designers since it is the lead in petrol which tends to prevent detonation. Clearly in an engine with a high compression ratio, the anti-knock qualities of the fuel used must be high. This is especially so if the engine is turbo-charged or supercharged, since in either case the fuel-air mixture will already be at a relatively high pressure on reaching the inlet manifold. If conditions exist where detonation is likely to present a problem then tetra-ethyl-lead (TEL) may be added to the fuel as an anti-knock agent. The addition of TEL can raise the octane rating of the fuel to a value greater than 100, at which point it becomes correctly referred to as a performance number.

47. Since the anti-knock qualities of a fuel depend upon the fuel/air mixture ratio it used to be the practice to assign two performance numbers to aviation gasolines, these being the ratings for a lean and a rich mixture. Examples of this practice were 100/130, 115/145 and 80/87 octane fuels.

48. The addition of TEL is accompanied by ethylene dibromide, which prevents the lead from oxidising onto the various harmful components within the combustion chamber.

Fuel Grades

49. Aviation gasoline (Avgas) is only generally available in one grade, known as Avgas 100 LL (low lead). It is coloured blue, has a specific gravity of approximately 0.72, and must comply with Directorate of Engineering Research and Development (DERD) specification 2485. 100 LL is approximately equivalent to the old grade of 100/130. It contains a maximum of 2 millilitres of TEL per gallon.

50. There are two other categories of Avgas that may be available. Avgas 100 is coloured green and has similar anti-knock characteristics to 100 LL, but contains more tetra-ethyl-lead (TEL) than 100 LL and is therefore considered to be environmentally unfriendly. It is the nearest equivalent fuel to the old 115/145 rating. Avgas 80 is coloured red and equates approximately to the old octane rating of 80/87.

Water Content

51. Water in aviation gasoline presents a serious hazard for two main reasons. The first of these is that it will block the small jets in the carburettor if present in all but minimal quantities. Such blockage will inevitably result in engine failure. Secondly, at low temperatures water in the fuel will precipitate and form ice crystals which will block fuel filters and obstruct the flow of fuel to the engines.

52. Whilst great efforts are made to ensure freedom from water pollution in aviation gasoline storage and distribution facilities, condensation will readily form in partly filled aircraft fuel tanks during times when atmospheric temperature changes are taking place. To prevent this it is important to ensure that aircraft tanks are kept filled, especially in aircraft parked overnight. The maximum permissible water content in aviation gasoline is approximately 30 parts per million.

Alternative Fuels

53. The virtual non-availability of Avgas 80, or 80/87 octane, has created a problem for the owners and operators of aircraft with piston engines designed to use this grade of fuel. In many European countries, including the UK, the authorities have permitted limited use of automotive gasoline, provided that it conforms to certain specifications. This fuel is known as Mogas.

54. Automobile gasoline (Mogas) may be available, but its use in the UK implies limitations specified in CAA Airworthiness Notices 98 and 98A. It is more volatile than Avgas and thus more prone to cause vapour lock in fuel lines and pumps, especially at high temperatures and/or altitudes. It does not possess the anti-knock qualities of Avgas and may lead to pre-ignition and detonation in some aircraft piston engines. Detergents used in Mogas may be harmful to aircraft fuel system components and alcohol, found in some grades of Mogas, will cause deterioration of the seals in most aircraft fuel systems, with the obvious hazard to flight safety.

55. Most major aircraft manufacturers have refused to approve the use of Mogas and using it may invalidate the engine manufacturer's guarantee. CAA Airworthiness Notice Number 98 permits its use in certain light aircraft/engine combinations, but stipulates detailed conditions of fuel specification, source of supply, storage and operating precautions. In particular, in order to minimise the risk of vapour lock, the temperature of fuel in the aircraft tanks must be less than 20°C and the aircraft must not be flown above 6000 feet. Because of the higher volatility of the fuel (and possible water content) carburettor hot air systems must be functional and operated, since the risk of carburettor icing is greater than with Avgas.

56. Any pilot considering using Mogas should first consult CAA Airworthiness Notice Number 98 and CAA General Aviation Safety Sense Leaflet Number 4.

The Mixture

57. A fuel-air mixture will be combustible over a range of mixture ratios, varying between around 8 parts of air to 1 of fuel (a very rich mixture), to around 20 parts of air to 1 of fuel (a very lean mixture).

58. A mixture ratio of 15:1 is most likely to give complete combustion, and this ratio is termed the chemically correct mixture. It is at this ratio that all of the carbon and hydrogen in the fuel combines with the oxygen of the air to form carbon dioxide and water vapour.

59. Unfortunately, at the chemically correct ratio, the ignition temperature is too high and, apart from the risk of detonation occurring, a loss of power results from a process known as dissociation. This dissociation occurs because the various products of combustion split momentarily into their separate elements. This process absorbs heat, and this decreases the combustion pressure. Although the various elements recombine later on in the power stroke, and the lost heat is regained, this comes too late in the process to be of any real value.

60. If a mixture that is 10% richer than the chemically correct mixture described above were used, the excess fuel would absorb sufficient latent heat as it vaporised to make detonation extremely unlikely and to prevent dissociation.

61. Because of the difference in inertia between fuel and air there is a problem in supplying each engine cylinder with mixture at the same ratio. Whilst the aircraft is in the cruise it is therefore normal to produce a mixture at the carburettor or fuel injection unit which is 15% richer than the chemically correct value. This ensures that the leanest cylinder receives a mixture that is on the rich side of the chemically correct value.

62. During the periods where the engine is working continuously at high rpm, such as in the climb, it is normal to further enrich the mixture to counteract any tendency towards detonation as the working temperatures increase under the heavy workload. The degree of enrichment may be as much as 20% or 30% over the chemically correct value, depending on the particular engine that is employed.

63. The excess fuel that is supplied to the cylinders, as discussed above, acts as a coolant but does not produce extra power. In order to increase power it is necessary to increase the mass of fuel-air mixture, not simply to increase the mass of fuel within that mixture. The unburned fuel will leave the exhaust manifold as fuel vapour, or will combine with oxygen to form the deadly carbon monoxide gas.

64. Strangely enough, a cooler burn can be achieved not only with mixtures that are richer than the chemically correct value, but also with mixtures that are weaker. This occurs because the rate of burn (the flame rate) is slower with a weaker mixture, because of the high levels of nitrogen that are present. With very weak mixtures the burn may take so long that combustion is still occurring after the end of the ignition stroke. This will obviously shorten the useful lives of the valves.

65. The slow burn associated with weak mixture results in a decrease in power output, but also a decrease in fuel consumption measured in gallons per hour. If a mixture having approximately 80% of the strength of the chemically correct value is used, the engine's best specific fuel consumption (SFC) is likely to be achieved. The SFC is a measure of the amount of fuel used for each unit of distance flown in still air. The lowest SFC will therefore give the most economical flight conditions.

66. The final problem occurs at idling rpm. As discussed earlier, the exhaust valve remains open during the initial part of the induction stroke. At idling rpm the exhaust gas velocity is low, and there is a tendency for some of the exhaust gases to be sucked back into the cylinder, thereby weakening the mixture for the subsequent cycle. In order to overcome this problem it is therefore necessary to progressively enrich the mixture as idling rpm is approached.

Engine Priming Systems

67. In order to avoid unnecessary engine cranking when starting a cold engine, a quantity of neat fuel may be supplied to the induction manifold, so that a rich fuel/air mixture is drawn into the cylinders when the engine starts to rotate. Depending upon the type of carburettor the fuel may be supplied by one of the following methods.

(i) Some float carburettors are fitted with a carburettor tickler. This is a manually operated plunger which may be used to force down the carburettor float. This action allows the fuel level in the float chamber to rise, and eventually fuel will flow from the discharge nozzle into the induction manifold.

(ii) Where carburettors are fitted with a throttle-operated accelerator pump, the action of opening the throttle will result in fuel being sprayed into the induction manifold.

(iii) Where a seperate fuel primer is installed, a priming pump (manually or electrically operated) will draw fuel from a fuel tank and discharge it to various points in the induction manifold via a system of priming pipes and nozzles. To operate a manual priming pump it is usually necessary to twist the plunger in order to release it. It then springs outward, opening the supply valve. Failure to re-lock the plunger will result in the engine drawing excess fuel through the open supply valve on start-up. The engine will stall after a short period due to over-enrichment (flooding).

(iv) With fuel injected engines priming is usually accomplished by operating the fuel booster, or auxillary, pump for a brief period with the mixture control set so as to permit a flow of fuel to the injector nozzles.

68. In order to avoid flooding the engine with neat fuel, a drain is fitted to the lowest point in the induction or supercharger casing so that any surplus fuel that may have collected will pass through this drain.

Fuel Sampling

69. Before flight each day a small quantity of fuel should be drawn from each of the aircraft tank drain valves and inspected in a glass container. Fuel should be considered unfit for aircraft use if visual inspection shows:

(i) More than a trace of sediment

(ii) Globules of water

(iii) Cloudiness

(iv) A positive reaction to water finding paste, or a paper or chemical detector.

The following should serve as a guide to the visual assessment of aircraft fuels:

(i) **Colour.** AVGAS is dyed blue, AVTUR JET A-1 is undyed and can vary in appearance from colourless to straw yellow.

(ii) **Undissolved water** (free water) will appear as droplets on the sides of the sampling vessel, or as bulk water in the bottom.

(iii) **Suspended water** (water in suspension) will cause the fuel to appear hazed or cloudy.

(iv) **Solid matter** (rust, sand, dust, scale, etc) may be suspended in the fuel or settle on the bottom of the sampling vessel.

(v) The fuel sample should appear clear and bright. Clear refers to the absence of sediment or emulsion and bright refers to the sparkling appearance of fuel which is free from cloudiness or haze.

70. Because of its lower SG, fuel floats on water. If there is no obvious separation of fluids, test the sample by smell, colour and finally by evaporation test. Empty the sample onto a smooth surface and see how quickly it evaporates. The fluid in the container could be all water !

Piston Engine Power Transmissions and Propellers

Piston Engine Power Transmissions and Propellers

Introduction

1. The purpose of the propeller is to convert the rotary power, or torque delivered by the engine crankshaft into propulsive power, or thrust to drive the aircraft through the air. In the case of all but light single engined aircraft one form of rpm reduction is necessary between the engine and propeller.

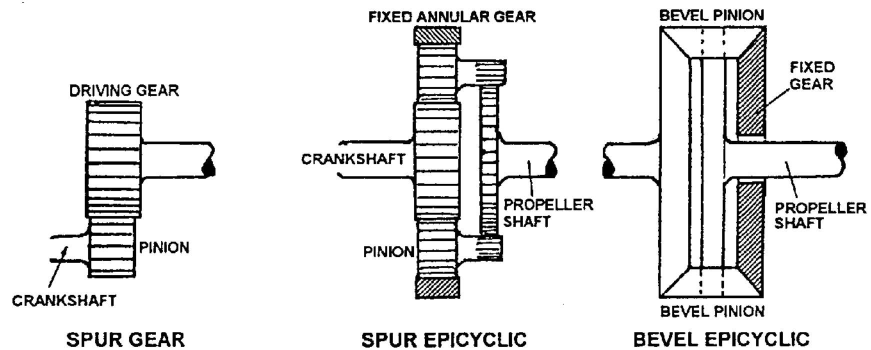
Propeller Reduction Gearing

2. In order to achieve optimum power output, a piston engine ideally needs to operate at a crankshaft speed which is well in excess of efficient propeller speeds.

3. The propeller reduction gear reduces the engine crankshaft speed to the speed suitable for efficient operation of the propeller. Epicyclic or planetary reduction gears are always used on radial engines whilst spur gear reduction gearing is generally used on in-line engines. Any of the above may be used on horizontally opposed engines.

4. Roller bearings will normally support the propeller shaft, while a ball or roller thrust bearing transfers propeller thrust to the airframe. Occasionally plain bearings and a thrust washer perform these functions. Lubrication is normally from the engine lubricating oil system. Three types of reduction gear are illustrated at Figure 17-1.

FIGURE 17-1

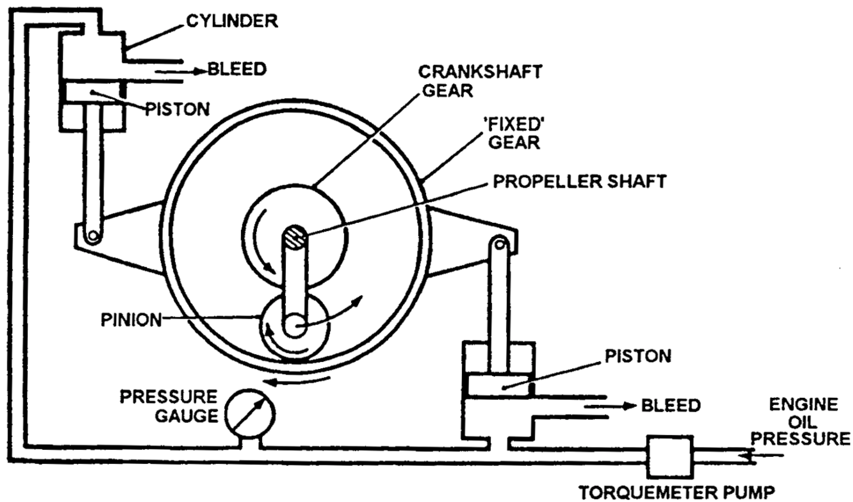


Torquemeters

5. As the crankshaft gear rotates it drives the propeller pinions which exert a thrust on the fixed gear teeth. This fixed gear will tend to rotate in the opposite direction to the crankshaft fear, and the thrust produced will be directly proportional to the power output. In order to measure the thrust which is applied to the fixed gear, the fixed gear is allowed to float, being attached to the structure via pistons and oil-filled cylinders, as shown at Figure 17-2. Engine oil pressure, boosted by a torquemeter pump, is fed to the cylinders and each cylinder is fitted with a bleed back to the engine oil system.

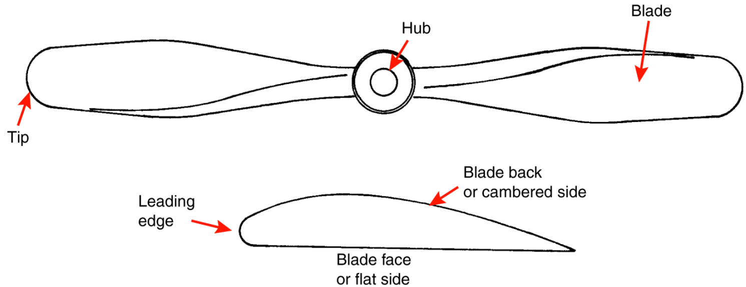
6. At low engine power, thrust on the fixed gear is at a minimum. The bleed ports are fully open with a resulting low oil pressure in the system. Consequently the torquemeter pressure gauge reading will be low. As power is increased, the thrust on the fixed gear will increase and the pistons will be forced further in to their cylinders. Now the bleed ports will be partially covered and the oil pressure on the pistons will be increased to match the thrust on the fixed gear. The torquemeter pressure gauge is usually calibrated to read BMEP (brake mean effective pressure).

FIGURE 17-2



7. The aircraft propeller consists of two or more blades mounted on a central hub, which is attached to the engine crankshaft. As a general rule piston engine propellers have a maximum of four blades. A typical two-bladed propeller is illustrated at Figure 17-3, with a cross-section through the blade indicating the standard propeller blade nomenclature. It will be seen that the blade has an aerofoil section similar to that of a wing. As the blade rotates the aerofoil creates lift in the same way as a wing, but since the plane of rotation of the blades is perpendicular to the aircraft's longitudinal axis this lift acts longitudinally and is referred to as thrust.

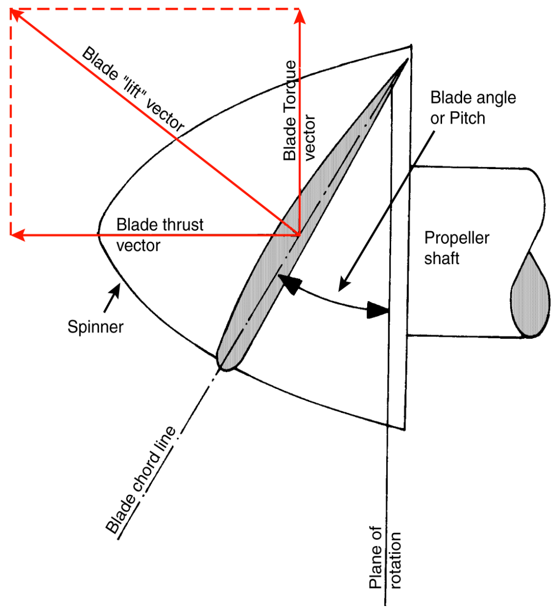
FIGURE 17-3
Propeller Blade Nomenclature



Fixed Pitch Propeller

8. The pitch of a propeller is determined by the angle made between the chord line of the propeller blade and the plane of rotation. This angle is known as the geometric pitch angle or, more commonly, the blade angle and is illustrated at Figure 17-4.

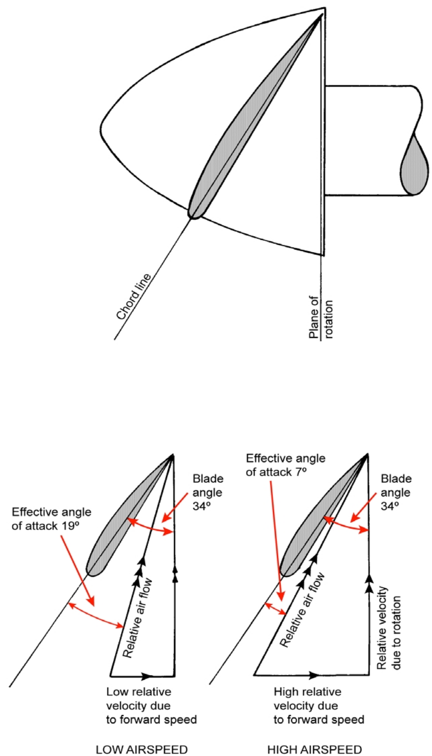
FIGURE 17-4
Basic Propeller Geometry



9. A fixed pitch propeller is one in which the blade angle is fixed and cannot be changed whilst the propeller is rotating. The thrust developed by the propeller blades will depend upon the speed of rotation of the propeller and the angle of attack of the blades. From Figure 17-5 it will be seen that the angle of attack of the propeller blade is dependent upon three factors. These are the blade angle, the speed of rotation of the propeller and the forward velocity of the aircraft.

FIGURE 17-5

Fixed-Pitch Propeller Angle of Attack



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10. The blade angle is, by definition, fixed so increasing the forward speed of the aircraft with the propeller rotating at constant speed (rpm) will cause the effective angle of attack of the blades to decrease. Conversely, increasing propeller rpm will cause the blade angle to increase. There is only one angle of attack at which any aerofoil is at its most efficient, namely about 4°.

11. It follows therefore that, at a given propeller rpm, there is only one airspeed at which the propeller will be at maximum efficiency. At lesser angles of attack it will produce less thrust, but blade drag will be lower so less torque is necessary from the engine. At greater angles of attack the blades will produce more thrust, up to the stalling angle of about 15°, but blade drag will increase and the engine torque needed to maintain rpm would be progressively greater.

12. For reasons of economy, most fixed pitch propellers are designed to operate at maximum efficiency when the aircraft is in the cruise. At lower speeds the effective angle of attack of the blades is greater and the propeller is operating at less than full efficiency. The option to increase rotational speed (rpm), and reduce the effective angle of attack is limited because of the extra torque required. Furthermore, at the low forward speed of take-off or climb the propeller must clearly be capable of producing thrust and therefore the effective angle of attack of the blades must not exceed 15°.

13. The available range of flight speeds for an aircraft with a fixed pitch propeller is therefore clearly limited by the range of effective angles of attack of the blades. Since the useful rpm range of a piston engine is also very limited, to increase the range of possible flight speeds the only remaining variable is the blade angle, or pitch of the propeller blades. Hence the development of the variable-pitch propeller.

Variable Pitch Propellers

14. There are a number of types of variable pitch propeller, but the aim in all cases is to achieve maximum propeller efficiency over as wide a range of aircraft speeds as possible. By this means the conversion of engine torque into thrust is achieved with the greatest efficiency over a range of airspeeds instead of only at one specific airspeed.

Ground-Adjustable Propeller. In this type of variable pitch propeller the blades can be rotated in the hub, using special tools, to alter the blade angle. This can only be done on the ground with the engine shut down. The system offers a very limited range of possible blade angles, by which the economical cruising speed of the aircraft can be varied slightly. It is a very early attempt at variable pitch propellers and is rarely found nowadays.

Two-Position Propeller. With this type of propeller the blade angle can be changed by the pilot in flight to give either a low blade angle (low or fine pitch) for low speed flight (e.g. take-off), or a high blade angle (high or coarse pitch) for high speed flight. Hence the propeller has two settings at which it achieves maximum efficiency and a reasonable range of flight speeds over which it achieves acceptable efficiency. One of the worst aspects of the system is the danger of selecting coarse pitch for take-off, whereupon the aircraft is incapable of achieving flight speed before it runs out of runway. Two-pitch propellers were superseded by controllable-pitch propellers, of which the most popular for use with piston engines is the constant speed propeller.

Constant Speed Propeller. The constant speed propeller utilises a governor, situated in the propeller hub, to maintain constant propeller rpm regardless of engine throttle setting or airspeed. The governor achieves this by altering the propeller blade angle, to maintain the effective angle of attack of the blades constant, through a hydraulic or electric pitch-change mechanism. The setting of the governor-controlled propeller speed is adjustable by means of a propeller lever in the pilot's cockpit. Having selected a desired rpm, anything that tends to alter the propeller speed will result in the governor adjusting the blade angle to maintain constant speed.

15. Suppose for example the aircraft is put into a dive and forward speed increases. We know that this will reduce the effective angle of attack of the propeller blades (see Figure 17-5). The reduced blade torque, or resistance to rotation, will cause the propeller to rotate faster because the torque being applied to it by the engine has not changed. However, immediately the governor senses the increased rpm it drives the blades to a coarser angle, which increases their angle of attack and blade torque. Consequently propeller speed is held constant and the effective angle of attack of the blades is returned to its optimum value.

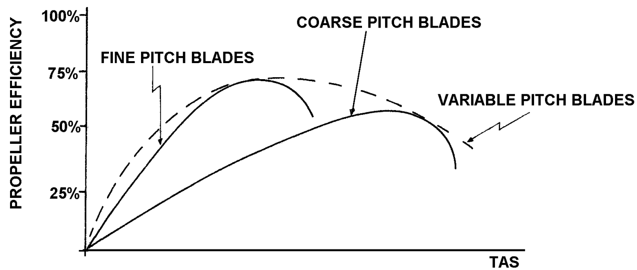
16. Conversely if the aircraft were put into a climb the forward speed would decrease and the effective angle of attack of the blades would increase. Propeller rpm would tend to fall, so the governor would drive the blades to a finer pitch to reduce blade torque and maintain constant propeller rpm. Once again the effective angle of attack of the blades returns to the optimum value

17. Similarly, if the pilot were to open the throttle, increasing engine power, the engine would tend to speed up and drive the propeller faster. The governor, sensing the initial rpm increase, would drive the propeller blades into coarser pitch. Blade torque would therefore increase, preventing any further rise in propeller speed and restoring rpm to the selected constant value. Furthermore, the coarser blade angle would enable the propeller to absorb the additional torque, and airspeed would increase until the blade angle of attack is restored to the optimum angle for maximum efficiency.

18. A reduction in engine power has the reverse effect on blade angle, to maintain constant propeller rpm and optimum angle of attack.

19. Figure 17-6 compares the efficiency with varying airspeeds of a two-pitch propeller and a variable pitch propeller.

FIGURE 17-6
Propeller Efficiency



20. With a variable pitch propeller it is possible to incorporate various propeller blade ranges which fall outside the normal fine and course blade angle limits:

(a) In order to prevent the propeller of a failed engine windmilling and therefore creating a great deal of drag, many piston engines or turboprop aircraft are fitted with **feathering** propellers. When the propeller is feathered the leading edge of the blade is more or less **edge on** to the oncoming airflow. In other words the angle of attack is 0° and the pitch angle in the region of 85°.

(b) The limit of fine pitch used for take-off, landing and any flight condition would normally be in the order of 10° to 12°. With the aircraft taxiing (hopefully) very slowly on the ground it is desirable that the blade angle be made even finer, to maintain a sensible angle of attack. It is therefore possible to fit a device which will enable the propeller to be shifted into a **ground manoeuvring range** of blade angles. Obviously this is desirable **only** when the aircraft is on the ground. Were a ground manoeuvring blade angle to be selected in flight it is probable that the angle of attack would be come negatives, and you've invented airborne reverse thrust. For this reason the ground manoeuvring selector is normally guarded by an undercarriage squat switch.

The range of ground manoeuvring blade angles is sometimes known as the **BETA range,** and the BETA range is often taken to include the negative blade angles involved in producing reverse thrust.

(c) **Braking propellers** provide what is effectively reverse thrust. Now the blade angle is made negative and the angle of attack becomes negative. The airflow from the propellers now opposes the direction of travel of the aircraft.

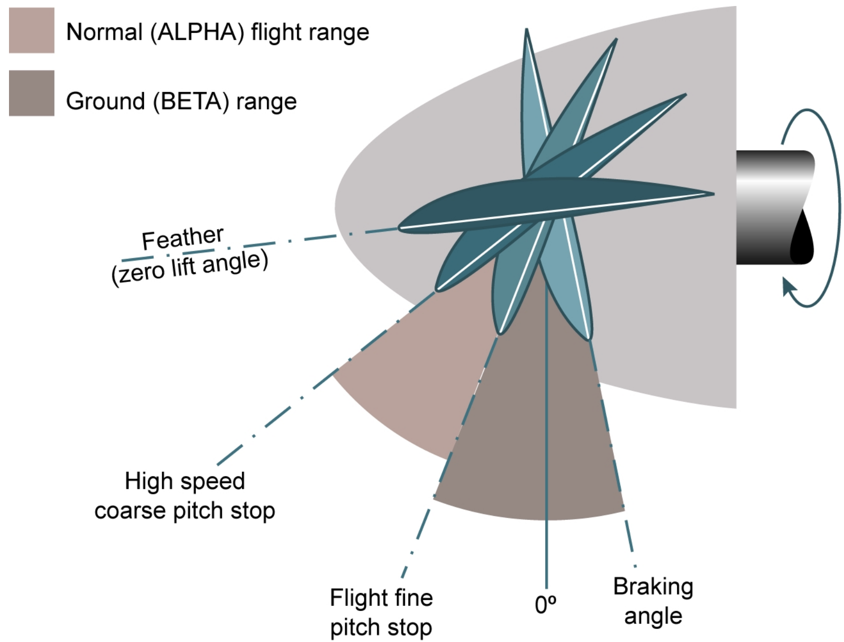
Braking propellers make a lot of noise when reverse thrust is selected and certainly help to decelerate the aircraft. It is important not to confuse the noise level with the level of efficiency. Something like 80% of the aircraft's kinetic energy will still be dissipated through the wheel braking system.

When selecting reverse thrust on a multi-engined aircraft it is not a bad idea to dwell for a second or two at 20% reverse power to ensure that you do not have an asymmetric reverse situation, which will obviously create directional control problems when 100% reverse power is selected. Like the ground manoeuvring selector, the reverse selector is normally protected by an undercarriage squat switch.

Do not confuse a **braking propeller** (one that can provide reverse thrust) with a **propeller brake.** This is a device which, as the name suggests, prevents a feathered propeller from rotating in the airflow.

The diagram at Figure 17-7 shows the blade angle ranges described above.

FIGURE 17-7



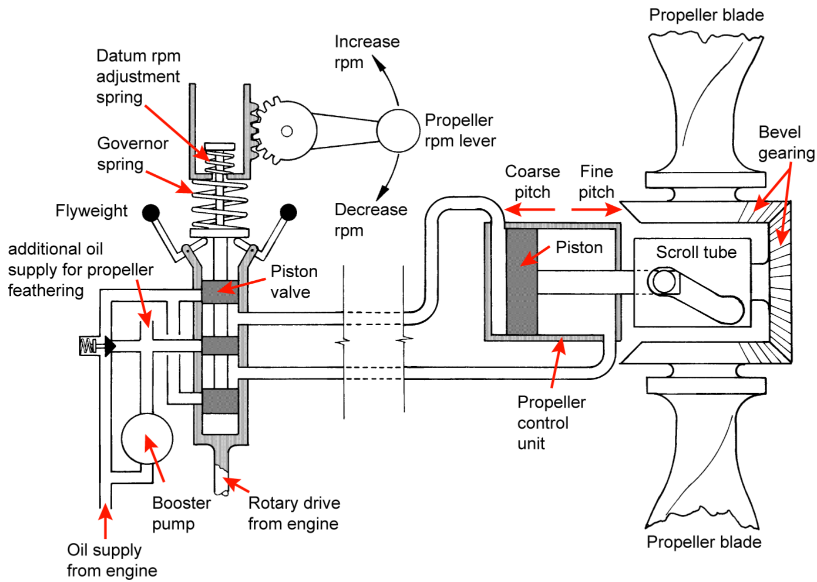
21. The centrifugal turning moment of the blade tends to force the blade towards a low pitch angle. In the event of an engine failing and the propeller not being feathered, the propeller would therefore tend to go to the fully fine position, with a consequent high drag condition.

22. Since the situation described above is obviously undesirable, it is a common feature of many constant speed propeller systems that counterweights or alternative devices are used to ensure that the uncontrolled/unfeathered propeller maintains the fully coarse blade angle, thereby minimising drag.

Constant Speed Units

23. A type of hydraulic constant speed propeller governor is illustrated at Figure 17-8. This is known as a double-acting unit since the governor supplies hydraulic power to the propeller pitch-change mechanism to drive the blades toward either fine or coarse pitch, as necessary. A slightly simpler, single-acting system will be explained subsequently.

FIGURE 17-8
Double-Acting
Pitch Change Unit



24. A piston valve is attached to a spindle, which is positioned axially by either the governor flyweights or the pilot's propeller rpm lever. The piston valve slides in a ported cylinder to which the flyweights are attached and which rotates, driven by the engine. The ports in the rotating cylinder are connected to a pressure oil supply from the engine lubricating oil system, to a return line to the engine sump and to either side of the propeller pitch control mechanism. The piston valve is carefully profiled so that it controls the oil supply to and from the coarse and fine pitch sides of a piston in the Pitch Change Unit (PCU).

25. Because considerable force is needed to move the propeller pitch-change piston against the forces acting on a rotating propeller, a high-pressure oil supply is required and this is achieved by means of a booster pump. This pump is supplied from the engine lubricating oil system and gear-driven from the crankshaft.

26. The governor flyweights are attached to L-shaped arms, known as bellcranks, which are pivoted to lugs at the upper end of the cylinder. The higher the speed of the engine, the faster the governor flyweights rotate and the greater the centrifugal force acting upon them. Under the influence of this force the flyweights move outward from the axis of rotation and the bellcranks lift the piston valve in its cylinder, against the force of the governor spring.

27. If engine speed is reduced the governor spring force overcomes the centrifugal force of the flyweights and the piston valve is moved down in its cylinder.

28. Movement of the pilot's propeller rpm lever may adjust the governor spring force. Reduction of the spring force will allow the piston valve to move up, compressing the spring until centrifugal force matches spring force. Compressing the spring and increasing the spring force will force the piston valve down until the spring has expanded sufficiently to balance the two forces once more.

29. It will be noted that a smaller spring is located above the governor spring. The tension of this spring can be adjusted with the engine stationary to set the rpm datum, when the spring and flyweight forces are in equilibrium.

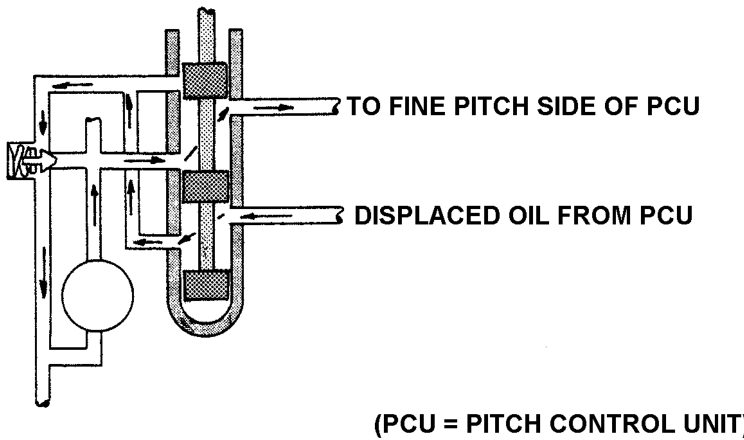
30. Let us now consider the operation of the Constant Speed Unit (CSU) in the three principal operating situations. These are firstly the on-speed condition, when the propeller is operating at the rpm selected by the pilot, secondly the under-speed condition when the propeller rpm is less than that selected and thirdly the over-speed condition, when propeller rpm is higher than that selected.

The on-speed condition. This is the condition illustrated at Figure 17-8. In this situation the propeller is rotating at the rpm selected by the pilot and the centrifugal force exerted by the governor flyweights balances the force exerted by the governor spring. The piston valve is held in mid-position, where its "lands" cover the oil supply and return ports in the cylinder. Oil cannot flow to or from the PCU, and so the PCU piston is hydraulically locked, holding the propeller blades at their present blade angle, which is achieving the optimum angle of attack and ensuring that the propeller is operating at maximum efficiency.

The under-speed condition. This situation will occur for one of two reasons. The first is simply that the pilot has moved the propeller rpm lever, selecting a higher rpm than was previously set. The second situation might occur when the aircraft is put into a climb without adjusting the throttle setting. Initially the propeller would behave like a fixed pitch propeller, and the rpm would decay.

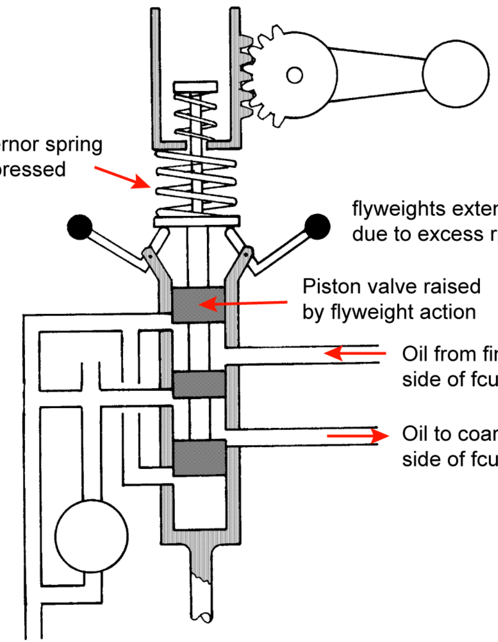
31. The result of this lower rpm would be that the governor spring force would override the reduced flyweight centrifugal force. The piston valve will therefore move downwards, uncovering the oil supply and return ports. This will feed high-pressure oil to the fine pitch line, and allow oil that is displaced from the pitch change cylinder to flow back through the coarse pitch line to the inlet side of the booster pump. This situation is shown at Figure 17-9. The propeller will now move to a finer pitch, which will result in the desired increase in rpm. As rpm reaches the desired value the governor flyweights return the piston valve to its mid, or on-speed position

FIGURE 17-9
Under-Speeding Propeller



The over-speed condition. This situation is the reverse of that described in the under-speed case. Now the governor flyweights exert more force than the throttle spring. The piston valve moves upwards and oil pressure is supplied to the coarse pitch side of the PCU piston. The fine pitch side of the piston is connected to the inlet side of the booster pump and the pressure difference across the PCU piston moves it to change the propeller blade angle to a coarser pitch. The propeller rpm consequently falls until flyweight/spring equilibrium is restored and the on-speed condition is regained. This is illustrated at Figure 17-10.

FIGURE 17-10
Over-Speeding Propeller



Propeller Feathering

32. When the engine is not driving the propeller of an aircraft in flight, the airflow over the aerofoil section blades creates a lifting force causing them to rotate in the normal direction of rotation, exactly like a child's toy windmill. The finer the pitch of the propeller blades, the higher the windmilling rpm of the propeller. This is a highly undesirable situation, since the windmilling propeller creates a great deal of drag. Furthermore, the propeller driving a failed engine may well lead to greater damage, overheating and even fire.

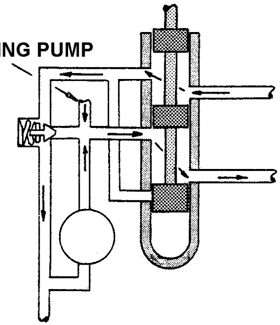
33. To avoid these problems a further operating condition is possible with many constant speed propeller control units, known as the feathering condition. This permits the propeller blades of a failed or stationary engine to be moved through coarse pitch to a position where their leading edges face into the air stream and the airflow over them creates no lift, so that they do not rotate.

The feathering position. To feather the propeller the propeller rpm lever is moved through the low rpm (fully coarse) position. This physically lifts the piston valve, completely overriding the rpm spring and governor weights and allowing the booster pump to drive the PCU piston to the full extent of its travel in the coarse pitch direction.

34. At the same time the engine throttle is closed, to shut down the engine. As propeller/engine rpm decays the output from the engine-driven booster pump falls, so an alternative feathering oil pressure supply is provided to maintain oil pressure until the propeller is fully feathered. This supply may be from an electrically driven pump or an accumulator.

35. The CSU with feather selected is illustrated at Figure 17-11.

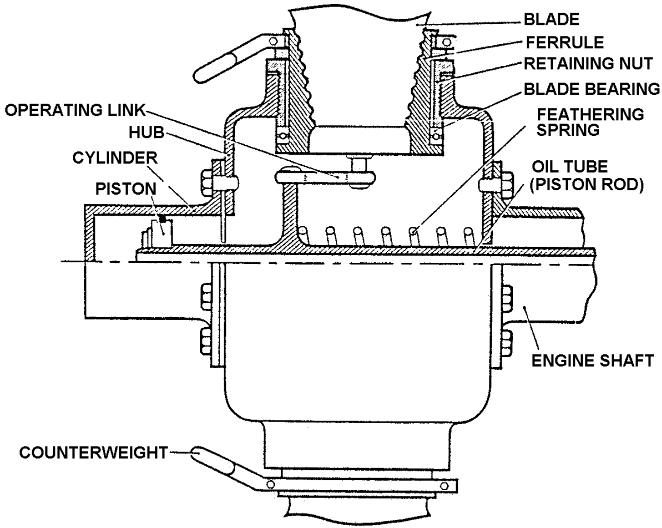
FIGURE 17-11
Feathered Propeller



36. The constant speed propeller system described in the foregoing text is a double-acting unit, in which oil pressure is used to drive the PCU to coarse or fine pitch. Many aircraft employ single-acting propellers in which oil pressure from a governor drives the blades toward fine pitch, but the blades automatically move toward coarse pitch under the influence of a strong spring.

37. A single acting propeller control unit mechanism (as fitted to many light and medium sized aircraft) is shown at Figure 17-12.

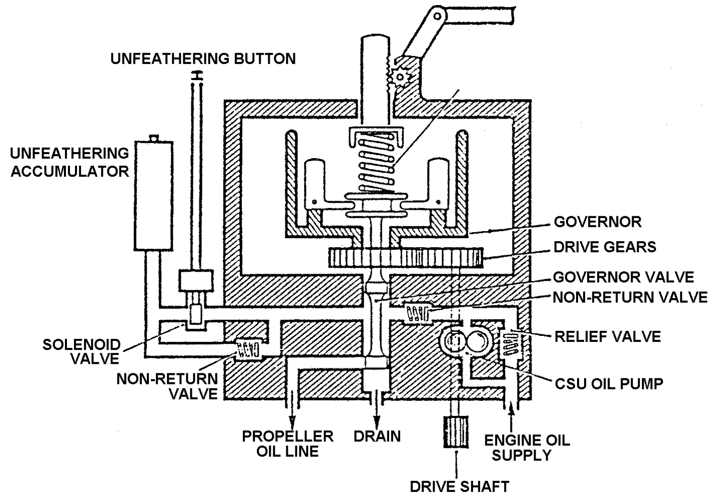
FIGURE 17-12
Single-Acting Pitch Change Unit



38. The PCU and its associated CSU are termed single acting since oil is responsible for changing the propeller pitch angle in one direction only. The PCU cylinder contains a piston, and oil under pressure is fed through the centre of the piston rod to the front face of the piston. The piston and rod move axially under the influence of oil under pressure to change the angle of the propeller blade. As oil is introduced, the piston moves to the rear of the cylinder, and the propeller blades move to a finer pitch angle. In order to achieve a coarser pitch, oil pressure is released from the PCU cylinder and the spring plus the counterweights force the blade to a coarser angle. This system has the advantage of ensuring that the propeller will automatically go to the fully coarse or feathered position in the event of loss of the high-pressure oil supply.

39. The constant speed unit governor mechanism for the single-acting propeller is simpler in construction and operation than the double-acting unit, since it only has to control the supply and return of oil on one side of the PCU piston. Such a unit is illustrated at Figure 17-13.

FIGURE 17-13
CSU Governor for Single-Acting Propeller



40. The CSU at Figure 17-13 controls the propeller blade angle via the centrifugal governor flyweights, the governor valve, and the engine-driven CSU oil pump which boosts engine oil pressure to a suitable level. The centrifugal force which moves the governor fly weights is opposed by the control spring, while the datum setting or loading of the control spring is set by the position of the pilot's propeller control lever. The position of the governor valve is determined by the balance of forces achieved between the centrifugal force of the governor flyweights, and the force exerted by the control spring. When the two forces are in balance, the propeller oil line is closed by the governor valve, and oil pressure is trapped in the cylinder ahead of the piston (see Figure 17-12).

41. During normal operation, there are several different situations to be considered.

On the ground. Here the pilot's propeller control lever will be set to maximum or full increase rpm, while the throttle will be set to low power. The governor valve will be fully down (see Figure 17-13), and therefore the oil from the CSU oil pump will be directed under pressure to the forward side of the piston. This pressure overcomes the pressure of the feathering spring, forcing the piston into its rearward position, and therefore the propeller blades to the fully fine setting.

42. As the power and rpm are increased, centrifugal force acting on the weights will progressively lift the governor valve until, eventually, a point is reached where the governor valve is blocking the flow of oil into the propeller oil line. The propeller is now rotating at maximum rpm, and any increase in this rpm will cause the governor valve to lift still further, allowing oil to flow from the pitch control cylinder. The feathering spring will now push the piston forward in the cylinder, and the propeller will move to a coarser blade angle, thereby preventing any further increase in propeller rpm.

The on-speed condition. As with the double-acting propeller system, in this condition the force exerted by the governor fly weights is balanced by the force of the control spring, the governor valve blocks the inlet to the propeller oil line, and the status quo is maintained.

43. Should the pilot change the desired rpm setting (move the position of the propeller control lever), the system will respond to re-establish the status quo. Assume that a lower rpm is selected (the propeller control lever is moved rearwards). The loading on the control spring will be reduced, and this will allow the governor weights to lift the governor valve. Oil will flow out of the pitch control cylinder, the blade angle will coarsen under the influence of the feathering spring, and the rpm will decrease, until the balance between the governor weights and the control spring is re-established.

The under-speed condition. If the propeller loading increases because of say, a decrease in airspeed, the rpm will initially decrease. The governor valve will descend to admit oil under pressure into the PCU cylinder to drive the blades to a finer pitch and thus increase the rpm to the original setting.

The over-speed condition. This is the exact opposite of the under-speed condition described above. The governor valve will rise to allow a controlled escape of oil pressure from the PCU cylinder and the feathering spring will move the propeller blades to a coarser pitch, reducing rpm to the desired value.

The feathering position. When the propeller lever is moved fully rearward, the governor valve is lifted fully up, in effect negating the imbalance between the governor weights and the control spring. Oil flows out of the pitch control cylinder as the piston moves forward under the influence of the feathering spring, and the propeller blades travel through coarse pitch to the fully feathered position.

44. The high drag of a windmilling propeller is particularly undesirable in a multi-engine aircraft, where it will inevitably give rise to asymmetric yawing forces. In twin-engine aircraft this asymmetric yaw may be so severe as to render the aircraft difficult or even impossible to control in certain situations. Consequently, feathering propellers are essential in multi-engine installations, but are rarely found in single-engine aircraft.

Reverse Thrust

45. A propeller will produce reverse thrust (i.e. thrust in a rearward direction) whenever it is at a negative angle of attack. Perhaps the simplest example of this is when the aircraft is stationary on the ground (no forward velocity), and the propeller blades are moved to a negative blade angle. This produces a negative angle of attack and thrust will be produced in a rearward direction, enabling the aircraft to reverse taxi. This capability is a feature of many gas turbine powered propeller aircraft (turbo-props), but is rare nowadays in piston engined aircraft. Apart from one or two highly specialised stunt examples, it is not a feature that is commonly found in single-engine aircraft.

46. Some aircraft, whilst not having a reverse capability, do have the capability to use, on the ground, a very fine blade angle, often known as ground fine or superfine. With the blades in such a position the thrust produced will be minimal. This may be useful whilst taxying, particularly downhill, and it may also be used to minimise propeller torque on start-up or to increase drag during the landing run.

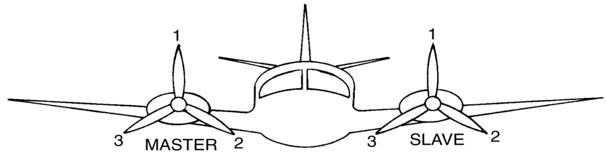
47. With either a superfine only or full reverse capability fitted, they may only be used on the ground. In the air the propeller may only be driven to the flight fine position, and flight fine pitch stops are fitted in the hub. The range of movement available in the air is known as the alpha range. On the ground, any blade angle less than flight fine is known as the beta range. The beta range is usually selected via a gate in the propeller rpm lever or a reverse gate in the power lever.

Synchroniser and Synchrophaser Systems

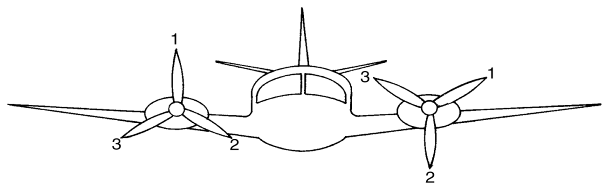
48. To ensure balanced thrust it is clearly desirable that the propellers of a twin-engine aircraft rotate at the same speed. Since constant speed propellers are continuously subject to small adjustments from their CSU's, if the two propeller control systems were acting independently, the two propellers would rarely be operating at exactly the same speed. To overcome this problem, some twin-engine aircraft with constant speed propellers are fitted with an automatic synchroniser system to ensure that both propellers are always operating at the same rpm. In such a system one propeller is selected as Master and the other is slaved to it.

49. To avoid vibration due to out-of-phase frequencies it is necessary to ensure that the two propellers are rotating in phase with each other. This usually means ensuring that the relative blade positions of the two rotating propellers, when viewed from ahead or behind, remains the same. This is illustrated at Figure 17-14.

FIGURE 17-14
Synchrophased Propellers



PROPELLER BLADES IN PHASE



PROPELLER BLADES OUT OF PHASE

50. Many twin-engine aircraft employ a combined control system that accomplishes both synchronising (matching propeller rpm) and synchrophasing (maintaining the relative phase angle of the propeller blades) which also minimises propeller and hence flight deck noise.

Propeller Checks

51. Aircraft propellers must be regularly checked for signs of damage in the form of indentations or gouges that could lead to out-of-balance forces and vibration, or cracking that could cause stress failure.

52. Wooden Fixed Pitch Propellers are easily damaged by stones and other hard objects, and so should be inspected frequently for nicks, cracks, delamination, scores, breaks in the surface finish, and for security of the leading edge sheath. Deep cuts and deep damage must be removed and an insertion repair carried out. If inner trailing edge or tip damage is removed by sanding to a new profile, then the other propeller blade should be reshaped in a similar manner. Small longitudinal cracks and minor indentations may be repaired by using a mixture of sawdust and glue as filler and sanding smooth.

53. Indentations and scores in metal Fixed Pitch Propellers will cause stress concentrations, which in the end will cause failure. This type of propeller needs to be inspected for nicks, dents, cuts and corrosion or for any other surface damage. Minor repairs may be carried out using a smooth file and emery cloth to remove metal from the damaged area. The result should be a smooth and shallow depression.

54. In Variable Pitch Propellers, cuts or gouges which may lead to cracks should be blended out as soon as noticed, though dents and minor erosion may be left until the propeller is removed.

55. Metal Propeller blades that are twisted, cracked, bent or have severe surface damage must be considered unserviceable. Any damage found on the inner third of a propeller blade, or cracks on any part of the propeller, must be considered as immediate cause for removal of the propeller. Damage on the middle third of the propeller blade may be blended out, while damage on the outer third of the propeller blade may be removed by using a hacksaw.

Checking Propeller Track

56. It is important that the blades of a propeller all rotate, or track, in the same plane. Blades that are out of track do not produce equal thrust and the thrust imbalance causes propeller induced vibration. On an installed propeller blade tracking can be checked statically by manually rotating the blade tips past a fixed reference point and checking that clearance from it does not vary by more than a limited amount, typically about 1½ mm. It is usual to put variable pitch blades into fine pitch whilst carrying out the check.

57. A dynamic check of blade tracking can be made with the engine running and using a strobe light to illuminate strips of reflective tape placed accurately on the tip of each blade.

Checking Propeller Balance

58. It is also important that the propeller blades are all balanced in terms of weight distribution to avoid out-of-balance vibration. This is checked statically with the propeller removed from the aircraft and placed on a balance stand. Propeller balance can be checked dynamically with the propeller installed by devices that sense the amplitude and phase of vibration when the propeller is rotating at various speeds.

59. To take a simple example, suppose one blade of a two-bladed propeller is slightly heavier than the other. When the propeller is rotating this will cause an out-of balance force that gives rise to a once-per-revolution vibration. An accelerometer in a fixed location, sensitive only to radial movement, is used to measure the amplitude of the vibration and a strobe light or magnetic pick-up is used to measure blade phase angle. By matching phase angle to vibration frequency it is possible to determine which blade is causing the vibration.

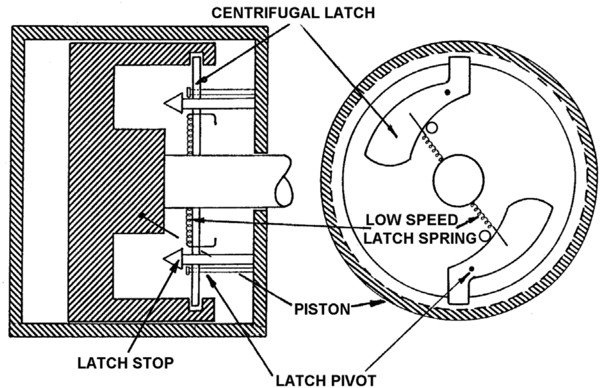
Aircraft and Engine Protection

60. A number of devices are associated with constant speed propellers to protect the engine against overspeeding, which could occur if propeller pitch were allowed to become too fine, and overtorquing, which could occur if pitch were allowed to become too coarse. These pitch-limiting devices are grouped together under the heading of pitch locks.

61. Similarly, aircraft handling must be protected by ensuring that the propeller of a failed engine either feathers automatically, or can be quickly and easily feathered by the pilot. For example, we have seen how the feathering spring of a single-acting PCU will automatically drive the propeller blades to the feather position in the event of governor oil pressure failure.

62. However, when the engine is shut down on the ground, the pressure in the cylinder will slowly bleed away as a result of leakage through the CSU. Thus the propeller will slowly reach the feathered position under the influence of the feathering spring. Since this could well cause unacceptably high loads on the engine during the subsequent start, a centrifugal latch is fitted in the pitch control cylinder to prevent the piston moving to the feathered position, as illustrated at Figure 17-15

FIGURE 17-15
Propeller
Centrifugal Latch



63. When the engine is shut down, or when the engine is at ground idle (the beta range), the latch weight tension springs position the latch weights in such a way that the piston can only be moved forward a short distance, holding the blades at ground fine pitch. With increasing rpm, centrifugal force will disengage the latch weights from the latch stops to allow the piston full range of movement from fully fine to feather pitch.

64. It should be noted however that this system has serious potential hazards in flight. In the event of an engine failure caused by a major mechanical fault (for example, seizure due to loss of oil), the rate at which the engine decelerates may be rapid. Under these circumstances it is imperative that the pilot takes immediate action to feather the propeller. Should he fail to do so before the rpm falls into the latch engagement regime (typically about 1000 rpm) it will be impossible to feather the propeller. The pilot is now faced with a serious asymmetric drag problem that, if it were to occur at the low airspeed associated with climb-out, could prove irrecoverable.

Pitch Stops

65. The constant speed range will not be maintained at very low power settings. Consider a piston engined aircraft, and assume that the rpm selector (the propeller lever) is set to 2,300 rpm and the throttle to 23" Hg (inches of mercury). As the throttle is closed the engine produces less power. The CSU senses a reduction in rpm and fines off the propeller to overcome this. Eventually the PCU piston will contact a fine pitch stop located in the cylinder, and the blade angle will now remain constant. Since the propeller has become effectively a fixed pitch unit, the low power setting must now result in a decreasing rpm.

66. The position of the fine pitch stop in the PCU may be such that it ensures the propeller blade cannot move to such a fine pitch that reverse thrust would result with the aircraft in flight. In this case it becomes known as a flight-fine pitch stop.

67. During ground manoeuvring a finer pitch setting is desirable, for operating efficiently at low taxi speeds and ensuring that maximum taxi speed is not exceeded. The flight-fine pitch stop is therefore withdrawn, either automatically or by pilot selection, when the aircraft weight is on the wheels.

68. Should the aircraft be equipped with reverse-pitch propellers it will be necessary for the ground fine pitch stop to be withdrawn to permit the PCU to drive the blades into reverse pitch. This action is often achieved by moving the propeller lever through a gate, into the reverse range. Similarly, if it is necessary to limit the extent of reverse, or negative pitch a reverse pitch stop may be incorporated in the PCU.

69. The travel of the PCU piston in the coarse pitch direction is similarly limited, either to a maximum coarse pitch to prevent engine overtorquing or to the feather blade angle, typically about 85°. Where a coarse pitch stop is incorporated, it is withdrawn when the pilot selects 'feather'.

Hydraulic Pitch Lock

70. The natural tendency for a windmilling variable pitch propeller is for the blades to move towards fine pitch. Having considered the double-acting CSU, one means of preventing this is its ability to achieve hydraulic pitch lock. This is a system that will be equally effective in the event of a failure of the engine, or of the oil booster pump which supplies the high-pressure oil to the CSU.

71. All that is involved is strategically placed spring loaded valves, which close when the pressure of oil in the feed lines to the pitch control mechanism falls below a pre-set value. This now effectively isolates the cylinder of the pitch control unit. The hydraulic lock ensures that the piston within the cylinder does not move, and therefore that the propeller blade angle does not change.

72. Once the hydraulic lock is established then the propeller has effectively become a fixed pitch propeller. The blade angle, however, is a matter of circumstance rather than design. If the failure occurred in the climb the blade angle would be low (fine pitch), and the rpm would be high. Subsequent control of rpm would be effected with the throttle, and great care must be taken to avoid an engine over-speed condition at high power settings.

Enhanced Propeller Pitch Lock

73. A slightly more sophisticated system, normally only found on larger transport aircraft, for preventing excessive over-speed of a propeller should control be lost through governor failure, is the enhanced propeller pitch lock system. As with the system described above, a propeller pitch lock will form automatically, the pilot cannot control it.

74. As engine rpm increases during take-off, a flyweight-operated valve will close at typically 95% of maximum rpm, in order to prevent a rapid decrease in blade angle. The valve is then offset by the propeller governor to permit 100% of maximum rpm. The bleed shut-off valve will now close if rpm increases by 10% and the governor is no longer controlling the propeller. A hydraulic lock is formed and any further rpm increase is prevented. The bleed valve will remain locked in this position until the propeller rpm falls to some low figure, typically 50% of maximum. Even with the pitch lock engaged the propeller could be feathered if necessary.

Auto Feathering

75. The double acting propeller system can easily be adapted to incorporate auto-feathering devices. The auto-feather sequence is initiated by a torque switch, which activates when the engine torque falls below a pre-set threshold. The torque switch opens a solenoid valve that enables oil from a feathering pump to flow into the CSU, (see Figure 17-8). At the same time it causes the CSU piston valve to be lifted fully, allowing the feathering pump supply of oil to be fed to the coarse pitch feed line, driving the propeller through the fully coarse position to the feathered position.

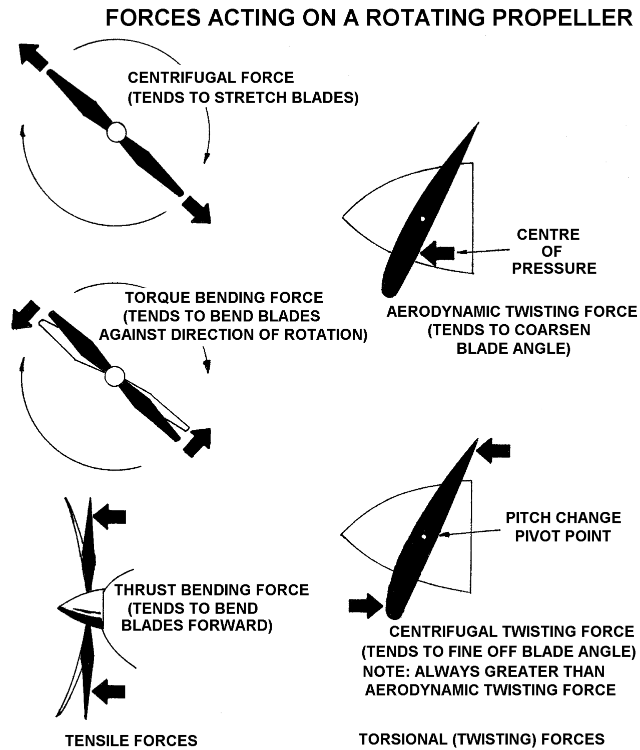
Forces Acting on a Rotating Propeller

76. A rotating propeller is acted upon by centrifugal, twisting and bending tensile forces. Centrifugal force tends to stretch the blades away from the hub, air resistance to rotation applies a torsional force that tends to bend the blades opposite to direction of rotation and the thrust load on the blades tends to bend them forwards.

77. Variable pitch propellers have further forces acting upon them, which tend to alter the blade angle (pitch). Aerodynamic twisting force tends to turn the blades towards coarse pitch, but a much greater centrifugal twisting force tends to move the blades towards fine pitch.

78. These forces are illustrated at Figure 17-16.

FIGURE 17-16
Forces Acting On A Propeller



Piston Engine Operation and Handling

Piston Engine Operation and Handling

Handling Procedures

1. Please note that the following notes are general information and in no way supersede the manufacturer's operating procedures for particular aircraft engines.

Pre-Start Inspection. Ensure the aircraft wheels are chocked, that the propeller arc is clear of equipment and personnel, and that there are no loose objects lying around which may be drawn into the propeller. Check ignition OFF, throttle closed, mixture to IDLE CUT-OFF before connecting ground power. Check parking brake ON.

Starting Procedures (Carburettor Fuel System). The following procedure should be adopted:

(a) Master Switch and boost pump ON.

(b) Throttle approximately ½ OPEN.

(c) Mixture FULL RICH.

(d) Propeller (constant-speed) to FULL INCREASE and ensure arc clear.

(e) Ignition to START - this energises both magnetos and starter motor.

(f) Release ignition switch to BOTH position. Both magnetos energised, starter de-energised.

(g) Check oil pressure rises to normal range within 30 seconds of start-up. SHUT DOWN IF IT DOESN'T.

Starting Procedures (Injection Systems). The following procedure should be adopted:

(a) The mixture setting is set to IDLE CUT-OFF initially.

(b) After switching on the Fuel Boost Pump the mixture control is set to FULL RICH for a brief period to prime the engine. In some systems the Auxiliary Fuel Pump switch is set to PRIME for a few seconds prior to starting.

After Start and Operating Checks. The following procedure should be adopted:

(a) Check that the oil pressure rises to the normal range within 30 seconds and remains within limits.

(b) Allow the oil temperature to rise to the normal operating range before operating the engine at high power.

(c) Allow the cylinder head temperature to rise to the normal operating range. Mixture control should be RICH during ground operations to prevent overheating.

(d) Check that the manifold pressure is normal for power/rpm settings. Complete a static boost check.

(e) Move IGN switch from BOTH to L and BOTH to R and in each case check for a drop in engine rpm which is within the manufacturer's limits (typically 50-125 rpm).

(f) Check engine response to operation of propeller controls, that is to say MAP against rpm.

Power Adjustments. The following procedure should be adopted:

(a) Throttle movements should be made slowly and smoothly, typically 2 to 3 seconds from full open to closed or vice versa.

(b) After take-off, reduce power setting to CLIMB as soon as practical, to avoid excess CHTs.

(c) In the event of high CHT, reduce power gradually, to avoid cylinder head cracking due to over-rapid cooling.

(d) In constant-speed propeller aircraft always;

(e) increase rpm in advance of throttle (MAP)

(f) reduce MAP in advance of rpm in order to avoid excessive torque between engine and propeller.

The basic procedures are;

(a) to increase power enrich mixture, increase rpm, advance throttle

(b) to decrease power retard throttle, reduce rpm, adjust mixture.

2. During prolonged low power operation, for example during a long descent, periodically advance the throttle for a short period to clear the plugs.

3. Ensure that lean mixtures are only used during the cruise, in accordance with the manufacturer's instructions. At high powers and during descent and landing the mixture control setting should be FULL RICH or RICH. The mixture setting should be decreased at high altitudes on normally aspirated engines.

4. If there is any possibility of carburettor icing, particularly likely during reduced power (descent) operation, set carburettor heat to ON.

Shut Down Procedure. Allow the CHT and oil temperature to fall to normal low power values (less than 400°C CHT) before shutting down an engine. If this procedure is not followed conducted, residual heat may cause damage to bearings and seals. Shut down by setting mixture to IDLE CUT-OFF and switch OFF ignition after the engine has stopped. Leave cowl flaps OPEN until the engine has cooled, check all cockpit switches to OFF. In wet sump lubrication systems, wait an appreciable time before checking engine oil contents, to allow oil to drain back to the sump.

Engine Life

5. The life of a piston engine can be considerably lengthened if care is taken in its operation and handling to ensure that stresses are kept as low as possible. The engine is subject to two types of stress, thermal and mechanical.

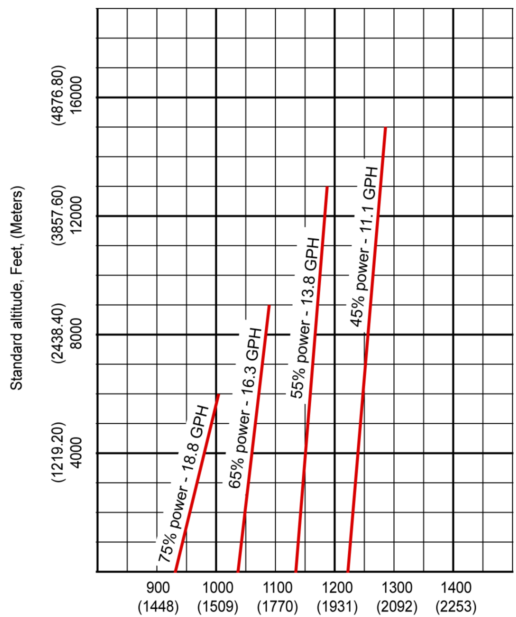
6. Thermal stresses are a function of CHT and oil temperatures. Cylinder head temperatures are kept within limits by correct use of the mixture control and avoiding excessive power (MAP) settings. Oil temperature is controlled by correct use of the oil cooler controls (cowl flap opening). It should be borne in mind that too low an oil temperature can place unacceptable stress on the engine, as well as too high a temperature.

7. Mechanical stresses are a function of rpm and brake mean effective pressure (BMEP). The abrupt reversal of travel of the pistons at the end of each stroke causes large inertia loadings on the pistons, connecting rods and crankshaft. The higher the rpm, the greater these loadings become. The brake mean effective pressure in the cylinders is dependent upon power setting, clearly the higher the power setting the higher the cylinder pressure during the power stroke.

8. Within the limitations of the manufacturer's operating data it is always beneficial to operate at the lowest recommended rpm and the highest recommended MAP for each required operating condition.

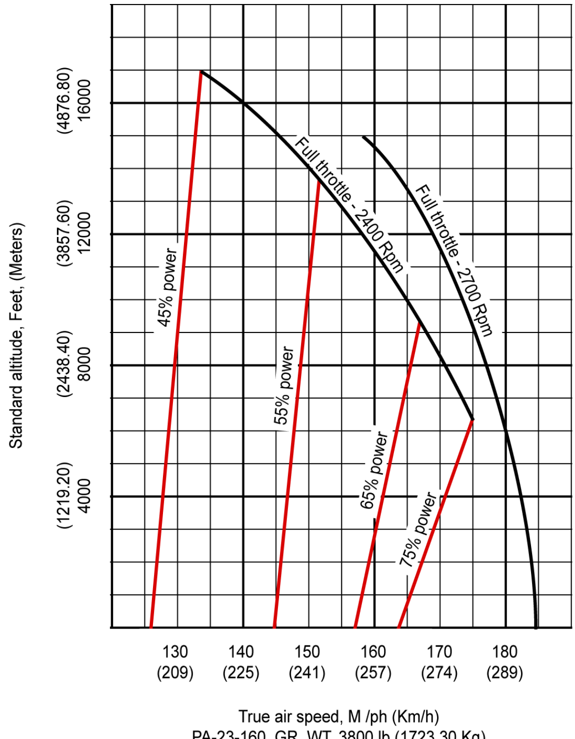
Range and Speed Charts for Power Setting

FIGURE 18-1
Effects of Power Settings on Range (Piston Engine)



Range Miles, (Kilometres)
PA-23-160, GR. WT. 3800 lb (1723.30 Kg)
Mixture leaned, AUX. tanks -108 Gal (408.7 Lt)

FIGURE 18-2
Effects of Power Settings on TAS (Piston Engine)



9. Figure 18-1 and Figure 18-2 show the range and speed charts for a Piper Apache aircraft. Figure 18-1 shows the effects of power settings on range, whereas Figure 18-2 shows the relation between power settings and true airspeed (TAS). By referring to the charts it is possible to determine the correct power setting to select for any flight, within the range of the aircraft, given flight altitude, distance and desired flight time.

10. Suppose a flight of 600 miles is to be made at an altitude of 4000 feet. If it is desired to make the flight in the shortest time then the highest cruise power (75%) would be chosen. From Figure 18-2 it will be seen that, at Full Throttle setting and 2400 rpm, the TAS at this engine power will be approximately 171 mph. Fuel consumption will be 18.8 gallons per hour, so it is a simple matter to calculate the fuel required in still-air conditions.

11. If, alternatively, it is required to make the flight at best economy the minimum cruise power would be chosen (45%). From Figure 18-2 TAS will be approximately 128 mph and fuel consumption, from Figure 18-1, will be 11.1 gallons per hour.

Power Ratings

12. The amount of power that can be generated by an engine is limited by the ability of its components to withstand the resulting temperatures, pressures, accelerations, and velocities. High temperatures in particular tend to cause a break down in the lubrication films between moving parts, resulting in rapid wear. All engines sustain damage continuously when running and the ultimate life of an individual engine depends upon the cumulative effects of this damage. The rate of damage is proportional to the power setting and the cumulative damage is proportional the duration of operation at any given power setting. The useful life of a given engine is therefore inversely proportional to the duration and magnitude of the power settings employed.

13. In order to ensure that engines can be operated safely throughout a specified lifetime, manufacturers authorise a number of different power ratings, some of these may be used continuously whilst others are subject to time limits. An aircraft engine will typically be authorised to operate continuously at a certain "maximum continuous" or "rated" power output and also at some higher power settings for limited periods during manoeuvres such as take-off and climbing. The time limits set for these higher power settings are calculated to ensure that cumulative damage remains within acceptable limits and catastrophic failure is avoided throughout the authorised life of the engine. Excessive use of settings above maximum continuous power are likely to result in material failure within the authorised life of the engine.

Typical power ratings are indicated below:

Power Rating	**Authorised duration**
Maximum Take-Off	Time limited (typically 1 to 5 minutes per flight)
Maximum Cruise	Unlimited throughout cruise flight
Maximum Continuous	Unlimited throughout flight

Self Assessed Exercise No. 4

QUESTIONS

QUESTION 1.

Define the terms "Density Altitude" and "Pressure Altitude" and state a simple equation to determine pressure altitude?

QUESTION 2.

Describe how the performance of normally aspirated piston aero-engines vary with changes in ambient pressure and temperature?

QUESTION 3.

Explain the effects of aircraft altitude on the power output of a normally aspirated piston aero-engine?

QUESTION 4.

Define the term "critical altitude" as applied to piston aero-engines?

QUESTION 5.

Summarise the reasons for fitting turbochargers and superchargers in aircraft engines and state the difference between ground boosting and altitude boosting.

QUESTION 6.

Describe the difference between turbochargers and superchargers?

QUESTION 7.

Describe the principle of operation of the turbocharger?

QUESTION 8.

Explain the function of an intercooler in a turbocharged or supercharged piston engine?

QUESTION 9.

Describe the purpose of the waste gate in an aircraft piston engine, its location, and its operating principle?

QUESTION 10.

List the methods of controlling the waste gate position in an aircraft piston engine?

QUESTION 11.

Describe the positions of the waste gate throughout a normal flight from including, before engine start up, engine idle, take-off power, maximum continuous power at low altitude, climb to high altitude, descent to low altitude and low power setting?

QUESTION 12.

Compare and contrast the curves of maximum power versus altitude of normally aspirated, turbo-charged, and supercharged engines, identifying significant points?

QUESTION 13.

Describe the purpose and operating principles of manifold pressure gauges (MAP and Boost)?

QUESTION 14.

Define the terms "full throttle height" and "rated altitude" as applied to aircraft piston engines?

QUESTION 15.

Define the term "turbo lag" as applied to turbo-charged aircraft piston engines?

QUESTION 16.

Describe how different fuel grades are identified?

QUESTION 17.

Define the term "octane rating" as applied to piston engine fuels?

QUESTION 18.

Define the term "detonation" as applied to piston engines, and describe its causes and effects?

QUESTION 19.

Identify situations and power settings that promote detonation?

QUESTION 20.

Describe the method of checking piston engine fuels for water contamination?

QUESTION 21.

Define the terms "chemically correct ratio", "best power ratio", "lean (weak) mixture", and "rich mixture" as applied to piston engines?

QUESTION 22.

Describe the advantages and disadvantages of weak and rich mixtures in aircraft piston engines?

QUESTION 23.

Describe the relationship between specific fuel consumption and mixture ratio?

QUESTION 24.

Describe the use of exhaust gas temperature as an aid to mixture setting?

QUESTION 25.

Describe the fixed pitch propeller, its operating modes and explain its disadvantage.

QUESTION 26.

Explain why propeller blades are twisted from root to tip?

QUESTION 27.

Describe the variable pitch propeller and explain its advantages by comparing it with a fixed pitch propeller?

QUESTION 28.

Describe the operating principle of the single acting and double acting variable pitch propellers?

QUESTION 29.

Describe the operating principle of a constant speed propeller system for both single and multi engine aircraft?

QUESTION 30.

Describe the operation of the constant speed propeller system during flight, including feathering, unfeathering, and changes in RPM selection and aircraft speed and altitude?

QUESTION 31.

Explain the purpose and basic operating principle of an auto-feather system?

QUESTION 32.

State the purpose and describe the operation of a low pitch stop (centrifugal latch)?

QUESTION 33.

Define the terms "Synchronising" and "Synchrophasing" as applied to aircraft propellers?

QUESTION 34.

Describe the basic operating principles of synchronising and Synchrophasing systems?

QUESTION 35.

Describe how the efficiency of fixed pitch and variable pitch propellers vary with flight speed?

QUESTION 36.

State the purpose of propeller reduction gearing?

QUESTION 37.

State the purpose of a torque-meter and describe its operating principle?

QUESTION 38.

Explain why it is necessary to check the propeller for its physical condition before flight?

QUESTION 39.

Describe the general procedures for setting the engine controls during normal flight from engine start up until shut down?

QUESTION 40.

State the possible use of time limits for take-off and climb power?

QUESTION 41.

Define the term "Rated Power" or "Maximum Continuous power"

ANSWERS:

ANSWER 1.

Density altitude is the altitude in the standard atmosphere at which the prevailing density occurs.

Density Altitude = actual altitude + (120 x temperature deviation from standard atmosphere)

Pressure altitude is the altitude above or below the standard atmosphere datum of 1013.25 mbs.

Pressure Altitude = actual altitude + (4 x temperature deviation from standard atmosphere x altitude/1000 feet)

ANSWER 2.

The power output of normally aspirated piston engines reduces with both decreasing pressure and increasing temperature. The principal effect of increasing temperature is to cause air to expand thereby reducing its density. This reduces the air mass flow through the engine at a given RPM and hence reduces its power output. Although reducing air pressure has the same fundamental effect, the overall consequences are modified somewhat due to the effects of reducing exhaust back pressure which tends to improve the volumetric efficiency of the engine. The overall effect is a gradual reduction in power output for a given RPM

ANSWER 3.

The principal effect of increasing altitude is a reduction in air pressure which tends to reduce air density, mass flow through the engine and hence power output. Below 36000 feet this effect is moderated somewhat by a reduction in temperature which partly offsets the density reduction. The overall effect of these factors is a gradual loss of power up to 36000 feet followed by a greater loss rate above this altitude. The average loss rate being approximately 3.5% per thousand feet overall.

ANSWER 4.

The term "Critical Altitude" applies only to turbo-charged piston engines. As a turbo-charged engine climbs at constant power setting, the waste gate gradually closes in order to increase compressor output to maintain a constant MAP and power output. This process continues until, at a certain altitude depending upon power setting, the waste gate is fully closed and no further increase in compressor output is possible. Any further increase in altitude will result in a decrease in MAP and engine power output. Critical altitude is the altitude at which the waste gate becomes fully open in an engine climbing at rated power. The term is used to distinguish this situation from the "Full Throttle Height" in a supercharged engine, where the throttle becomes fully open in attempting to maintain MAP.

ANSWER 5.

The principal reason for installing turbo-chargers and superchargers in aircraft piston engines is to maintain engine power output to higher altitudes by offsetting the effects of decreasing air pressure. The two basic approaches employed are ground boosting and altitude boosting. In ground boosting MAP is increased above ambient pressure at sea level and is maintained with increasing altitude, the overall effect being an increase in power at all altitudes. In altitude boosting MAP is not increased at sea level, but sea level MAP is maintained up to a higher altitude.

ANSWER 6.

Both superchargers and turbo-chargers are intended to maintain MAP and hence engine power output with increasing altitude. Superchargers are driven directly by a shaft from the engine and hence compressor speed is directly related to engine RPM. The power to drive the compressor is drawn directly from the engine. Because the ratio between engine RPM and compressor RPM is fixed, compressor output pressure is controlled by the throttle butterfly valve which is always located upstream of the compressor. In turbo-charger systems there is no direct link between the engine and the compressor which is driven by engine exhaust gas passing through a turbine. Compressor speed and hence output pressure is controlled by a waste gate which determines the proportion of engine exhaust gas passed through the turbo-charger turbine.

ANSWER 7.

The turbo-charger comprises a centrifugal compressor directly coupled to a centrifugal turbine which is driven by engine exhaust gasses. The speed of the turbo-charger and hence its output flow and pressure are determined by the mass flow of exhaust gas through the turbine. This in turn is controlled by a waste gate located in an exhaust gas passage in parallel with the turbine such that opening the waste gate increases the proportion of exhaust gas by-passing the turbine. In this way turbo-charger output is inversely proportional to the degree of opening of the waste gate. The position of the waste gate is controlled by a servo piston sensing variations in MAP, air temperature, and pressure drop across the throttle butterfly valve.

ANSWER 8.

In order to maintain MAP with increasing altitude the compressor must carry out mechanical work on the air passing through it. Although the principal objective is to increase the static pressure of the air, its temperature is also increased, mainly through the effects of friction. This increased temperature tends to cause a reduction in air density thereby reducing the beneficial effects of constant MAP. More importantly, at high altitudes the resulting cylinder temperatures might be sufficiently high to cause detonation even when using high octane fuels and rich mixtures. In order to overcome this problem some systems employ intercoolers located between the compressor and the engine in order to reduce the temperature of the air to acceptable levels.

ANSWER 9.

The purpose of the waste gate in a turbo-charger system is to provide a means of varying compressor speed and hence to maintain a constant MAP with changing ambient conditions and engine RPM. It achieves this by varying the proportion of engine exhaust gasses passing directly to atmosphere. The waste gate is located in an exhaust gas passageway connected in parallel with the turbocharger turbine. By varying the proportion of engine exhaust gas which by-passes the turbine to go directly to atmosphere, it controls the gas flow through and hence RPM of the turbine. Because of this relationship between waste gate angle and exhaust flow to atmosphere, turbo-charger RPM and hence compressor output are inversely proportional to the degree of opening of the waste gate.

ANSWER 10.

The position of the waste gate in an aircraft piston engine turbo-charger is controlled by a spring-biased servo piston. With the engine shut down there is no servo pressure and so the spring holds the waste gate in the open position. Servo pressure is controlled by varying servo spill off valves using the following methods:

(a) In the most simple systems a MAP controller employs an aneroid capsule sensing MAP to control a servo spill valve.

(b) In more advanced systems a density controller employing a nitrogen filled capsule sensing air temperature and a differential pressure controller employing a diaphragm sensing the pressure drop across the throttle, each control servo spill valves.

(c) Still more advanced systems comprise an absolute pressure controller employing an aneroid capsule sensing compressor discharge pressure, a rate controller employing a diaphragm sensing rate of change of compressor output pressure, and a pressure ratio controller employing an aneroid capsule sensing ambient pressure, each control servo spill valves.

ANSWER 11.

Prior to start up servo pressure is zero and the waste gate is held fully open by the servo piston spring. As the engine accelerates from start up to idle speed, rising servo oil pressure drives the waste gate to the closed position directing all exhaust gas to the turbine causing it to accelerate to the speed necessary to establish the required MAP. As the engine accelerates to take of power the increasing exhaust gas output causes the turbine to accelerate tending to increase MAP. The waste gate then moves to the almost fully open position to maintain constant MAP, the overall effect being a slight rise in turbo-charger RPM. As the aircraft climbs at rated power the waste gate gradually closes increasing turbo-charger speed to maintain constant MAP until, at critical altitude the wastegate is fully open. Above critical altitude the waste gate remains fully closed as MAP and power output fall with increasing altitude. As the aircraft descends the waste gate remains closed until critical altitude, below which it gradually opens to maintain MAP against increasing ambient pressure. As power is reduced during and after landing the waste gate continues to open, becoming fully open after engine shut down.

ANSWER 12.

In normally aspirated engines MAP decreases with increasing altitude. The power output of such engines therefore reduces with increasing altitude at a rate of approximately 3.5% per 1000 feet.

In supercharged piston engines MAP is maintained at a constant level between sea level and full throttle height, above which it reduces in the same way as that for normally aspirated engines. Also as ambient pressure reduces with increasing altitude, this results in a gradual reduction in exhaust back pressure. This has the effect of improving the volumetric efficiency and hence power output of the engine. Finally the power extracted to drive the supercharger results in a reduction in power output.

The overall result of these factors is a reduction in power at ground level, followed by a gradual increase in power output up to full throttle height, above which power reduces at a rate of approximately 3.5% per 1000 feet.

In a turbo-charged engine MAP remains constant up to critical altitude, above which it reduces in the same way as in a normally aspirated engine. The power necessary to drive the turbocharger is not drawn directly from the engine so no significant loss of power occurs at ground level. However, the reduction in exhaust back pressure and improved volumetric efficiency with altitude are largely negated by the restriction caused by the turbine. The overall effect is a gradual reduction in power from sea level up to critical altitude above which power reduces at approximately 3.5% per 1000 feet.

ANSWER 13.

When a supercharged engine climbs at constant power setting the automatic boost controller will gradually open the throttle to maintain constant MAP. At some altitude the throttle will be fully open and any further increase in altitude will result in a reduction in MAP and power similar as if the engine were normally aspirated. The altitude at which the throttle becomes fully open is termed the "full throttle height". There will be a different full throttle height for every possible power setting and for a climb conducted at rated power the full throttle height is termed the "rated altitude".

In the case of a turbocharged engine a similar process occurs, but in this case it is the waste gate that becomes fully closed rather than the throttle becoming fully open. The altitude at which this occurs in a rated power climb is termed the "critical altitude".

ANSWER 14.

The term "turbo-lag" refers to the phenomenon whereby the turbo-charger systems requires a finite time to respond to changes in throttle position selected by the pilot. Each time the throttle position is changed the various sensors and controllers within the automatic MAP control system must detect and react to the changing conditions before reaching a new equilibrium condition. This can lead to an initial reduction in MAP immediately after rapid throttle opening, followed by an overshoot in MAP as the system reacts. In extreme cases several cycles may occur before equilibrium is established, or else the pilot might be required to make repeated minor adjustments, fine tuning the system to the required condition.

ANSWER 15.

Different fuel grades are identified by a combination of title, number, and in some cases letters indicating additives. For example AVGAS 100LL indicates that the fuel is aviation gasoline (AVGAS), of 100 octane rating (100), with a low lead content (LL). Similarly MOGAS 85 indicates motor gasoline (MOGAS), of 85 octane rating (85). AVTUR (FSII) indicates aviation turbine fuel (AVTUR) with a fuel system icing inhibitor additive (FSII). In addition to the above markings on fuel tankers and containers coloured dyes are sometimes added as an aid to identification. AVGAS 100LL for example is dyed light blue.

ANSWER 16.

The term octane rating refers to the process whereby the detonation resistance of fuels are compared with mixtures of two chemicals of vastly different detonation characteristics. The first of these is iso-octane which is extremely resistant to detonation, whilst the second, heptane detonates easily. By testing the fuel under standard conditions its detonation performance is compared with those of varying iso-octane :heptane mixtures. The octane rating number indicates the mixture with the same characteristics as the fuel. For example an 85 octane gasoline has the same characteristics as a mixture of 85% iso-octane and 15% heptane. A 120 octane gasoline has characteristics 20% better than those of pure iso-octane.

ANSWER 17.

The term detonation describes the phenomenon where the fuel air mixture explodes spontaneously rather than igniting and burning. In normal burning the flame front progresses through the mixture at 60 to 80 feet/second whereas in detonation it progresses at up to 1000 feet/second. The resulting rapid heating and expansion of the mixture produces very high pressures within the cylinders. Because the pistons cannot react quickly enough to absorb the energy of this reaction it is wasted in the form of heat causing high temperatures and localised hot spots. In extreme or prolonged cases the engine is likely to suffer structural damage.

Detonation may be caused by any or all of the following:

Excessive induction air temperatures due to inappropriate use of carburettor heat, alternate air, or excessive supercharging.

Weak mixtures or fuel of too low an octane rating.

Ignition too far advanced.

Excessive torque/power selected at too low an engine RPM in a constant speed propeller system.

ANSWER 18.

The following situations and power settings are likely to result in detonation:

Too weak a fuel:air mixture.

Ignition too far advanced.

Selected torque too high for RPM.

High charge temperatures due to inappropriate use of carburettor heat alternate hot air in fuel injection systems, or excessive supercharging.

Selection of high power at low RPM with a constant speed propeller.

Too low a fuel octane rating.

ANSWER 19.

The following methods are employed in testing piston engine fuels for water contamination:

Visual inspection. A small quantity of fuel is drained from the lowest point in the tanks or from drain points and inspected visually for signs of water which collects at the bottom of the specimen.

Water finding paste. A small quantity of water finding paste is smeared onto the end of a dipstick which is then dipped to the bottom of the fuel tank. The presence of water is indicated by a change in the colour of the paste.

Water detecting capsules. A capsule incorporating a paper element impregnated with water detecting chemicals is placed onto the end of a syringe. A specified quantity of fuel is then drawn into the syringe causing it to pass through the capsule. The presence of water is indicated by a change in the colour of the paper element in the capsule.

ANSWER 20.

The chemically correct air fuel ratio is that at which complete combustion occurs causing all of the carbon and hydrogen in the fuel to combine with all of the oxygen in the air. This occurs at a ratio of approximately 15:1.

The best power air fuel ratio is that at which the engine produces the most power for a given RPM. A lean (weak) mixture is one in which the air fuel ratio is greater than 15:1.

A rich mixture is one in which the air fuel ratio is less than 15:1.

ANSWER 21.

The principal advantage of employing weak mixtures is that they tend to reduce fuel consumption rate and hence increase flight endurance for a given fuel load. The main disadvantage of weak mixtures is the increased probability of detonation. Rich mixtures aid engine starting and provided additional cooling when operating at high power settings. In this case the excess fuel is not burned but by absorbing latent heat in evaporating it cools the engine. Because the excess fuel is not burned fuel consumption is increased. Rich mixtures can also result in fouling of spark plugs.

ANSWER 22.

The specific fuel consumption of an engine is the quantity of fuel required to produce a unit of power for a unit of time. It is measured in lbf/HP hour. Specific fuel consumption varies with mixture in a complex way. At high power settings a rich mixture is required in order to provide excess fuel to cool the engine. This excess fuel contributes nothing to power output and hence specific fuel consumption is high under such conditions.

For economical cruising flight at lower power settings, a weaker mixture is employed with all of the carbon and hydrogen in the fuel being combined with oxygen to contribute to power output. The lowest specific fuel consumption is achieved under these conditions.

At still leaner mixtures the engine runs at lower temperatures because the flame rate is lower in the weak mixtures. This lower flame rate also reduces cylinder pressures and hence power output. The overall effect of this is an increase in specific fuel consumption.

Because of the above factors SFC is high at low fuel:air ratios, gradually reducing to a minimum value at a fuel:air ratio of about 0.066, before increasing to very high values as mixture strength increases.

ANSWER 23.

Because cylinder temperature varies with fuel:air ratio, selection of the best economy mixture for cruise flight can be carried out using EGT as a guide. As mixture ratio is reduced from full rich EGT will gradually increase attaining a maximum at best economy mixture. Further leaning beyond this point will result in a reduction in EGT. Best economy mixture is selected by carefully adjusting the mixture to locate the maximum EGT point.

ANSWER 24.

The fixed pitch propeller comprises of two or more aerofoil section blades mounted upon a central hub. In modern light aircraft it is usually attached directly to the engine crankshaft. The blades are twisted from root to tip to provide a uniform angle of attack along their length. Due to its fixed pitch this type of propeller has only two modes of operation, normal in which it is driven by the engine, and windmilling in which the airflow causes it to drive the engine after engine failure or in flight shut down. The principal disadvantage of the fixed pitch propeller is that it can be designed to operate at one on RPM/airspeed combination and is hence inefficient throughout most of the operating range.

ANSWER 25.

Although the inflow velocity due to forward speed is more-or-less constant across the entire propeller disc, the rotational velocity increases as a direct function of radius from root to tip. This means that the angle of attack of an untwisted blade would be much higher at the tip than at the root, with the effect that only one part would be at its most efficient angle, whilst all others would vary from very inefficient due to low angle of attack, to approaching the stall. Propeller blades are twisted from root to tip to provide a uniform angle of attack along their entire length.

ANSWER 26.

The principal disadvantage of the fixed pitch propeller is that it can operate efficiently at only one RPM/airspeed combination. Under all other conditions it operates inefficiently due to either too low an angle of attack at high airspeeds, or too high an angle of attack at low airspeeds. A further disadvantage is that it generates considerable drag when windmilling following in-flight engine failure.

In variable pitch propellers the blade angle can be varied automatically in flight to achieve optimum efficiency at all power settings and airspeeds. In the case of in-flight engine failure it can be feathered in order to prevent windmilling and minimise drag.

ANSWER 27.

In variable pitch propeller systems blade angle is altered by means of a piston, usually located in the spinner. In the case of the double acting system, blade angle is controlled by varying the ratio of oil pressures acting on both sides of the piston. In the single acting system oil pressure acts on only one side, with the other being subjected to the effects of a powerful spring. If the propeller is part of a constant speed system the magnitude of the oil pressure acting on the piston is determined by a centrifugal governor sensing propeller RPM. Decreases in RPM below the set value cause blade pitch to increase and increases above selected RPM cause it to decrease. In this way the drag on the propeller is varied to maintain a constant RPM. Also as airspeed increases reducing angle of attack reduces drag on the propeller blades tend to make them accelerate. This is sensed by the centrifugal governor which increases the blade angle to restore angle of attack, RPM, and propeller efficiency. The reverse process occurs in the case of a reduction in airspeed.

ANSWER 28.

Propeller angle is altered by means of a piston, usually located in the spinner. The flow of servo oil pressure to and from the piston is determined by a centrifugal governor sensing propeller RPM. Variations in RPM caused by changes in altitude or power setting cause blade pitch to increase or decrease varying propeller drag forces to maintain a constant RPM. Also as airspeed increases, reducing propeller blade angle of attack reduces drag on the propeller causing it to accelerate. This is sensed by the centrifugal governor which increases the blade angle to restore angle of attack, RPM, and propeller efficiency. The reverse process occurs in the case of a reduction in airspeed. Changes in RPM selection are achieved by adjusting the spring loading on the centrifugal governor in order to establish the required datum RPM.

In the case of multi-engined aircraft this system is sometimes refined further to include a facility for propeller RPM synchronisation. In this case one propeller is selected as master and the speed of the others is matched to it by varying spring loading in the centrifugal governors.

ANSWER 29.

Propeller angle is altered by means of a piston, usually located in the spinner. The flow of servo oil pressure to and from the piston is determined by a centrifugal governor sensing propeller RPM. Variations in RPM caused by changes in altitude or power setting cause blade pitch to increase or decrease varying propeller drag forces to maintain a constant RPM. Also as airspeed increases, reducing propeller blade angle of attack reduces drag on the propeller causing it to accelerate. This is sensed by the centrifugal governor which increases the blade angle to restore angle of attack, RPM, and propeller efficiency. The reverse process occurs in the case of a reduction in airspeed. Changes in RPM selection are achieved by adjusting the spring loading on the centrifugal governor in order to establish the required datum RPM. In the event of engine failure, selection of feather position overrides the centrifugal governor causing the piston to be driven through coarse pitch to the feather position. Alternatively in autofeather systems this is initiated by a torque switch sensing loss of engine torque.

ANSWER 30.

The purpose of the auto-feather system is to automatically feather the propeller in the event of in-flight engine failure or shut down. The autofeather sequence is initiated by a torque switch sensing the loss of engine torque. The system then operates a solenoid directing oil pressure from an auto-feather pump to the constant speed unit. This lifts the selector valve directing oil pressure to the coarse pitch side of the piston, thereby driving the blades to the feather position.

ANSWER 31.

The purpose of the low pitch stop (centrifugal latch) is to prevent single acting variable pitch propellers from moving to the feather position as servo oil pressure reduces to zero after engine shut down. If the propeller is permitted to move to the feather position after shut down this is likely to impose unacceptable torque loading due to drag forces during subsequent attempts to start the engine. The mechanism comprises a centrifugal latch which, under the action of springs locks the blade angle actuating piston at low RPM. As RPM increases after start up centrifugal force overcomes spring force and disengages the latch.

ANSWER 32.

The term "Synchronising" describes a system in which the propellers of a multi-engined aircraft are all made to run at the same speed in order to minimise harmonic vibrations. The term "Synchrophasing" describes a system in which this process is taken a step further whereby in addition to RPM matching, the angular position of all the propellers are matched.

ANSWER 33.

In the synchronising system the RPM of each propeller is sensed using some form of tachometer system. One propeller is selected as the master and the speed of the other propellers is altered until all the tachometer system speed signals are the same. In the case of synchrophasing, one propeller blade is set as the master and an individual blade in each propeller is set as the master for that propeller. The RPM of each propeller and the angular position of each master blade are sensed. The system then alters the speed of the non-master or slave propellers until both RPM and master blade positions for all propellers are the equal.

ANSWER 34.

Because its blade angle is fixed, the angle of attack of the blades of a fixed pitch propeller vary with RPM and aircraft forward speed. At a given RPM the efficiency of such propellers increases from zero when the aircraft is stationary, to some maximum value, before decreasing rapidly with further increases of forward speed. The speed band of maximum efficiency is very narrow and the efficiency lapse rate is very steep above this band. By varying blade pitch, variable pitch propellers maintain optimum angle of attack over a much wider speed range. This has the effect of achieving maximum efficiency at lower airspeeds, maintaining it over a wide speed band, and reducing the efficiency lapse rate above this band.

ANSWER 35.

The efficiency of the propeller and engine each vary with torque and RPM. The most efficient combination for propellers is relatively low RPM coupled with high torque. Conversely piston engines are most efficient at high RPM and relatively low torque. The purpose of the reduction gearing is to enable both the engine and propeller to operate close to their most efficient torque/RPM ranges.

ANSWER 36.

The torque meter in a propeller system is fitted to provide the pilot with an indication of the amount of torque being applied by the engine. This is used in setting the RPM and throttle selections for a given power output or flight condition. The system works by varying the oil bleed from a number of cylinders such that the resulting pressure is directly proportional to engine torque.

ANSWER 37.

Because of its high RPM and exposed position at the front of the aircraft then propeller is liable to be damaged by impacts with stones and similar debris. Such damage can affect the propeller in a number of ways. Changes in blade cross-section will reduce the efficiency of the propeller. Changes in balance will cause vibration, particularly at high RPM. Finally damage in the load bearing areas of the blades is likely to result in fatigue damage and ultimately catastrophic failure. To prevent these problems it is essential that propellers be thoroughly examined for any signs of such damage before engine start up and after shut down.

ANSWER 38.

Engine start-up. Set master switch and booster pump to on, throttle to 1/2 open, mixture to full rich, propeller RPM to full increase, and ignition switch to start. After start-up release the ignition switch to the both position and check normal oil pressures are attained within 30 seconds.

Power adjustments. When making power adjustments in a constant speed system there is a danger of applying excessive torque to the propeller shaft and overheating or stalling the engine unless the correct sequences are employed. The sequence for increasing power is to enrich the mixture, increase RPM setting, increase throttle setting. The sequence for reducing power is to retard the throttle, reduce RPM, then adjust the mixture.

Shut-down. Before shut-down the engine must be run at idle for a short period to allow temperatures to stabilise below 400 degrees Celsius. The engine is shut down by setting the fuel mixture to idle cut-off and allowing the engine to stop before switching off the ignition systems. Cowl flaps must be left open until the engine has cooled.

ANSWER 39.

The amount of power that can be generated by an engine is limited by the ability of its components to withstand the resulting temperatures, pressures, accelerations, and velocities. All engines sustain damage continuously when running and the ultimate life of an individual engine depends upon the cumulative effects of this damage. The rate of damage is proportional to the power setting and the cumulative damage is proportional to the duration of operation at any given power setting. The useful life of a given engine is therefore inversely proportional to the duration and magnitude of the power settings employed.

In order to ensure that engines can be operated safely throughout a specified lifetime, manufacturers authorise a number of different power settings, some of these may be used continuously whilst others are subject to time limits. An aircraft engine will typically be authorised to operate continuously at a certain "maximum continuous" or "rated" power output, and at some higher power settings for limited periods during manoeuvres such as take-off and climbing. The time limits set for these higher power settings are calculated to ensure that cumulative damage remains within acceptable limits and catastrophic failure is avoided throughout the authorised life of the engine.

ANSWER 40.

In order to ensure that engines remain serviceable for a specified life, manufacturers authorise a number of different power settings, some of which are subject to time limits. The time limits set for these higher power settings are calculated to ensure that engines will reach their authorised service life without failure. Rated, or continuous power is the highest power setting at which an engine is authorised to operate without time limits throughout its life.

Gas Turbine Principles of Operation

Gas Turbine Principles of Operation

Introduction

1. The gas turbine engine is a relatively new form of aircraft propulsion unit, having first flown successfully as an experimental power plant during the Second World War and only truly beginning to replace the piston engine in commercial aircraft in the early 1950's.

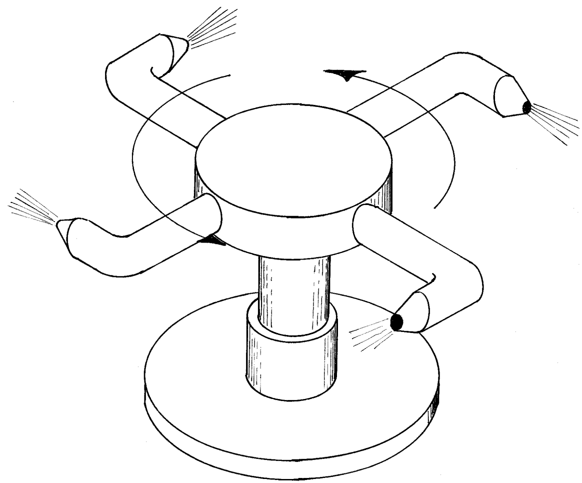
2. The gas turbine may be used to accelerate a mass of gas through the engine and a propelling nozzle, in which case it is known as a turbo-jet, to drive a propeller (turbo-prop), a large fan (turbo-fan), or to drive the rotor of a helicopter (turbo-shaft). Early examples of aircraft using the first two propulsion methods were the De Havilland Comet and the Vickers Viscount. Most present day large transport aircraft use turbo-fan engines.

3. All aircraft propulsion systems employ Newton's third law of motion which states that for every force (or action) there is an equal and opposite reaction. The amount of forward thrust created is proportional to the product of the mass of air affected and the rearward acceleration imparted to it. In the case of propeller driven aircraft and turbo-fans a large mass of air is given a relatively small rearward acceleration. In turbo-jets a much smaller mass of air is subject to a much greater acceleration.

Jet Propulsion

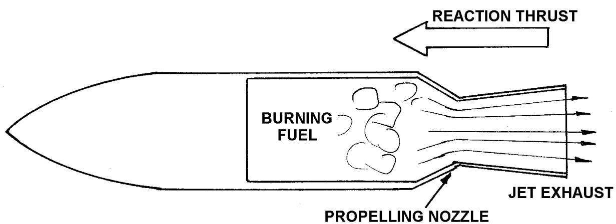
4. One of the best known examples of jet reaction thrust producing movement is the rotary garden sprinkler. The acceleration of a mass of water from the jets produces a force (F = ma), or action. It is the reaction to this force that causes the arm, upon which the jets are mounted, to rotate. This is illustrated at Figure 19-1.

FIGURE 19-1
Reaction Thrust
(Garden Sprinkler)



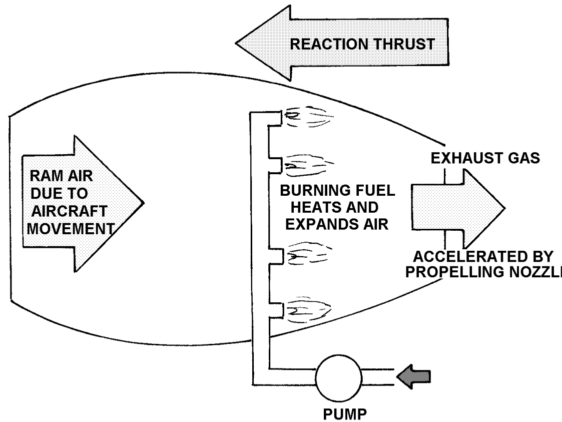
5. A rocket is another example of jet propulsion. The propulsive thrust to drive the rocket is obtained by accelerating a mass of gas rearwards. The reaction to the force produced propels the rocket forwards. This principle is illustrated at Figure 19-2.

FIGURE 19-2
Reaction Thrust (Rocket)



6. Another type of jet engine that has aeronautical uses (such as missile propulsion) is the ram jet, illustrated at Figure 19-3. Provided the engine is attached to a moving vehicle, air is forced into the air intake at the front of the engine. Burning fuel then increases the total energy of the air and the expanding heated gas is accelerated rearwards through a propelling nozzle. The force resulting from the acceleration of a mass of gas rearwards produces reaction thrust forwards.

FIGURE 19-3
Ramjet



7. Any mass of gas possesses energy in a number of forms including thermal energy in the form of its temperature, static pressure, and kinetic energy which is a function of velocity. The kinetic energy of a gas is evident in the form of dynamic pressure acting in the direction of gas flow. If no energy is added to or removed from the gas then the sum of these energies remains constant. The distribution of energy between the three forms can however be changed with, for example dynamic pressure being exchanged for static pressure and temperature when the velocity of the gas is reduced.

8. When such a gas passes through a divergent duct (Figure 19-4) its velocity decreases to maintain constant mass flow through the increasing cross-section of the duct. This causes the dynamic pressure of the gas to be exchanged for higher levels of static pressure and temperature with the overall energy content remaining constant. When passing through a convergent duct (Figure 19-5), this process is reversed, with increasing velocity to maintain mass flow causing an increase in dynamic pressure coupled with compensating reductions in static pressure and temperature.

9. The operation jet engines is based upon such exchanges of energy, together with those of combustion and mechanical compression using moving ducts, both of which involve the introduction of additional energy to the gas. In the ram jet engine, air entering the intake passes through a divergent duct. The resulting velocity reduction converts some of its dynamic pressure into static pressure and temperature. Thermal energy is then added by the combustion of fuel thereby increasing the temperature and total energy or the gas. The high energy gas is then ejected through a convergent duct where the increase in velocity converts some of the static pressure and temperature into dynamic pressure. The thrust of the engine being the reaction to the acceleration of the gas through the convergent duct. In order to initiate the ram jet working cycle the engine must first be accelerated to a high velocity to provide the required inlet air flow. Such engines therefore have no practical use in current commercial aircraft.

FIGURE 19-4
Flow through
Divergent Duct

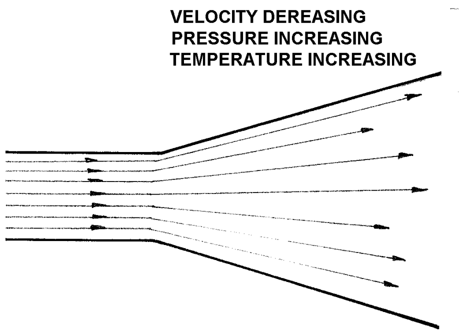
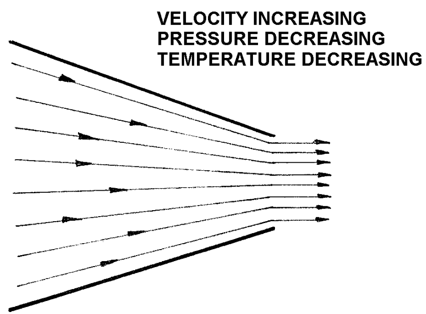


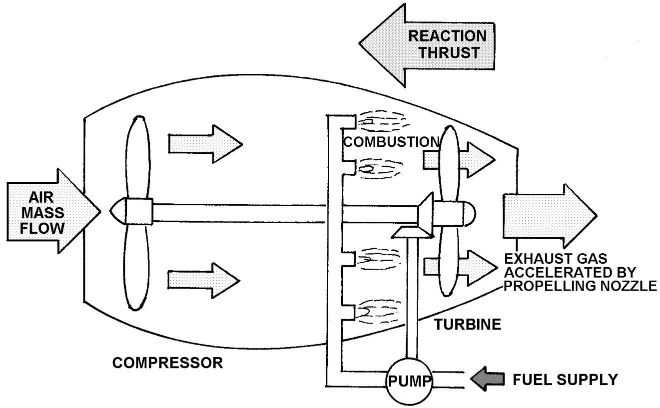
FIGURE 19-5
Flow through
Convergent Duct



10. In the ram jet engine, air entering the intake passes through a divergent duct and part of it is converted to static pressure energy. Adding heat energy from combustion increases the total energy. The expanding hot gas passes through a convergent duct (the propelling nozzle) where some of its heat and static pressure energy are converted to dynamic (velocity increase or acceleration). The ram jet has little practical application as an aircraft power plant, since it needs initial high velocity before it can develop thrust.

11. A practical aircraft engine must be capable of producing thrust when the aircraft is stationary and at low forward speeds. To achieve this it must be capable of inducing an air mass flow into the intake, even when the aircraft is stationary, in order to complete the energy conversion/transfer processes described previously. The inflow of air is induced by means of an engine driven fan, or compressor. The air is the heated by the combustion of fuel and the expanding hot gas is accelerated out of the rear of the engine by a propelling nozzle. Turbines powered by the expanding hot gases produced by combustion are currently employed to power the compressors in the engine. This is illustrated at Figure 19-6.

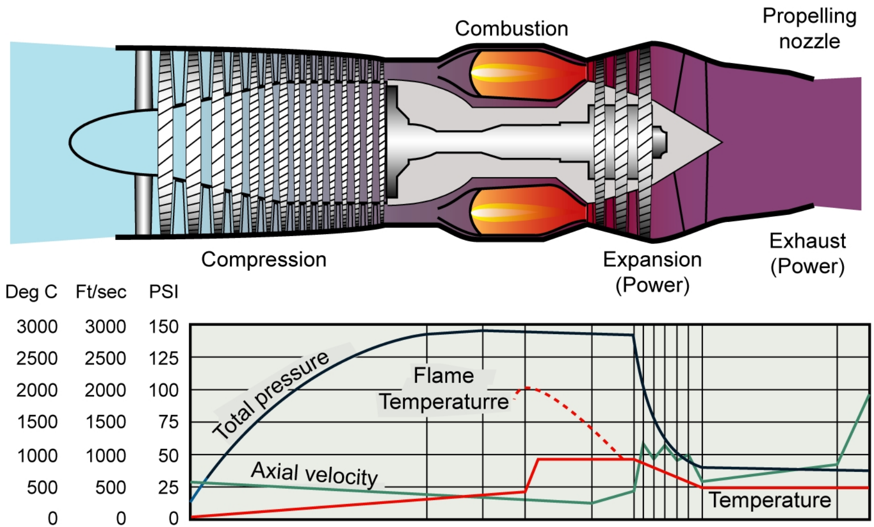
FIGURE 19-6
Basic Gas Turbine



Gas Turbine Working Cycle

12. The working cycle of a gas turbine is basically the same as that of a piston engine - induction, compression, combustion power and exhaust, as shown at Figure 19-7.

FIGURE 19-7
Gas Turbine Working Cycle



13. The major differences are that in jet engines combustion takes place at constant pressure, whereas in the piston engine it takes place at constant volume, and the processes of the cycle are continuous in the gas turbine. Often in jet engines the power extraction process links places both in the turbine and the propelling nozzle. In the piston engine the processes are sequential and power is only produced during the power stoke. In the gas turbine the processes are continuous and so greater power can be produced for a given size of engine.

14. It will be seen from Figure 19-7 that the combustion chamber is not an enclosed space and, consequently, there is no pressure increase due to heating of the air during combustion. Instead, the hot air expands through the turbine, causing the turbine and its attached compressor to rotate. The rotating compressor draws air in through the air intake to replace that expanding through the combustion chamber.

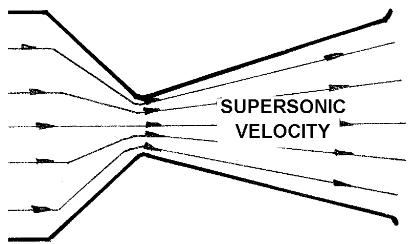
15. As indicated in Figure 19-7, the highest gas temperatures occur in the combustion and turbine sections of the engine. In the case of the turbines, these high temperatures coupled with high centrifugal loading mean that the turbine discs and blades operate very close to the limit of their physical capabilities. It is therefore these materials that ultimately limit the maximum operating temperature and power output of the engine.

16. Within the temperature limits of the turbine and combustion chamber materials, the greater the heat, the greater the expansion, the faster the speed of rotation, the greater the air mass flow. The air mass flow through the engine is given a large acceleration by the propelling nozzle, and it is this acceleration that produces reaction thrust. The greater the air mass flow, and the more it is accelerated, the greater the reaction thrust.

17. The changes in pressure, velocity and temperature that take place through the gas turbine engine are shown graphically at Figure 19-7. During compression, pressure energy is added to the intake air. Total energy is further increased during combustion, with the addition of heat energy. Through the turbine expansion results in conversion of some of the total energy of the gas into useful work, the remainder of the energy conversion, into kinetic energy for propulsive thrust, takes place in the propelling nozzle.

18. The function of the propelling nozzle is to convert the energy of the turbine exhaust gas from pressure and heat energy into velocity (kinetic) energy. For this a convergent duct, or nozzle is used, so long as the gas velocity does not exceed sonic speed (the local speed of sound). Where supersonic gas velocities are encountered, as is the case in a rocket or ram jet engine, a convergent/divergent nozzle, or venturi, is necessary in order to obtain maximum conversion of energy. Such a nozzle is illustrated at Figure 19-8

FIGURE 19-8
Convergent / Divergent Propelling Nozzle



Engine Efficiencies

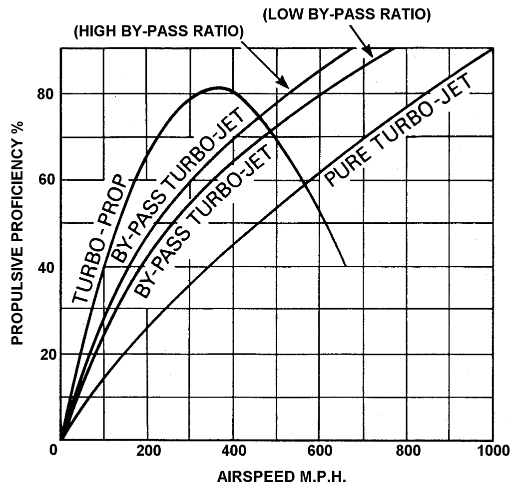
19. The efficiency of any engine is the ratio of output to input. One of the most important measures of the efficiency of a jet engine is the amount of thrust produced, divided by the fuel consumption. This is known as the thrust specific fuel consumption. The major factors that affect specific fuel consumption are propulsive efficiency and overall efficiency.

20. Propulsive efficiency is the ratio of the amount of thrust developed by the propelling nozzle to the energy supplied to the nozzle in a usable form. Another way of expressing this is:

$$\text{Propulsive Efficiency} = \frac{\text{Propulsive Work Done on Aircraft}}{\text{Propulsive Work Done in Aircraft} + \text{Work Wasted in the Exhaust Gas}}$$

21. A comparison of the propulsive efficiencies of various types of gas turbine engine is shown at Figure 19-9.

FIGURE 19-9
Propulsive Efficiencies



22. Overall efficiency is the ratio of the amount of energy produced by the engine in usable form to the total amount of energy available in the fuel. It is a combination of combustion efficiency, thermal efficiency, mechanical efficiency, compressor efficiency and so on. It is dependent upon the efficiency of each of the working cycle processes, whose function is to convert the energy in the fuel into a form that the propelling nozzle can turn into thrust (kinetic energy).

23. Thermal efficiency is defined as the ratio of the mechanical energy output of the engine to the heat energy available in the fuel consumed. It increases as the turbine inlet temperature increases. The thermal efficiency of a jet engine also increases with increased airspeed, due to ram effect at the compressor inlet. Under static sea level conditions the thermal efficiency of a jet engine is 20% - 25%, compared to 25% - 30% for a piston engine. However, the thermal efficiency of the piston engine decreases with increasing airspeed, becoming significantly lower than that of the jet engine at higher airspeeds.

$$\text{Overall Efficiency} = \text{Thermal Efficiency} \times \text{Propulsive Efficiency}$$

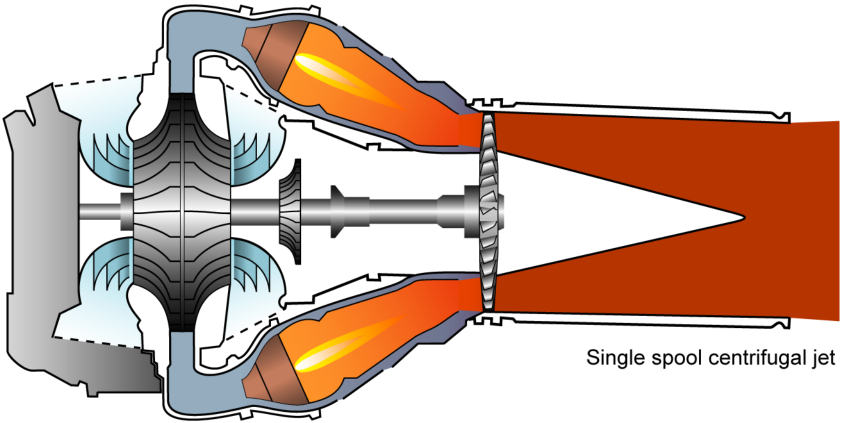
Engine Developments

24. As previously mentioned, jet engines may be arranged mechanically in a number of forms - turbojet, turbo-prop, turbo-shaft, turbo-fan. They may also be known as single-spool, twin-spool, triple-spool and high or low by-pass ratio engines. These various mechanical arrangements are explained further in the following text. In addition, the turbine-driven compressor may be of the centrifugal type (already encountered in piston engine superchargers) or of the axial flow type.

Single-Spool Engines

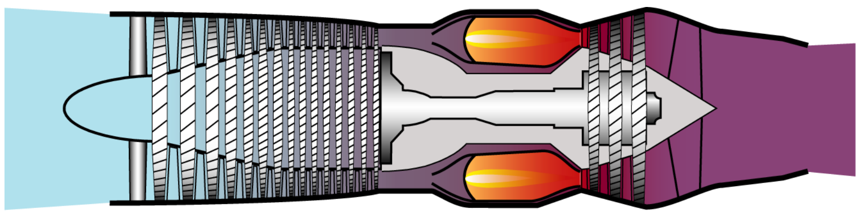
25. Figure 19-10 shows an early turbo-jet engine of the Rolls Royce Nene type. Air mass flow is provided by a double only centrifugal compressor. Hot, expanding gas from the combustion chambers powers the single stage turbine, which drives the compressor. The turbine exhaust is accelerated through the propelling nozzle to produce reaction thrust. Air mass flow in such an engine is limited by the low compression ratio inherent in centrifugal compressor, limiting the propulsive thrust available.

FIGURE 19-10
Early Jet Engine



26. Axial flow compressors, so called because the airflow through them is parallel to the axis of rotation of the compressor, are capable of a much greater mass flow. Hence much greater propulsive thrust is possible from the engine. Figure 19-11 shows a single spool turbo-jet engine with an axial flow compressor. The Rolls Royce Viper is a typical example of such an engine

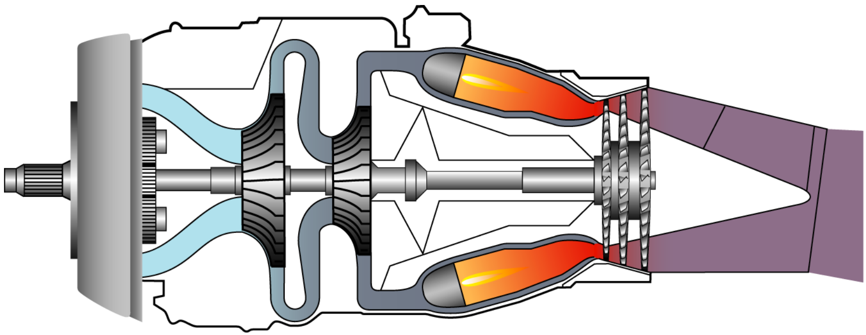
FIGURE 19-11
Single-Spool Axial Turbo Jet



Single spool axial flow turbo-jet

27. Reference to Figure 19-9 will show that, at airspeeds below about 450 mph, the propeller/engine power plant arrangement has better propulsive efficiency than the turbo-jet engine. Consequently, for aircraft designed to operate at speeds up to about 350 mph a marriage of the propeller to the gas turbine, with its excellent power-to-weight ratio, is attractive. Hence the development of the turbo-prop engine from the earliest days of the jet engine. Figure 19-12 shows a turbo-propeller engine of the type used on many transport aircraft of the 1950's and '60's. In turbo-propeller engines the majority of the energy in the hot gas is extracted by a turbine and transmitted through a drive shaft and reduction gearbox to the propeller. This process leaves little or no useful energy in the exhaust gas to provide reaction thrust. In the case of early turboprops a single turbine drove both the compressor and the propeller whereas in more modern engines a free power turbine, totally independent of the compressor is used to drive the propeller.

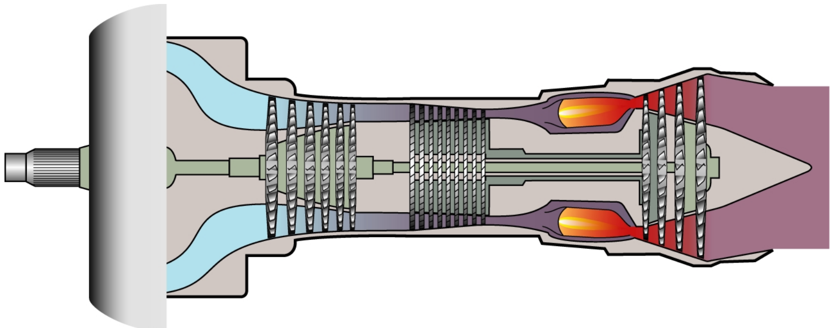
FIGURE 19-12
Early Turbo Prop Engine



Twin-Spool Engine

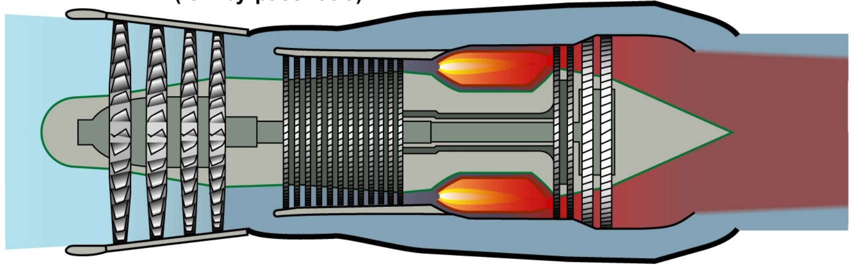
28. In order to increase the thrust of a turbo-jet or the shaft horsepower of a turbo-prop engine it is necessary to increase the air mass flow through the engine. This, in turn, depends upon the speed of rotation of the compressor and the compression ratio (pressure rise achieved across the compressor stages). Increasing the number of compression stages increases the compression ratio. For reasons to be discussed later, this is usually best achieved by having two or more axial flow compressors arranged in series (one after the other), each driven by its own turbine. This arrangement is shown in the turbo-propeller engine illustrated at Figure 19-13.

FIGURE 19-13
Twin-Spool Axial Flow Turbo Prop Engine



29. Hot expanding gas from the combustion chambers powers a high-pressure turbine, which is mounted on the same shaft as the high-pressure compressor, forming the HP spool. Exhaust gas from the HP turbine powers a second turbine, on the same shaft as the entry, or low-pressure compressor, forming the LP spool. This rotating assembly also drives the propeller through reduction gearing. It will be noted that the HP spool is hollow and concentric with the LP spool. The two spools are completely mechanically independent. The HP spool rotates much faster than the LP, because of the greater energy in the gas powering the HP turbine. This arrangement is known as a twin spool engine.

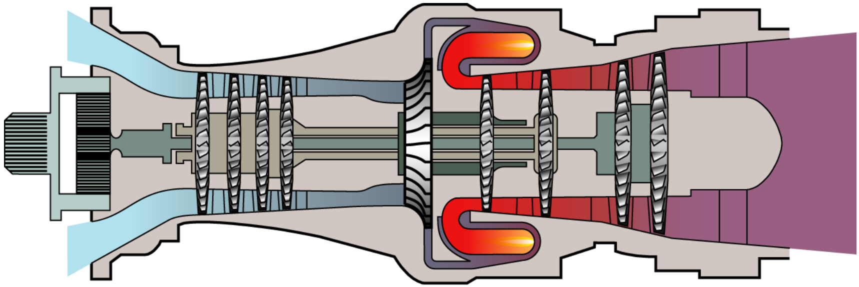
FIGURE 19-14
Twin-Spool Turbo Jet Engine



30. Figure 19-14 shows a twin-spool turbo-jet engine. The mechanical arrangement is very similar to that in Figure 19-13, except there is no propeller driven by the LP spool. Instead, the increased air mass flow from the duplicated compressors, accelerated by the propelling nozzle, gives increased thrust. Because the air mass flow is well in excess of that required for combustion, some of the output of the LP compressor by-passes the HP compressor, the combustion chambers, and the turbines to mix with the hot gas stream in the exhaust, just upstream of the propelling nozzle.

31. In such an engine the lower gas velocity (because some of the air is not heated and expanded) is offset by the much greater mass flow, to create significantly more thrust than the single-spool pure jet engine. Furthermore, the lower exit velocity of the gas means the engine makes less jet noise. Engines of this type are known as by-pass engines. The ratio of unheated (cold) air, which bypasses the combustion process and turbine, to (hot) air heated by combustion, is known as the by-pass ratio. In this case the ratio is less than 2:1 and is known as a low by-pass ratio

FIGURE 19-15
Twin-Spool Turbo Shaft Engine

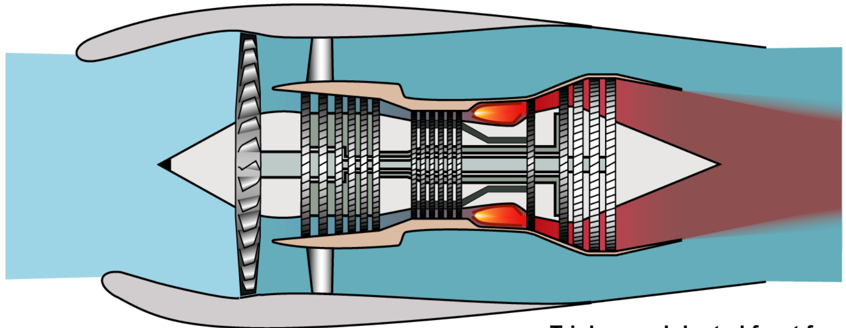


Twin spool axial flow turbo-shaft

32. Figure 19-15 shows a twin spool turbo-shaft engine in which there is a third rotating assembly comprising a turbine driving a power shaft connected to an output gearbox. Such a turbine is known as a free, or power, turbine. The Rolls Royce Advanced Turbo-Prop engine is an example of such an arrangement. It is still a twin spool engine because, although there are three shafts, there are only two turbine/compressor spools.

Triple-Spool Engines

FIGURE 19-16
Triple-Shaft
Ducted Fan Engine



Triple spool ducted front fan
(High by-pass ratio)

33. Figure 19-16 shows a triple spool ducted fan engine. The three spools are arranged concentrically with the HP turbine driving the HP compressor, the LP turbine driving the LP compressor and a final turbine driving a ducted fan. In such an engine the mass of air accelerated by the fan is at least five times greater than the mass passing through the engine. With virtually no tip losses, and acceleration of the fan air through a propelling nozzle, the thrust produced by the fan is much greater than would be produced by a propeller driven with the same shaft horsepower. Hence, the propulsive efficiency of this arrangement is better than that of the low by-pass engine and is maintained to higher airspeed that that of the turbo-prop (see Figure 19-9).

Gas Turbine Construction Part I – The Cold Section

Gas Turbine Construction Part 1 – The Cold Section

Introduction

1. This study considers in detail the major components which, together, make up the turbine engine. These may be sub-divided into those which make up the static assembly of the engine - the air intake, combustion chambers, exhaust and propelling nozzle and those which make up the rotating assembly of the engine - compressors and turbines.

2. Put into a sequential order these are; air intake, compressor, combustion chamber, turbine, exhaust and propelling nozzle.

3. The subject is divided into three parts for convenience. Part 1 covers the cold section, which comprises the air intake, the compressor and the diffuser. Part 2 covers the hot section, which includes the combustion chambers, turbine, exhaust and propelling nozzle. Part 3 considers the working cycle of the engine, thrust reversal and augmentation, effects of air bleed and the auxiliary gearbox.

Air Intakes

4. Normally the air inlet duct is considered to be an airframe component. Nevertheless, it contributes to efficient engine performance. The functions of the air intake are threefold. Firstly it is required to admit the maximum amount of air from the free stream to the engine. Secondly, it should diffuse the airflow, with minimum loss, so as to deliver the air to the compressor at the maximum total pressure with minimum disturbance (turbulence). Ideally, this delivered pressure should be as close as possible to total free stream pressure. Thirdly it must contribute as little as possible to aerodynamic drag.

5. In subsonic aircraft the inlet ducts are divergent and therefore act as diffusers, decreasing the velocity and thus raising the static pressure of the incoming air. With supersonic aircraft the problem is complicated by the fact that a divergent passage is required at subsonic speeds, to create an increase in static pressure, whereas at supersonic speeds a convergent duct is needed for the same effect. Inlet ducts on supersonic aircraft often have variable geometry.

Pitot Type Intakes

6. The optimum design of air intake for subsonic aircraft is the pitot type that is typically used in conjunction with podded engines mounted on wing pylons or on stub wings on the side of the rear fuselage. By minimising the diameter of such intakes, drag is reduced but this can result in unacceptable restrictions to airflow to the engines when running at high power setting on the ground. This problem can be overcome by the use of secondary air inlet doors around the periphery of the intake. At high power settings when stationary on the ground the engines demand for high airflow causes airflow to accelerate into the intake. This reduces static pressure in the intake causing the secondary doors to open to increase inlet area.

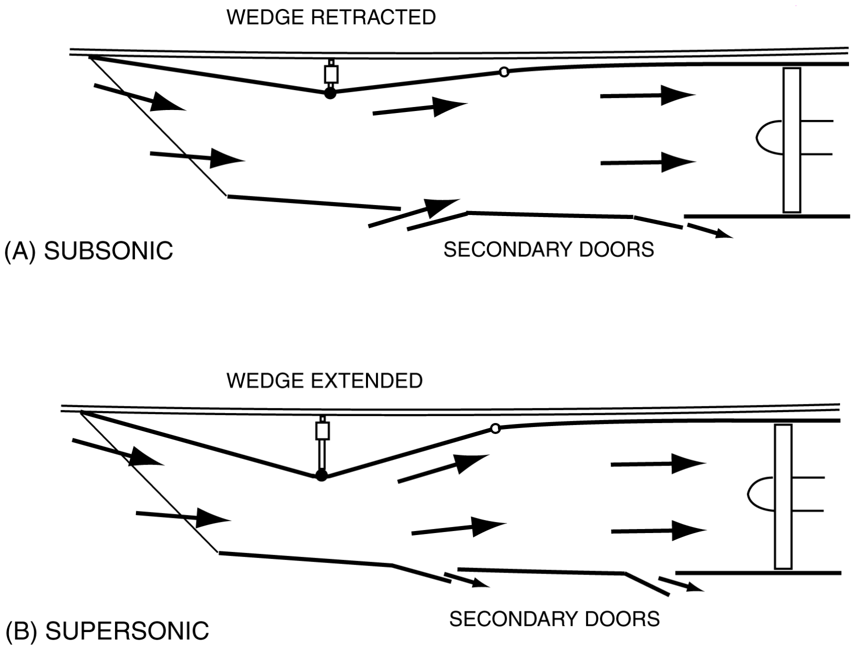
7. At low to moderate (sub-sonic) speeds, the intake acts as a divergent duct, converting some of the dynamic pressure of the incoming air into static pressure. This causes a reduction in air velocity and an increase in air temperature.

8. At high subsonic and transonic speeds there is a marked tendency for shock waves to form at the lip of such intakes, seriously reducing their efficiency.

Multi-Shock and Variable Area Intakes

9. Aircraft that operate in the transonic and supersonic range require a type of air intake that reduce the shock waves and, in the case of high Mach numbers, permit some of the excess air due to ram effect to be spilled back to atmosphere before it reaches the compressor inlet. Control of the shock waves can be achieved with the use of a variable area intake, and spilling of excess air can be achieved with secondary inlet doors fitted in the intake casing. These secondary doors often perform two functions, acting as a scoop in subsonic conditions to augment the airflow to the compressors and as a spill valve in supersonic flight to dump excess air. Figure 20-1 shows a variable area intake with secondary doors of the type fitted to Concorde.

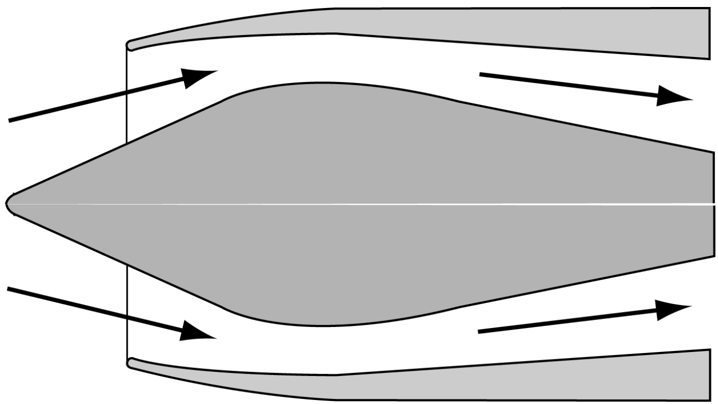
FIGURE 20-1
Variable Area Intake



10. The purpose of the variable area intake is to position the shock wave in such a way as to decrease the velocity of the airflow at the compressor inlet, whilst maintaining the total pressure within the duct as close as possible to ambient. Many modern military supersonic aircraft use specially designed intake ducts with secondary doors that are controlled in all stages of flight to achieve shock wave and duct pressure control without the need for variable geometry intakes.

11. Some early supersonic aircraft used a type of intake known as an external/internal compression intake, as illustrated at Figure 20-2. This achieved the desired effect by producing a series of mild shock waves in the intake throat, rather than a single strong shock wave at the lip.

FIGURE 20-2
External/Internal Compression Intake



12. **Airflow Separation.** Where the engine intakes are mounted on the side of the fuselage there is a danger of airflow separation when operating in strong crosswinds on the ground, or if the aircraft is allowed to sideslip in flight. This is because the air intake becomes partially obscured by the fuselage. It is especially the case with intakes buried in, or adjacent to, the wing root, but can also occur with rear fuselage mounted engines. In extreme cases the disruption of the airflow can be sufficient to cause engine flameout.

13. **Inlet Icing.** The formation of ice on and in the air intake can have serious effects on compressor and engine performance, due to changes in the effective inlet area and duct shape. There is also a considerable risk of flame-out and of compressor damage if significant amounts of ice are ingested. Engine inlet ducts, nose cowls and structural interconnecting vanes are invariably fitted with thermal anti-ice protection. It is important to appreciate that this system (which normally uses hot bleed air from a late compressor stage) is an anti-ice and not a de-icing system, since use of the system for the latter purpose would inevitably lead to ice ingestion. If, due to pilot inattention, de-icing of the engine intakes becomes necessary, the engines should be de-iced one at a time with the igniters switched ON to avoid flame-out in the event of ice ingestion.

Hazards and Factors Affecting Intake Efficiency

14. **Inlet Damage.** Distortion or damage to the air intake, especially at the lip, will disrupt the airflow into the inlet and adversely effect the duct pressure and velocity, resulting in loss of intake efficiency and possibly disturbed airflow at the compressor entry. At transonic and supersonic speeds such damage will disrupt the shock wave pattern and cause serious disruption of the airflow into the intake duct.

15. **Heavy In-Flight Turbulence.** Flight in heavy turbulence may give rise to airflow separation at the intake and cause significant disruption of the airflow to the compressor inlet, possibly to the extent of causing compressor stall or surge. Flameout of the engine is a possibility in such circumstances. If flight in heavy turbulence cannot be avoided it is a wise precaution to select engine ignition on (continuous).

16. **Foreign Object Ingestion.** Gas turbine engines, and particularly the compressor or fan, are highly prone to damage through the ingestion of foreign objects. The carefully profiled fan or compressor blades are easily damaged or broken if solid objects, such as stones from the ground surface or carelessly discarded tools or loose material in the intake duct, are drawn into the engine when it is rotating at high speed. More importantly, even minor damage to the highly stressed fan and compressor blades is likely to lead to fatigue failure as fatigue cracks grow from minor nicks and scratches. From the foregoing text on intake design, it is clearly impractical to mount protective mesh screens in the intake, and so great care must be taken to ensure that no loose material can enter the highly vulnerable intake.

17. Careful pre-flight inspection of the intakes and the avoidance of jet/propeller wash from other taxiing aircraft will help to prevent foreign object ingestion during ground operations. Such precautions are particularly important when operating at high power settings on the ground.

Compressors

18. Compressors in gas turbine engines are designed to achieve the following functions:

(a) Increase the air mass flow.

(b) Improve combustion characteristics.

(c) Increase the efficiency of the operating cycle.

(d) Increase the thrust produced by the engine.

(e) Improve fuel economy.

(f) Assist in the provision of a small and compact engine.

19. The purpose of the compressor is to increase the total energy of the air received from the inlet duct, compress it and discharge it into the combustion chamber in the right quantity and at the required pressure. In the compressor work is done upon the air to compress it adiabatically, and so the temperature of the air increases in direct proportion to the pressure.

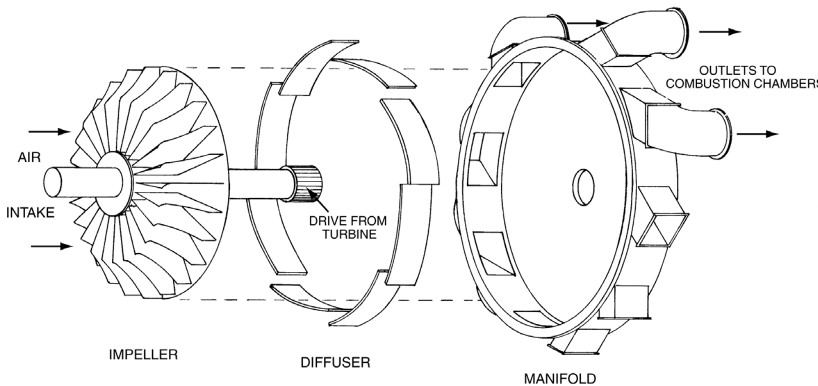
20. The amount of air passing through the engine depends upon compressor rpm, the atmospheric conditions at the engine inlet, such as the air pressure, density and temperature and the aircraft speed. The pressure ratio of a compressor is the ratio of its outlet static pressure to its inlet static pressure.

21. Most gas turbines use continuous flow rotary compressors, and two types are currently in use. They are centrifugal compressors (radial flow), and axial compressors (straight through flow). Both types of compressor work on the same general principle of imparting kinetic energy to the air in a high speed rotor and converting this energy into pressure in a set of divergent flow passages. Of the energy available in the gas after the combustion process, approximately 60 per cent is needed to drive the compressor and the engine accessories (generators, hydraulic pumps, etc).

Centrifugal Compressors

22. The centrifugal compressor consists of three main components, as shown at Figure 20-3.

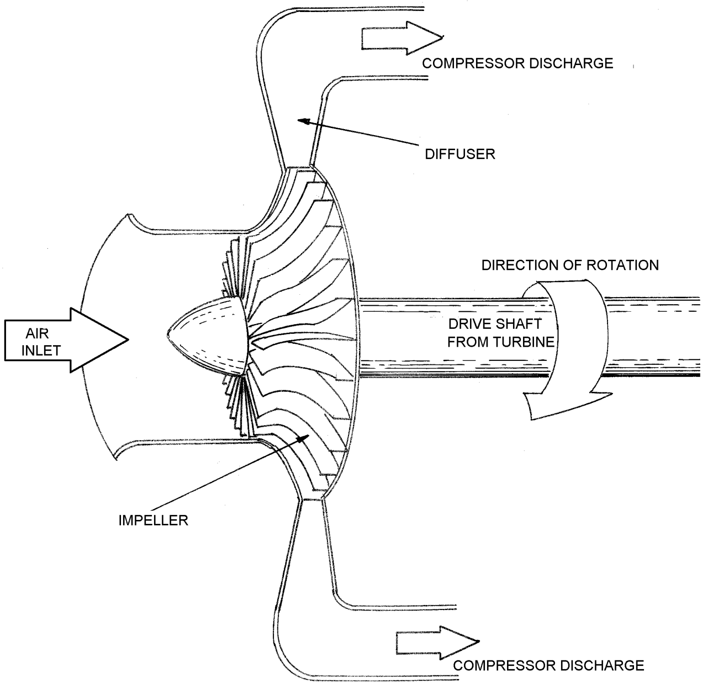
FIGURE 20-3
Centrifugal Compressor



23. The rotating assembly is called the impeller, which draws in air and discharges it through stationary divergent passages called a diffuser. The diffuser is mounted in an outer casing, which collects the diffuser discharge and delivers it to the combustion chambers through a manifold. The impeller may be single sided, as shown in Figure 20-3 or double sided as shown in Figure 20-5.

24. A simplified diagram of a centrifugal compressor, showing the airflow through it, is at Figure 20-4. The impeller is rotated at high speed by the turbine. Centrifugal force causes the air in the radial passages formed by the impeller vanes to be thrown outwards to the impeller tip, or circumference. Air is drawn into the centre, or 'eye', of the impeller to replace that thrown outwards. The radiating vanes on the face of the impeller form divergent passages, which cause a rise in pressure as air is thrown outwards through them.

FIGURE 20-4
Simplified Centrifugal Compressor



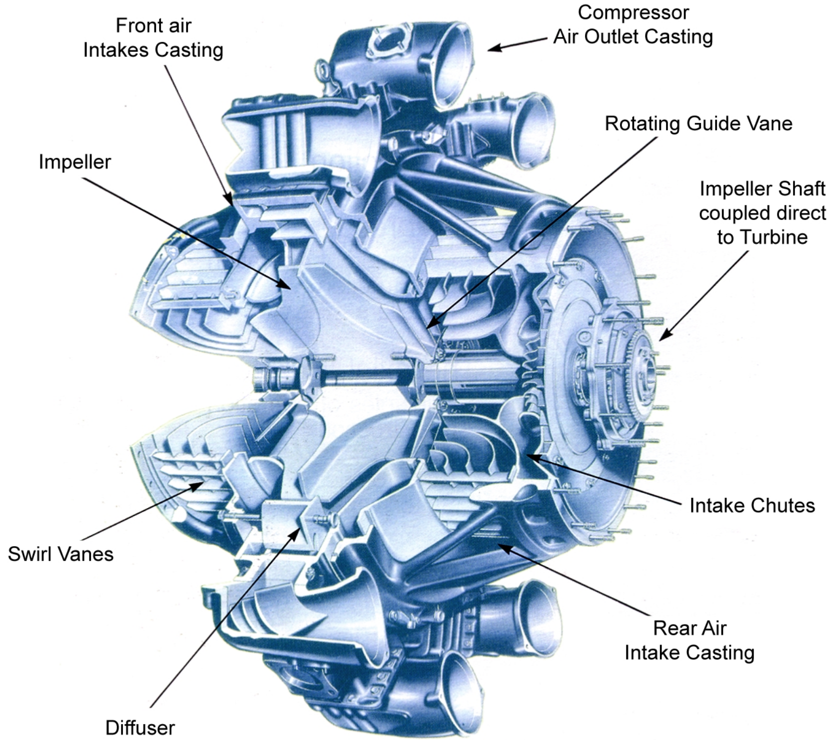
25. On leaving the impeller the air enters stationary divergent passages (diffuser ducts) in the casing of the compressor. These convert the kinetic energy of the air, due to its velocity, into pressure energy. The total pressure rise achieved by a centrifugal compressor is shared approximately equally between impeller and diffuser. Typical compression ratio (outlet pressure to inlet pressure) for a centrifugal compressor is about 4:1. The overall effect of the compression process is an increase in static pressure and temperature with little or no change in velocity.

26. As air flows through the radial passages between impeller and casing its velocity increases due to centrifugal action and its pressure increases due to the divergent passages formed by the impeller vanes. As the air flows through the stationary diffusers its velocity decreases and its pressure increases due to diffusion. From the diffusers the air is ducted to the combustion chambers.

27. The engine designer has the choice of either single entry or double entry centrifugal compressors (see Figure 20-5). The single entry compressor is claimed to give the best all-round efficiency with less risk of surging (to be discussed shortly) at altitude. The double entry compressor achieves greater mass air flow for less diameter, and therefore less frontal area. This is an advantage to the airframe designer, but the air to the rear side of the impeller tends to be preheated as it passes the discharge from the front side and this is a disadvantage as far as the engine designer is concerned. In some engines two single sided compressors in series (one after the other) are used, the Rolls Royce Dart is an example.

28. Where a double entry centrifugal compressor is employed it will double the mass flow of air entering the engine. Because each particle of air passes through only one side of the compressor it is subject to only one stage of compression. The double entry compressor does not therefore produce a higher pressure ratio than a single entry compressor. If higher pressure ratios are required compressors are arranged in series so that air passing through the first is then directed through the second. The overall pressure ratio of such an arrangement is the product of those of all the stages. Two compressors each of 4:1 pressure ratio for example would achieve approximately 16:1 if used in series.

FIGURE 20-5
Double Entry
Centrifugal
Compressor



29. Centrifugal compressors suffer from a number of disadvantages that have rendered them unsuitable for use in aircraft propulsion engines in all but a very few cases. The principal disadvantages of centrifugal compressors are:

(a) Low compression ratio (typically about 4:1) and therefore low air mass flow.

(b) Larger frontal area (increased aerodynamic drag).

(c) Impeller tip clearance is critical. If the tip clearance is too great the pressure losses will be excessive, if it is too small aerodynamic buffeting results.

30. However, centrifugal compressors have certain advantages, which have resulted in their continued use in smaller engines, such as auxiliary power units (APU) and helicopter turbo-shaft engines. The advantages of centrifugal compressors are:

(a) They are more robust and less prone to ingestion damage.

(b) They are less prone to stall and surge (lower pressure rise).

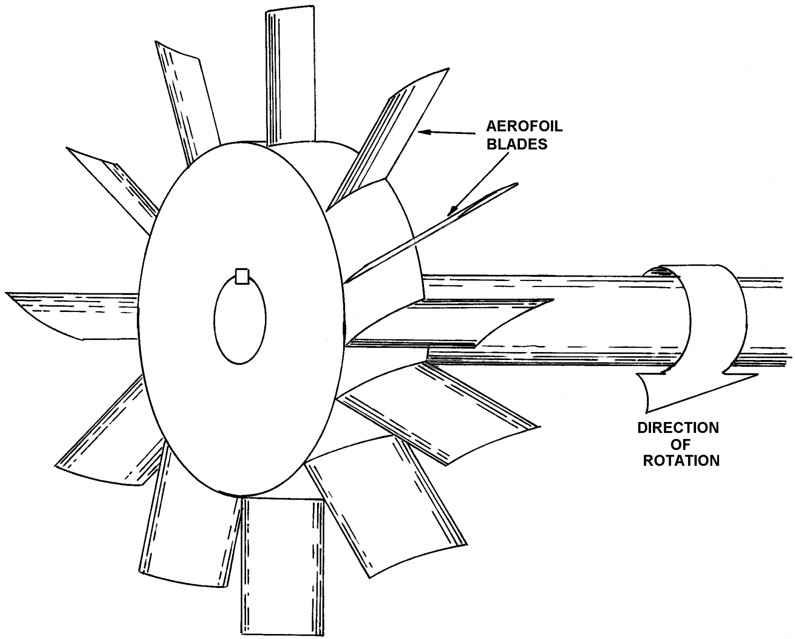
(c) They are much less expensive to produce than axial compressors.

(d) They are shorter and therefore occupy less space longitudinally. Consequently they are sometimes used as the final stage compressor in a multi-spool engine to reduce its overall length.

Axial Compressors

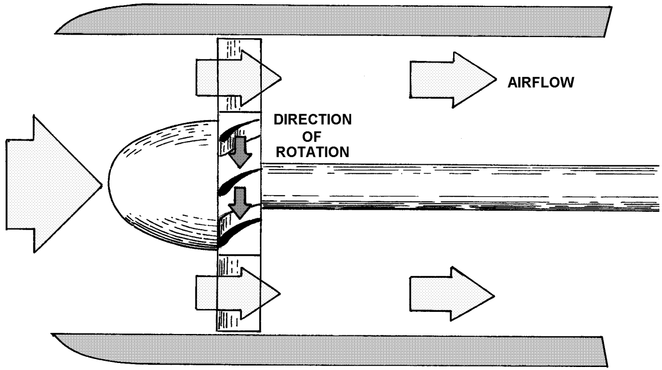
31. An axial flow compressor consists of a rotating assembly upon which aerofoil section blades are mounted. As the compressor rotor is spun at high speed by the turbine, the aerofoil blades induce airflow in the same way that a propeller blade or the wing of an aircraft induces airflow. This principle is illustrated at Figure 20-6.

FIGURE 20-6
Simplified Axial-Flow Compressor



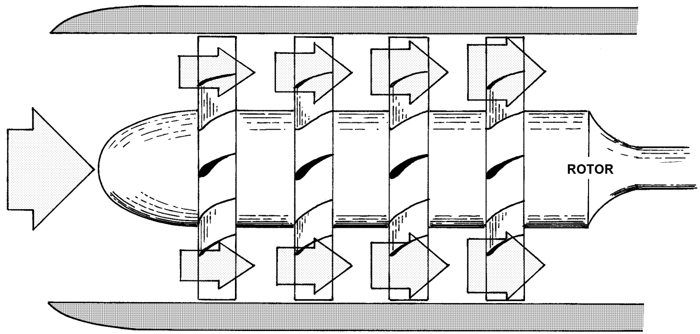
32. The rotating assembly is mounted within a stationary casing so that the induced airflow is ducted to the combustion chambers, turbine, and so on, as shown at Figure 20-7.

FIGURE 20-7
Axial Flow Compressor Induced Flow (One Stage)



33. Adding further rows, or stages, of compressor blades, as shown at Figure 20-8, increases the air pressure ratio.

FIGURE 20-8
Axial Flow Compressor Induced Flow (Multi Stage)



34. In order to ensure that air enters the rotating blades at the optimum angle of attack, inlet guide vanes are often fitted in the air intake. These are attached to the stationary casing (stator) of the compressor and angled to direct the incoming air into the blades. This is illustrated at Figure 20-9. The operation of inlet guide vanes is discussed later in this Chapter.

FIGURE 20-9
Rotor Blade Angle of Attack

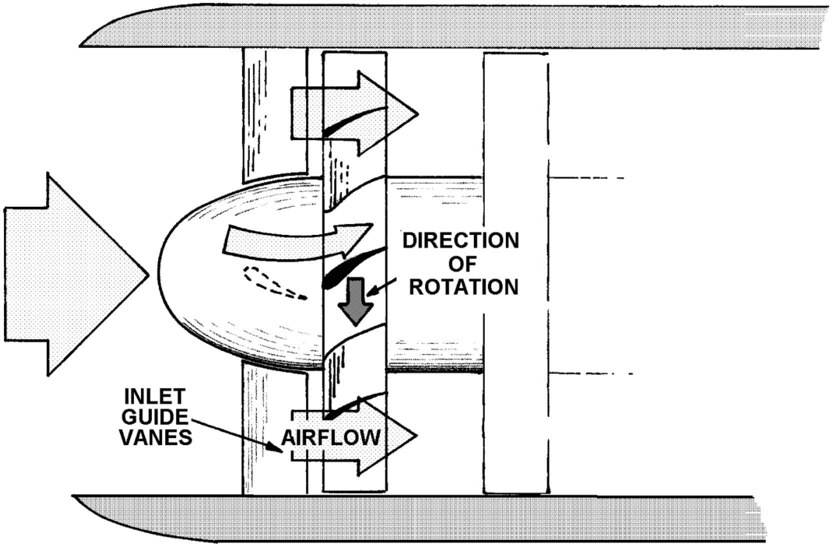
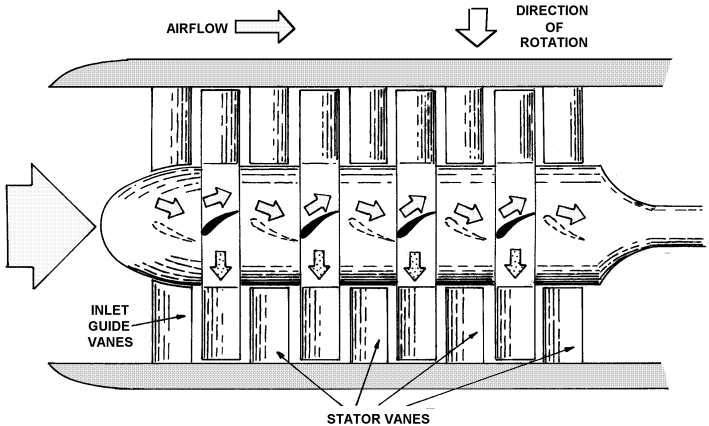
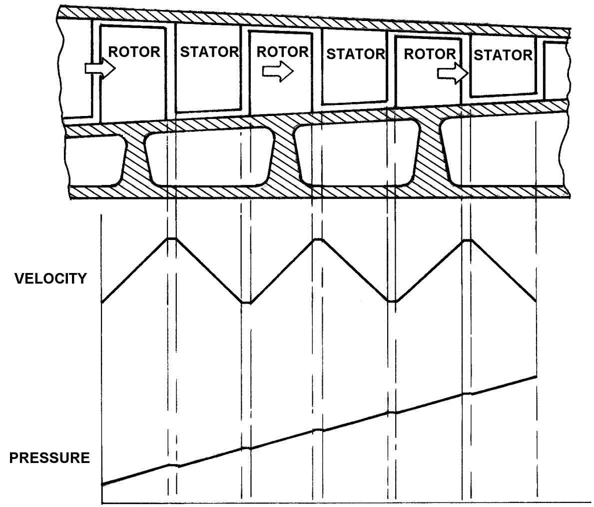


FIGURE 20-10
Axial-Flow Compressor Stator Vanes



35. The rotor blades of the compressor impart kinetic (velocity) energy to the air. This is partly converted into pressure energy because the air passage between adjacent blades is slightly divergent. From the rotating blades the airflow enters the stationary vanes. The air passage between these is also slightly divergent and so a further pressure rise occurs as kinetic (velocity) energy is further converted into pressure energy. The changes in pressure and velocity that occur across a multi-stage axial flow compressor are illustrated at Figure 20-11.

FIGURE 20-11
Pressure / Velocity Changes through Axial-Flow Compressor



36. The pressure ratio across each stage is less than 2:1 because the amount of diffusion has to be limited to avoid aerodynamic stalling of the blades. Nevertheless, the pressure rise is cumulative stage-by-stage and a compression ratio of up to 35:1 across the whole of a multi-stage compressor is not unusual. At such a compression ratio the outlet temperature will be up to 600°C

37. The simple axial compressor shown at Figure 20-10 has blades of equal length in all stages. As already explained, a pressure rise occurs across each stage, as velocity energy is converted to pressure energy in the stator (diffuser) vanes. Thus, across all the stages of such a machine there would be an overall velocity decrease and an overall pressure increase. Hence, it would be difficult to maintain airflow against the increasing pressure, or adverse pressure gradient.

38. In order to maintain uniform axial velocity, and thus keep the air flowing through the compressor from front to rear, the axial air passage is made convergent. Consequently, the rotor blades and stator vanes must be made progressively shorter from front to rear of an axial flow compressor, in order to be accommodated in the convergent passage.

Compressor Stall and Surge

Compressor Stall

39. The natural tendency of any fluid when acted upon by a pressure gradient is to flow from the high pressure area to the lower pressure area. Figure 20-11 illustrates the changes in air pressure through an axial flow compressor and from this it can be seen that the air is required to flow against an adverse pressure gradient, from an area of low pressure to one of higher pressure. For this flow to be possible it is necessary that the sum of static pressure and dynamic pressure acting downstream is greater than that acting upstream. This will be the case provided the air mass flow and compressor RPM are within the design specification for the compressor. Any factor that restricts the airflow into the compressor, causes an excessive increase in the adverse pressure gradient through it, or results in an unacceptable mass flow/RPM combination will cause the airflow to break down. Such break downs take one of two forms, compressor stall and compressor surge.

40. The blades of axial flow compressors are like all other aerofoils in that the airflow over them varies with angle of attack, and will detach and break down when their stalling angle is exceeded. In the case of compressors this phenomenon is termed compressor stall. Stall is brought about by any factor that causes the airflow to approach any single blade or group of blades at an angle greater than the stalling angle of those blades. Stall can therefore affect a single blade or stage or a group of blades or stages. When the entire compressor stalls the resulting total breakdown of airflow is termed a compressor surge. Compressor stall is illustrated in Figure 20-12.

41. Factors that are likely to promote compressor stall include the following:

(a) Low engine RPM during starting and idling.

(b) Strong crosswinds on the ground.

(c) Engine inlet icing.

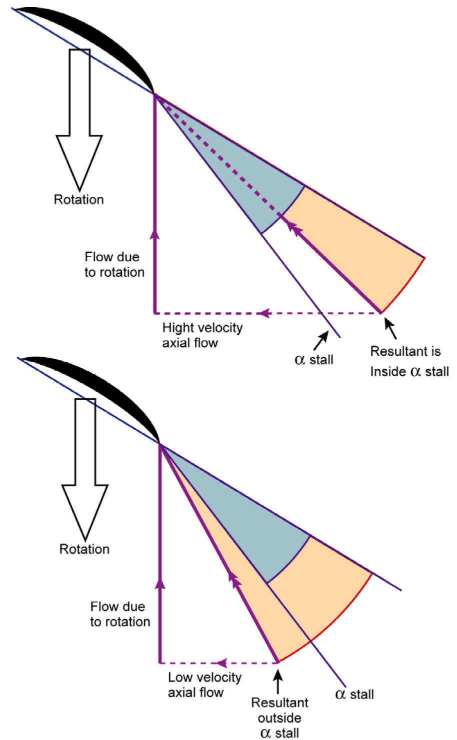
(d) Contaminated or damaged compressor blades.

(e) Damaged air intake.

(f) High angle of attack at the air intake caused by aircraft manoeuvring.

(g) Excessive increases in combustion chamber pressure caused by rapid opening of the throttle particularly at low engine RPM.

FIGURE 20-12
Compressor Aerodynamic Stall



Compressor Surge

42. The term compressor surge is generally taken to refer to the phenomenon in which a total breakdown of airflow through the compressor occurs. In some cases a surge also results in intermittent reversal of airflow through the compressor. In all cases the result is a loss of thrust often coupled with vibration, loud banging, rumbling or popping noises and rapidly increasing EGT. Although a great many scenarios might result in surge, most fall into one of three categories, all of which involve the creation of an unacceptably adverse pressure gradient through the compressor. The three common causes are aerodynamic stall of the compressor, unacceptable RPM/mass flow combination at low RPM, and unacceptable increases in combustion chamber pressure.

43. Whilst the efficiency of the compression process varies with RPM and air mass flow, the rate of reduction of cross-sectional area of the compressor annulus is fixed and can be perfectly matched to only one RPM/mass flow combination. When operating outside this combination, inefficiencies occur due to the air being either over compressed or under compressed to match the reducing cross-section. When operating at low RPM inadequate compression causes the volume of the airflow to be reduced insufficiently, causing the rear section of the compressor to choke. In effect the air is too big to pass through the casing at the intended uniform velocity. Under these circumstances airflow breaks down causing the blades in the upstream stages of the compressor to suffer aerodynamic stall. This further decreases the efficiency of the compression process exacerbating the choking of the rear stages.

44. For any given combination of engine RPM and air mass flow there will be a pressure ratio above which the compressor will surge. If these points are plotted for all practicable combinations it is possible to produce a map of the surge envelope of the engine. An example of a surge envelope map indicating the effects of rapid throttle opening at high and low RPM is illustrated in Figure 20-13. The surge line indicates the pressure ratio above which surge will occur. The working line indicates the point at which the engine will lie when operating in any steady state combination of RPM, air mass flow, and compression ratio. The distance between the working line and the surge line at any given RPM is a measure of the engines resistance to surge.

45. In order to provide an adequate margin between normal operating conditions and the surge line a working line defining normal engine steady state operating conditions is set. In order to increase the speed of the engine the throttle is opened increasing fuel flow to the burners. This increases combustion chamber pressure which eventually causes the turbine and compressor to accelerate to a higher point on the working. Because the combustion chamber pressure rise occurs before the increase in RPM and mass flow, the engine initially moves closer to the surge line before returning to the working line at a higher RPM as indicated by points A, B and C in Figure 20-13. In the case of excessively rapid throttle opening, the surge line will be crossed initiating a compressor surge. If the surge is sufficiently severe or prolonged, rising temperatures and loss of RPM and mass flow will prevent the engine from returning to the working line without pilot intervention. In extreme cases the engine will move deeper into the surge area rather than recovering as indicated by points D, E, F and G in diagram Figure 20-14. Because the margin between the working line and the surge line increases with RPM, surge is particularly likely when starting or accelerating from low RPM. To overcome this problem a number of stall and surge prevention devices are employed.

FIGURE 20-13
Surge Envelope Map Indicating the Effects of Acceleration at High RPM

A→B INDICATES INITIAL RISE IN PRESSURE RATIO DUE TO RAPID OPENING OF THROTTLE.

B→C INDICATES ENGINE RECOVERY TO THE WORKING LINE AT NEW RPM, MASS FLOW AND PRESSURE RATIO.

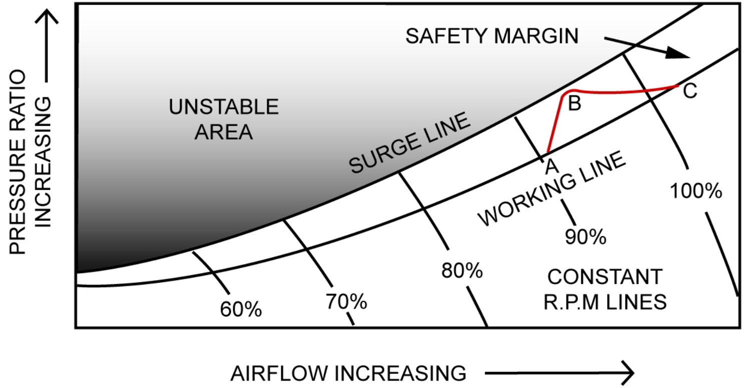
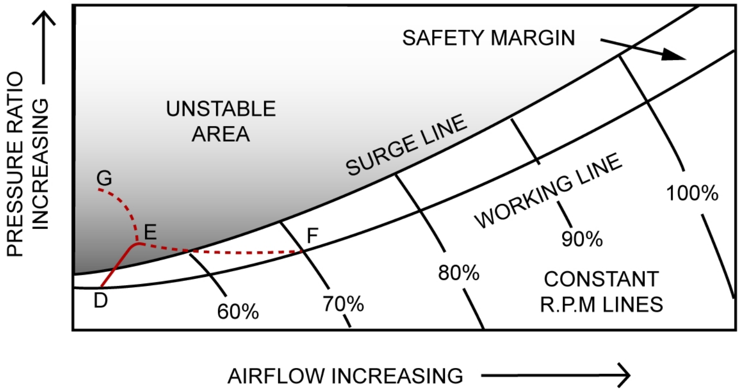


FIGURE 20-14
Surge Envelope Map Incicating the Effects of Acceleration at Low RPM

D→E INDICATES THE EFFECT OF THE INITIAL RISE IN PRESSURE RATIO DUE TO RAPID OPENING OF THROTTLE..

E→F INDICATES THE RECOVERY TO THE WORKING THAT WOULD OCCUR IN THE ABSENCE OF SURGE

F→G INDICATES ACTUAL PATH TAKEN AS RPM AND MASS FLOW REDUCE DUE TO THE EFFECT OF STALL



Indications of Stall and Surge

46. The existence of a stall and/or surge condition is indicated by any or all of the following symptoms:

(a) Unexpected loss of thrust. Because of the reduction in mass flow caused by stall and surge the thrust output of the engine is reduced. The magnitude of this effect depends upon the severity of the stall/surge.

(b) Abnormal engine noises and vibrations. Because of their effect on the airflow through the engine stall and surge usually result in the production of loud popping, knocking or rumbling noises and vibrations from within the engine.

(c) Uncommanded variations in engine RPM. The disruption of airflow and intermittent reversal of airflow often cause rapid variations in engine RPM.

(d) The breakdown of airflow through the engine coupled with continuing combustion results in rapidly increasing EGT.

(e) In extreme cases the reversal of airflow can result in the ejection of exhaust gasses out of the air intake.

Stall and Surge Prevention Devices

Variable Inlet Guide Vanes

47. In order to prevent compressor stall it is necessary to ensure that the airflow approaches the blades at an angle lower than the stalling angle under all operating conditions. One method of achieving this under conditions of low RPM is to employ variable angle inlet guide vanes (VIGVs) situated immediately in front of the first stage of the compressor. By varying the angle of the blades to match prevailing mass flow and RPM conditions the air is directed onto the first stage rotor blades at an acceptable angle. In many engines this technique is extended to include variable angle stator blades in several stages of the compressor. The effect of VIGVs and the general arrangement of VIGVs and variable angle stators are illustrated in Figure 20-15 and Figure 20-16.

FIGURE 20-15
Effect of Variable Angle Inlet Guide Vanes in Preventing Compressor Stall

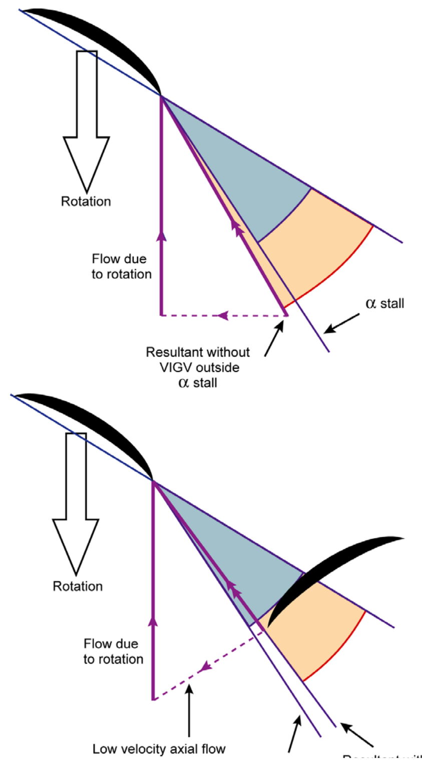
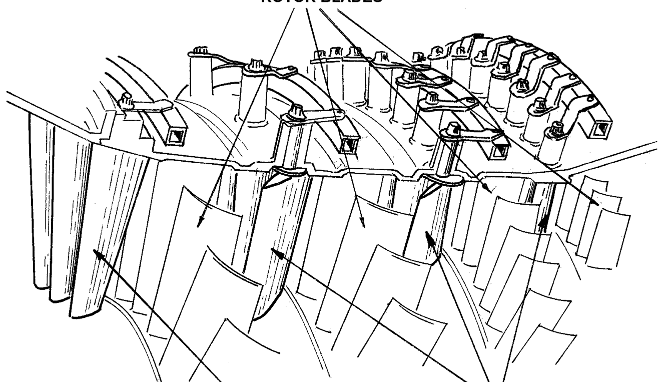


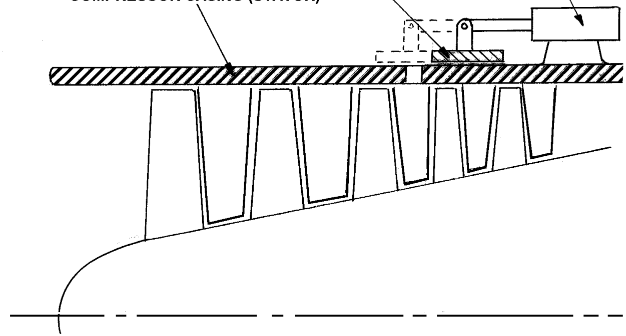
FIGURE 20-16
Variable Incidence Stator Vanes in Axial Flow Compressor



Anti-Surge Bleed Valves

48. The effects of low compressor efficiency at low RPM and the resulting surge are often relieved or prevented by means of anti-surge bleed valves. These are situated in the compressor casing towards the mid-section of the compressor. At low RPM they are open allowing excess air to exhaust to atmosphere thereby preventing the choking of the rear stages of the compressor. As RPM increases they automatically close off to increase mass flow through the engine and improve engine efficiency. The general arrangement of these valves is illustrated in Figure 20-17.

FIGURE 20-17
Surge Bleed Valves



49. In multi-spool engines the speed of spools can vary independently to ensure that each is matched to the prevailing mass flow conditions. Also high pressure spools tend to be relatively small and because of their low inertia, are able to react very quickly to increases in throttle angle, accelerating rapidly to the required higher RPM. This tends to prevent the creation of unacceptably high adverse pressure gradients in the low pressure and intermediate pressure spools. In this way the need for variable inlet guide vanes and anti-surge valves is reduced (but not eliminated) in such engines.

Pilot Actions to Prevent and Correct Compressor Stall and Surge

50. The principal actions to be taken by pilots to prevent stall are as follows:

(a) Ensure that all engine blanks and covers are removed before attempting to start the engine.

(b) Ensure that the aircraft is facing into wind before starting to avoid intake airflow disturbance and the ingestion of exhaust gas.

(c) Avoid following too close behind other aircraft when taxiing to avoid ingestion of exhaust gas.

(d) Avoid air turbulence and harsh flight manoeuvres whenever possible.

(e) Allow the engine to stabilise at idle RPM after start up before attempting to accelerate to higher speeds.

(f) Anticipate the need for power changes in order to avoid or minimise rapid throttle movements.

(g) Ensure that the appropriate power setting is used for all stages of flight

51. The following actions should be taken by the pilot in the event of compressor stall or surge:

(a) Reduce the rate of throttle opening if stall/surge occur during engine acceleration.

(b) If stall/surge persists reduce throttle setting as soon as other safety considerations permit.

(c) Terminate the start cycle immediately if stall/surge occur during starting.

(d) Reduce flying control deflections as soon as it is safe to do so if stall/surge occur during flight manoeuvres.

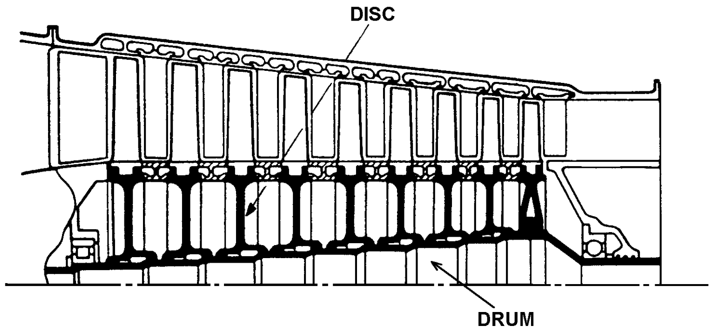
Axial Compressor Construction

52. The compressor rotating assembly, or rotor, upon which the rows of blades are mounted is supported in a number of ball and roller bearings. The rotor shaft is coupled to the turbine shaft to form a single spool.

53. The compressor cylindrical casing in which the rows of stator vanes are mounted, is usually made in two halves with a horizontal longitudinal centreline joint. The rotor shaft bearings are, typically, at either end of the stator.

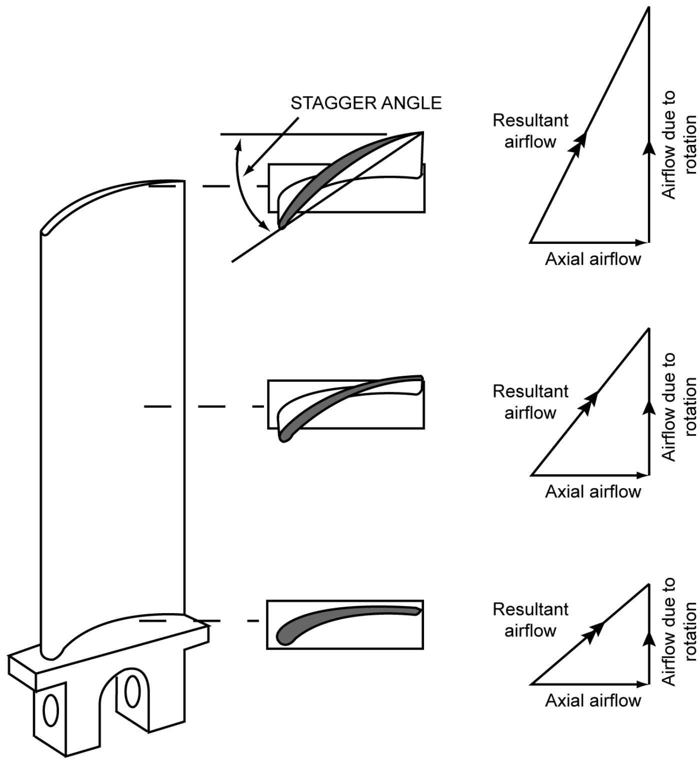
54. The rotor is usually made up of a number of discs, with the blades attached to the outer periphery of the discs. This method of construction is shown at Figure 20-18. The discs form an integral drum of low weight, but sufficient strength to withstand the centrifugal loads imposed by the blades at high rotational speeds

FIGURE 20-18
Rotor Disc Construction



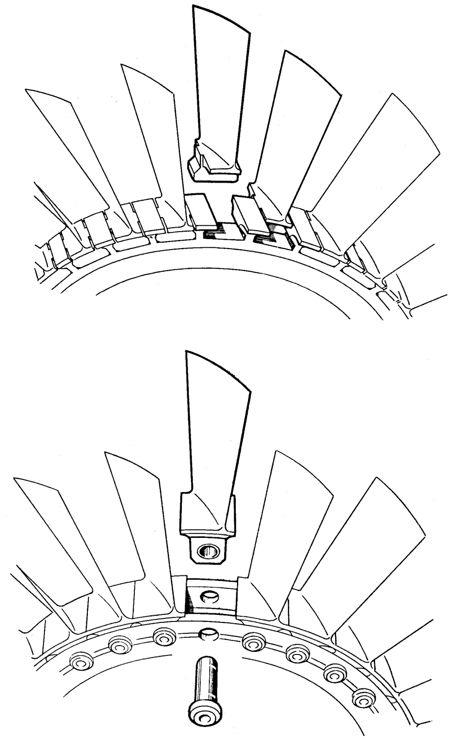
55. The rotor blades are of aerofoil section and are twisted such that the stagger angle measured between the blade chord and the axis of rotation increases from root to tip. This is intended to achieve two effects. Firstly because the axial airflow velocity is constant along the blade, whilst the airflow velocity due to rotation increase with radius, the angle of attack of an untwisted blade would increase from root to tip. Such a blade would therefore be liable to exhibit inefficiently low angles of attack at the root, coupled with excessively high angles of attack at the tip. Secondly the rotors impart a swirl to the airflow generating a centrifugal force tending to deflect it outwards towards the tips of the blades. To counteract this effect it is necessary to create a pressure gradient along the length of the blades. The rate of twist of the blades is designed to provide this span-wise pressure gradient whilst ensuring acceptable angles of attack and uniform axial airflow velocities across the rotor disc. The varying airflow velocities and stagger angle are illustrated in Figure 20-19.

FIGURE 20-19
Compressor Blade Twist



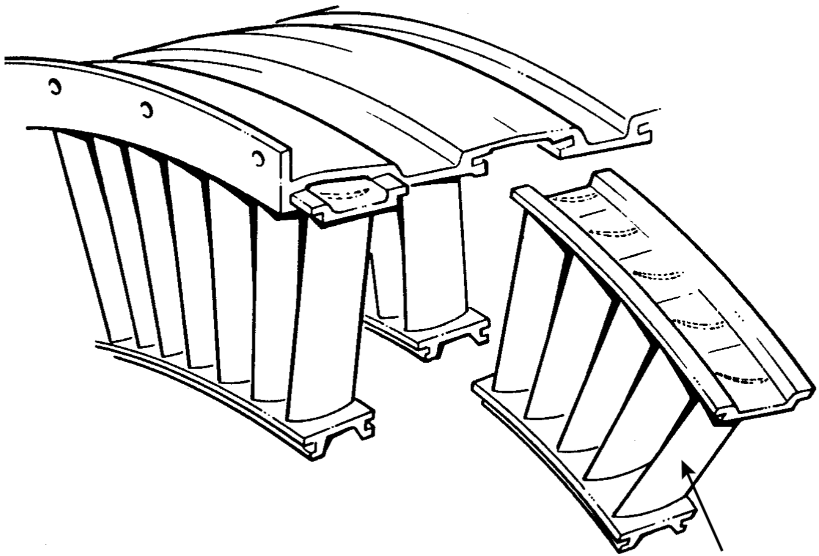
56. The rotor blades are secured to the discs in such a way that the load imparted to the disc is minimised. Examples are shown at Figure 20-20. The compressor blades are often free to rock or slide slightly in their root mounting in order to relieve stress concentration at the root due to centrifugal force. This "looseness" gives rise to the characteristic clicking noise often heard when the compressor rotates at very low rpm, such as when it is windmilling with the engine shut down on the ground.

FIGURE 20-20
Rotor Blade
Stagger
Attachment



57. The stator blades are also of aerofoil section and are secured in annular grooves in the casing. The blades may be joined together at their inner ends, to reduce any tendency to vibrate. The shrouding also forms a smooth surface along the annulus and reduces losses by monitoring leakage around the blade tips. An example of shrouded stator vanes is shown at Figure 20-21.

FIGURE 20-21
Shrouded Stator Vanes



58. Compressor rotor blades, discs and drums are usually made of titanium, which has a very high strength-to-density ratio and is thus able to withstand the high centrifugal forces whilst maintaining low weight. Stator blades are usually made of steel or nickel based alloys, an important feature being the need to retain high strength even when 'notched' by foreign object ingestion damage (FOD).

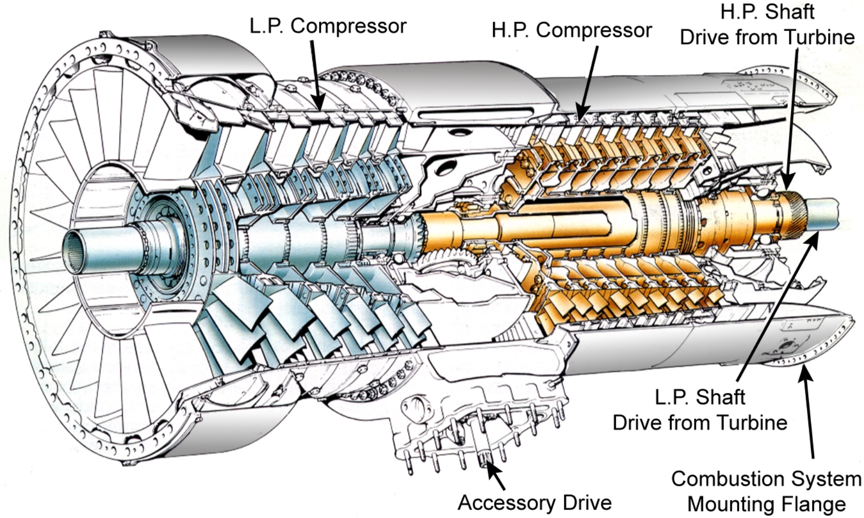
59. The very long blades used in high by-pass ratio fan engines have a tendency to bend and 'flap' under high aerodynamic loads. To prevent this, a mid-span support (known as a 'snubber' or 'clapper') is used, forming a strengthening ring when the fan blades are mounted on their rotor. Under the effects of shock, such as FOD ingestion, bird strike, overspeed or surge these snubbers may overlap each other - a condition known as 'shingling'.

Multi-Spool Compressors

60. Single spool compressors are used in pure jet engines, that is to say engines with no by-pass air, in which all the air mass flow is heated by combustion, expanded to power the turbine and accelerated by the propelling nozzle.

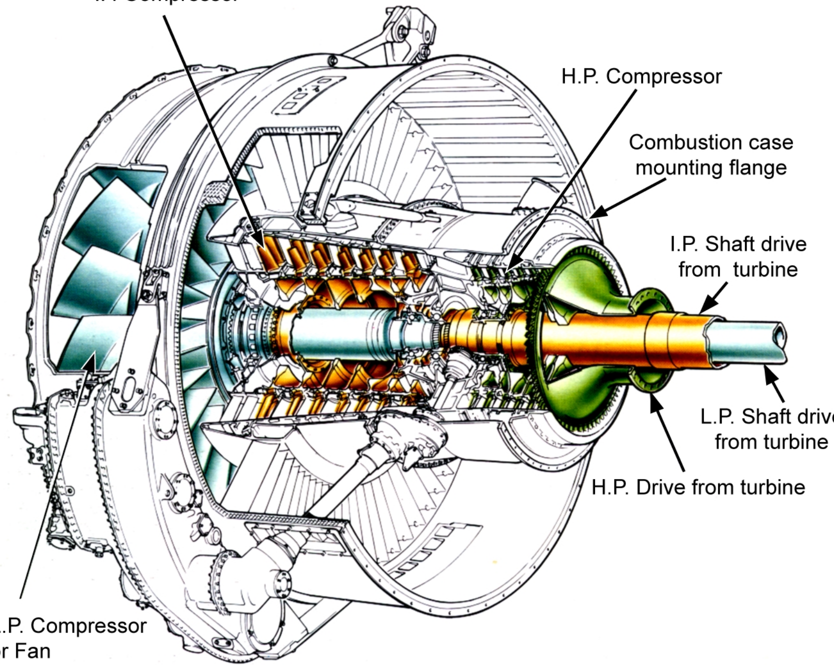
61. Low by-pass engines are of the twin-spool type in which the first, or LP compressor creates an air mass flow greater than that required by the second HP compressor. The excess airflow passes around the HP compressor and mixes with the hot turbine exhaust gas, to be accelerated in the propelling nozzle. The result is that the jet velocity is more closely matched to the aircraft speed requirements, giving much greater propulsive efficiency, and lower fuel consumption, than the single-spool pure jet. The latter type of engine is rarely found in modern aircraft. An example of a twin-spool compressor is shown at Figure 20-22.

FIGURE 20-22
Twin-Spool Axial Compressor



62. The high by-pass turbo-fan engine takes this concept a stage further. The single or multi-stage fan at the front of the engine delivers only about one-fifth of its output to the core of the engine, known as the gas generator. The remainder of the fan output is ducted to a 'cold' propelling nozzle to produce most of the engine thrust. The 'core' airflow then passes, typically, through two further compressors before entering the combustion chambers, turbines and exhaust system. In a triple-spool engine the rearmost turbine drives the fan. An example of a triple-spool compressor is shown at Figure 20-23.

FIGURE 20-23
Triple-Spool Compressor



63. Twin-spool and triple-spool compressors offer the following advantages over single spool systems.

(a) Because the spools are not physically connected each is able to operate at the RPM which best match the prevailing air mass flow and pressure ratios thereby increasing overall efficiency and reducing the danger of compressor stall and surge.

(b) Because the HP spool is required to handle only a small proportion of the total air flow, its diameter and mass can be greatly reduced. The resulting low inertia enables it to react very quickly to throttle increases rapidly accelerating to the required higher RPM, mass flow, and pressure ratio. This greatly reduces the probability of compressor stall and surge during rapid throttle increases.

(c) Because only a small proportion of the air mass flow passes through the HP spool, and the remainder passes at a much lower velocity around the outside, friction losses are greatly reduced thereby improving the thermal and mechanical efficiency of the engine.

(d) Because the velocity of the by-pass air is more closely matched to that of the aircraft, propulsive efficiency is improved at low to moderate airspeeds.

Diffusers

64. On exit from the compressor the air passes through a diffuser casing, which forms a divergent passage. This casing prepares the air for entry into the combustion chamber at a low velocity, by converting some of its kinetic energy into pressure energy further increasing static pressure.

Gas Turbine Construction Part 2 – The Hot Section

Thrust Augmentation Systems

Air Bleeds and Auxillary Drives

Gas Turbine Construction Part 2 – The Hot Section

Introduction

1. This section deals with those major engine components that are located in what is known as the hot section. These are the combustion chamber or chambers, the turbine, and the jet pipe, which comprises the propelling nozzle and the exhaust pipe between turbine outlet and propelling nozzle.

Combustion Chambers

2. The combustion chamber (or burner section) is a region where fuel is burned to increase the temperature of the gas (air), adding heat energy.

3. The purposes of the combustion chamber are as follows:

(a) To provide a suitable environment in terms of pressure, velocity and airflow pattern for efficient sustained combustion.

(b) To mix fuel and air in the required proportions for efficient combustion.

(c) To mix the hot gasses resulting from combustion with the remainder of the airflow to produce a uniform temperature gas flow.

(d) To deliver the hot gasses to the turbine nozzle guide vanes at the optimum velocity, pressure and temperature with the minimum losses.

4. The following conditions are required for sustained efficient combustion:

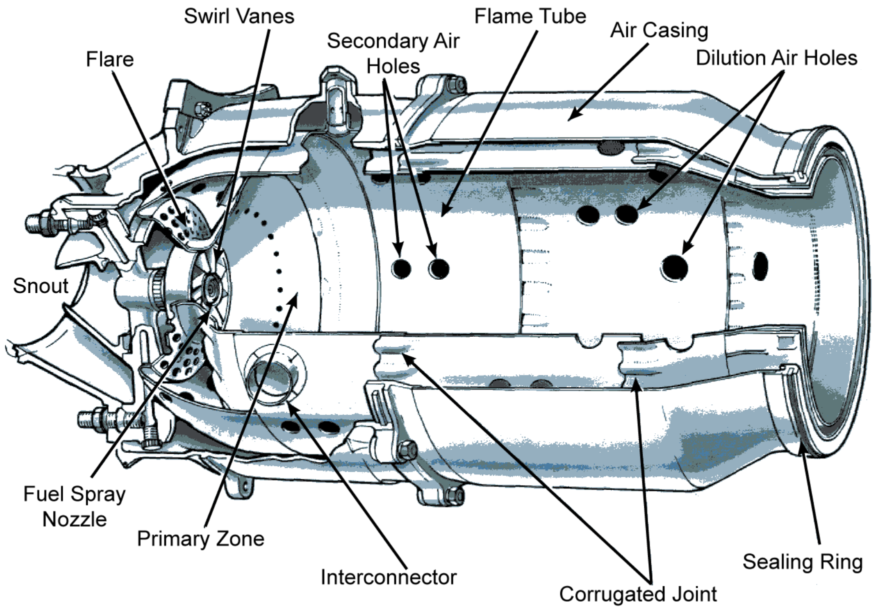
(a) An air:fuel ratio within the range 8:1 to 30:1.

(b) Thorough mixing of the fuel and air by atomisation and/or vaporisation.

(c) An airflow velocity not exceeding that of the flame front which is typically between 60 and 80 feet per second.

5. Figure 21-1 shows an early type of combustion chamber. Airflow from the compressor outlet enters through the snout of the chamber from where the flow is divided. Some is directed through swirl vanes to the combustion zone within the flame tube, or combustion liner, and the remainder flows along an annular passage between the outer casing of the chamber and the flame tube. This annular airflow assists with heat insulation, whilst progressively mixing with the combustion gases through holes in the wall of the flame tube. Interconnector pipes from the flame tube serve to connect adjacent combustion chambers and maintain similar conditions in each. The corrugated joints in the flame tube permit expansion.

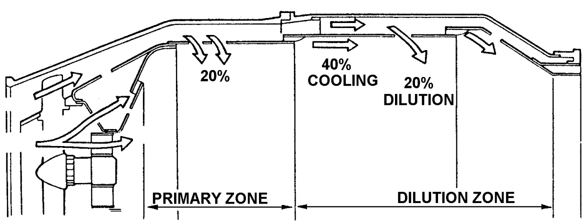
FIGURE 21-1
Early Combustion Chamber (Rolls Royce)



6. The total airflow through the hot section of the engine, as a ratio to fuel flow, may vary from 45:1 to as much as 130:1. This is obviously far too high for the optimum combustion ratio of 15:1. The combustion chamber is therefore designed to introduce a small proportion of the airflow (up to about 20%) into the primary or combustion zone of the chamber. Here it will mix with the fuel at a ratio of about 15:1. This proportion of the airflow is called the primary air. Downstream of this area of the combustion chamber is the dilution zone, where the remaining 80% (called the secondary air) mixes with the combustion products.

7. Since the temperature of the combustion gases is in the region of 1800°C to 2000°C, the secondary air serves to cool the gases to a level which the nozzle guide vanes of the turbine can withstand. The temperature of the gas is raised by the combustion gases to between 1000°C and 1500°C or even higher, the limit of acceptable temperature being determined by the materials from which the turbine guide vanes and first stage blades are made. Because of steady diffusion through the chamber, and virtually unimpeded exit, the gas velocity and pressure both fall slightly across the combustion chamber. The airflow and flame stabilisation in a typical combustion chamber is shown at Figure 21-2.

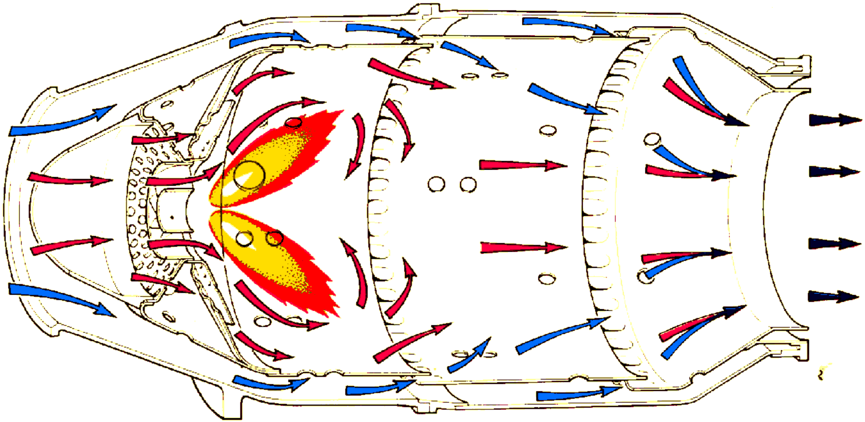
FIGURE 21-2
Combustion Chamber Airflow Split



8. Kerosene burns at a relatively slow rate of 60 to 80 feet per second, so the velocity of the air entering the combustion zone of the combustion chamber must be kept low, otherwise the burning fuel will simply be blown away. The diffuser section reduces the air velocity from the compressor to a lower value and further diffusion in the entry section of the combustion chamber reduces it still further, to about 60 feet per second. This is still too high for stable burning and so the airflow is given a swirling motion to further reduce its axial velocity.

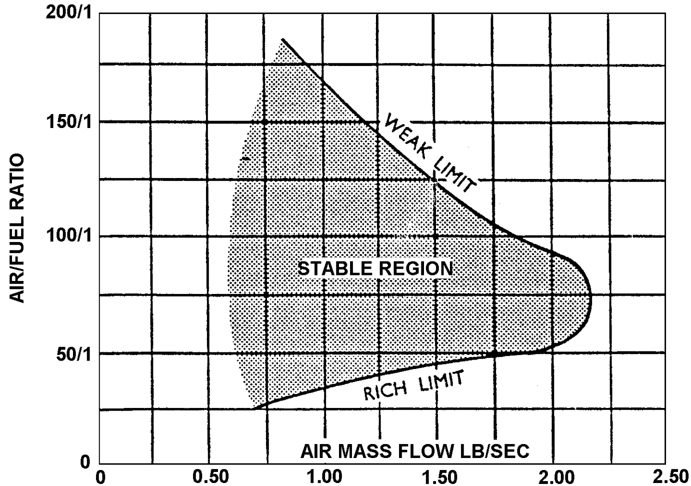
9. Immediately downstream of the snout or entry section of the combustion chamber, primary air passes through swirl vanes, which impart a rotational airflow. This combines with air entering through the primary air holes to produce a low velocity recirculatory airflow called a torroidal vortex (something like a smoke ring) which has the effect of stabilising and anchoring the flame, see Figure 21-3.

FIGURE 21-3
Flame Stabilisation



10. It is important that the combustion process remains stable, to ensure efficient burning of the fuel/air mixture, and to further ensure that the flame remains alight over a wide operating range. For any particular type of combustion chamber there are rich and a weak limits to the fuel/air ratio, beyond which combustion cannot be sustained and the flame will be extinguished. Flame-out is perhaps most likely to occur during a high speed descent with the engines at idle power, since in this situation there is a high mass airflow and a low fuel flow, resulting in a very weak mixture. A graph illustrating the limits of combustion stability is shown at Figure 21-4.

FIGURE 21-4
Combustion Stability



11. There are weak and rich limits beyond which ignition (as opposed to combustion) of the fuel will not be achieved, since it is always more difficult to establish combustion than it is to maintain it. These limits are within the stable region shown in the graph at Figure 21-4.

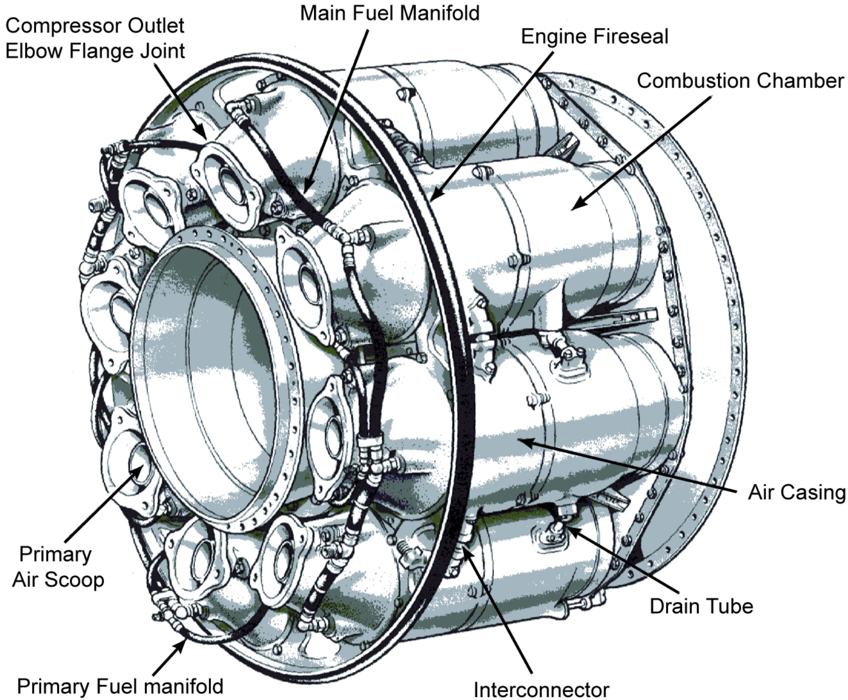
Types of Combustion Chamber

12. Combustion chamber design varies considerably according to the power plant requirements of the aircraft. The main types of combustion chamber layout are described below.

Multiple or Can-Type Combustion Chambers. The early layout, typical of centrifugal and early axial flow compressor fitted engines. Individual chambers, like that in Figure 21-1 and Figure 21-2, are disposed around the engine and interconnected only for purposes of pressure equalisation and to assist flame spread during initial start. Each chamber comprises an inner flame tube around which there is a cooling air casing. Cooling air comprises some 80% of the total airflow supplied to the entry section of the chamber (see Figure 21-2).

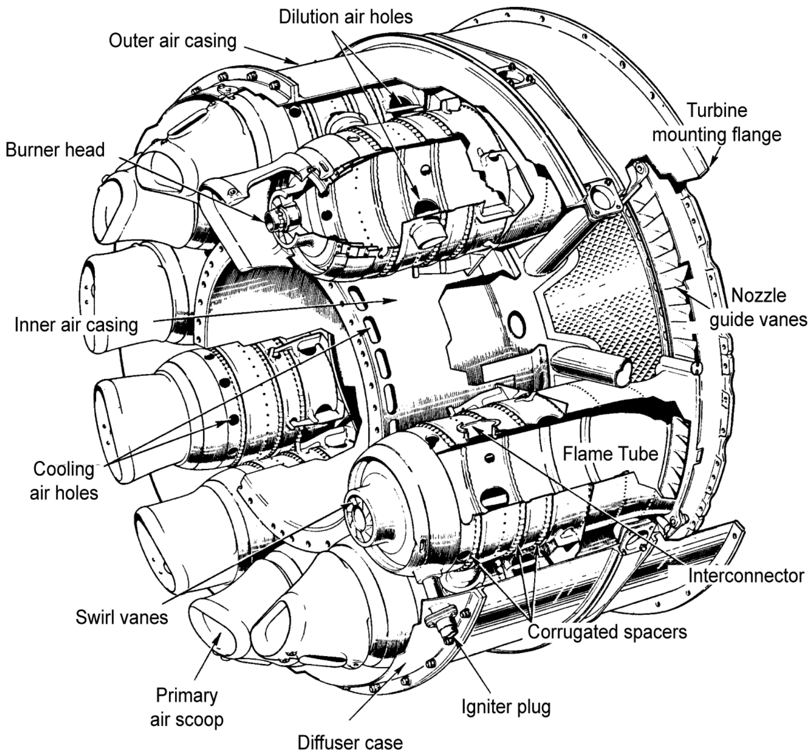
13. The principal advantage of this type of combustion chamber is its high resistance to distortion when heated. The main disadvantages lie in the amount of material needed for their construction and their uneconomical use of the available space. The layout of can-type, or multiple combustion chambers around the engine is illustrated at Figure 21-5.

FIGURE 21-5
Multiple Combustion Chambers



Tubo-Annular, Cannular or Can-Annular are all names given to the second type of combustion chamber. The layout is similar to that of the multiple combustion chamber, but the individual flame tubes are mounted within a common annular air casing. This simplifies construction, easing maintenance and inspection procedures. The diameter of the casing can be significantly reduced for the same mass airflow and there is a considerable saving in materials and weight, compared to the multiple combustion chamber arrangement. A typical layout is illustrated at Figure 21-6.

FIGURE 21-6
Cannular Combustion Chamber Layout

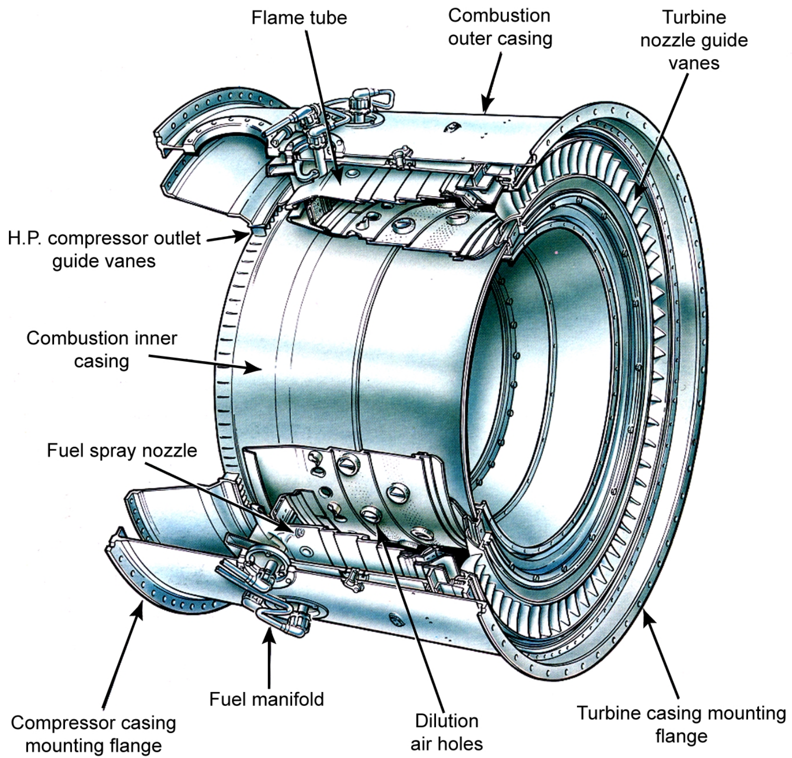


The Annular type of combustion chamber is the type found on most modern turbo-fan engines. It comprises a single combustion chamber annularly disposed around the engine (forming an 'annulus', or sleeve, encompassing the engine). It is open at the front to admit air from the compressor discharge diffusers, and at the rear to deliver hot, expanding gas to the turbine inlet nozzle guide vanes.

14. Fuel spray nozzles are disposed around the annulus near the front end. Since the whole of the area encompassing the engine is used for combustion, the combustion chamber can be shorter for the same overall diameter as the previous two layouts. Construction is both lighter and simpler. Flame propagation is more efficient, so there is less likelihood of local 'hot spots' or of flameout. Total surface area is significantly reduced with the consequence that less cooling air is required, giving greater combustion efficiency.

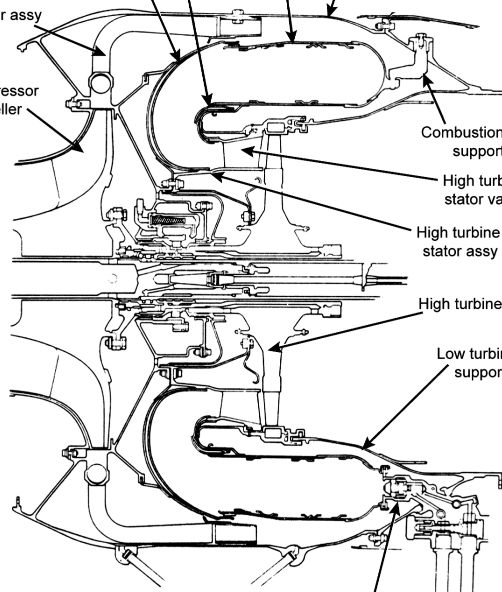
15. The annular combustion chamber virtually eliminates unburned fuel, reducing air pollution and increasing thermal efficiency. A typical example is illustrated at Figure 21-7.

FIGURE 21-7
Annular Combustion Layout



16. In order to reduce overall engine length the annular combustion chamber is often used in a modified reverse-flow form as indicated in Figure 21-8. The use of such chambers is particularly prevalent in small turbo-shaft helicopter engines.

FIGURE 21-8
Reverse Flow Combustion Chamber



17. The high temperatures in the combustion chamber, and the great temperature differences between it and the surrounding parts of the engine, places significant thermal loads upon the chamber itself and expansion loads upon the adjacent structure. The joint between combustion chamber and turbine inlet is often designed to allow for expansion and combustion chamber cooling is very carefully designed to minimise thermal loads.

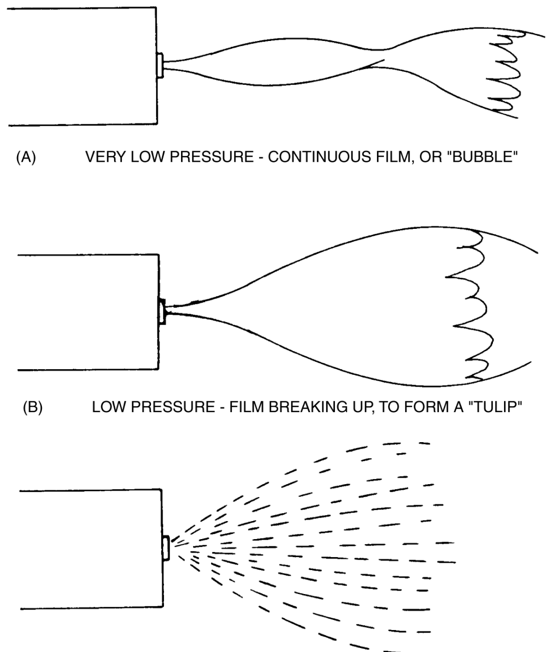
18. The combustion efficiency of most gas turbine engines is 100% at sea level, reducing to 98% at altitude because of reducing static air pressure and reducing combustion chamber inlet temperature.

Fuel Burners

Burners (Atomisers)

19. The purpose of the burners, or fuel injectors, is to supply a continuous spray of finely atomised fuel that will readily mix with air to form an easily combustible mixture. In order to achieve this the burner nozzle orifice, through which the fuel is sprayed, must be supplied with fuel at sufficient pressure and must be designed to produce a cone-shaped spray pattern. If the fuel supply pressure is too low the fuel will be discharged as a film, rather than a spray, as shown at Figure 21-9. If supply pressure is too high the nozzle is liable to produce a solid jet of fuel.

FIGURE 21-9
Stages of Fuel Atomisation



(A) VERY LOW PRESSURE - CONTINUOUS FILM, OR "BUBBLE"

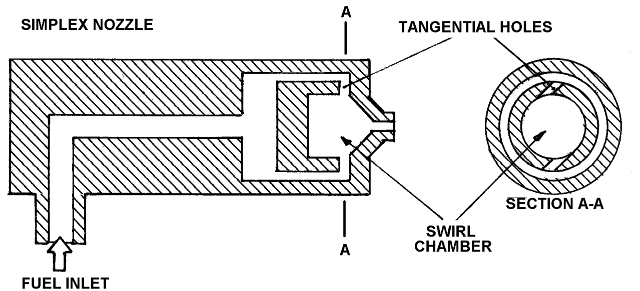
(B) LOW PRESSURE - FILM BREAKING UP, TO FORM A "TULIP"

(C) SUFFICIENT PRESSURE - FUEL EJECTED AS A FINELY ATOMISED SPRAY

20. There are various types of burner, the principal ones of which are described below.

Simplex Burners. The early gas turbine engines used a simple spray nozzle with a single discharge orifice as illustrated at Figure 21-10. Fuel is supplied through tangential holes to a chamber, before being discharged. The swirling motion imparted to the fuel assists with atomisation and varying its supply pressure varies the quantity of fuel discharged

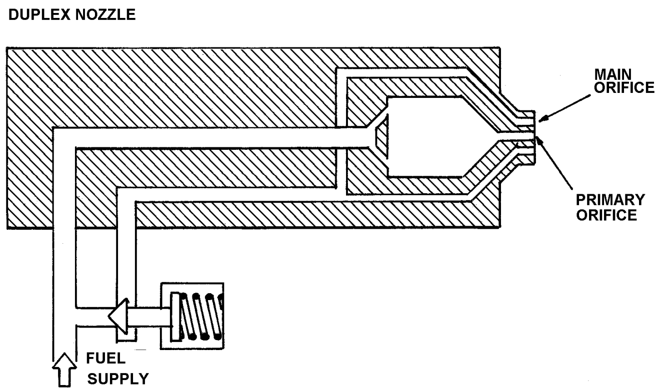
FIGURE 21-10
Simplex Burner



21. Because the flow of a liquid through an orifice is proportional to the square root of the pressure drop across it, the range of pressures needed to create the range of fuel flows required ranged from about 30 psi for minimum engine rpm to about 3000 psi at maximum rpm. Fuel pumps of the time could not produce such high pressures, and maintenance of the required spray pattern proved impossible over such a wide range of pressures. Consequently, alternative spray nozzles were developed in which the fuel flow range could be achieved with a narrower fuel supply pressure range.

Duplex (Duple) Burners. The Duplex (or Duple) burner has two nozzle orifices, as illustrated at Figure 21-11. At low supply pressure (small throttle openings), fuel flows only to the primary orifice. As the throttle valve is progressively opened, and fuel supply pressure to the burner increases, the spring-loaded pressurising valve progressively opens to admit fuel to the main nozzle orifices. Hence, for a relatively narrow supply pressure range, fuel flow can be varied over a wide range and pressure remains adequate for efficient atomisation

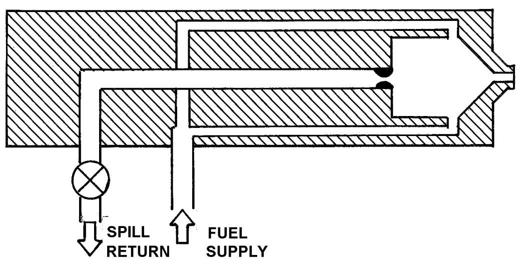
FIGURE 21-11
Duplex Burner



Spill Burners

22. A spill burner nozzle is illustrated at Figure 21-12. It is similar to the simplex nozzle, with the addition of a passage from the swirl chamber through which fuel can be ‘spilled’ back to the fuel pump supply at a controlled rate. Fuel is supplied to the swirl chamber at high pressure, ensuring a good spray pattern. At high demand rates (maximum engine rpm) the spill valve is closed and all the supplied fuel is sprayed from the nozzle. At low demand rates (minimum engine rpm at high altitude) the spill valve is opened so that most of the fuel supplied to the swirl chamber is spilled away. However, at the high supply pressure there is sufficient swirling motion to ensure that the small quantity discharged through the nozzle orifice is efficiently atomised

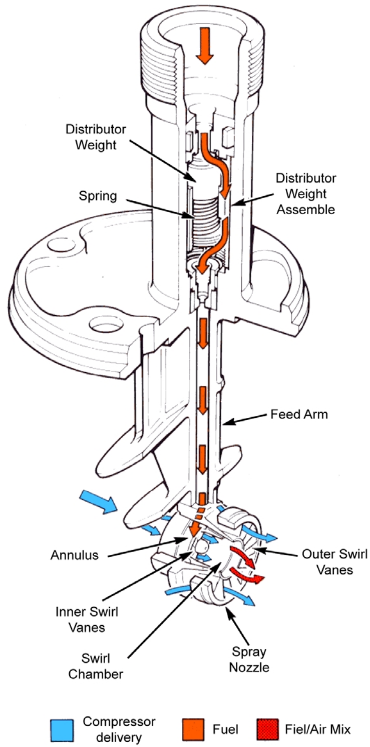
FIGURE 21-12
Spill Burner



23. The spill burner requires a complex fuel system to control both supply and spill pressures to meet the fuel flow requirements over the full operating range of the engine.

Air Spray Nozzles. Many modern gas turbine engines use air spray nozzles. Some of the primary air for combustion mixes with the fuel supply in the spray nozzle. This aeration of the fuel spray achieves more efficient combustion of the fuel and requires lower fuel supply pressures than other burner types. An air spray nozzle is illustrated at Figure 21-13.

FIGURE 21-13
Airspray Nozzle



Turbines

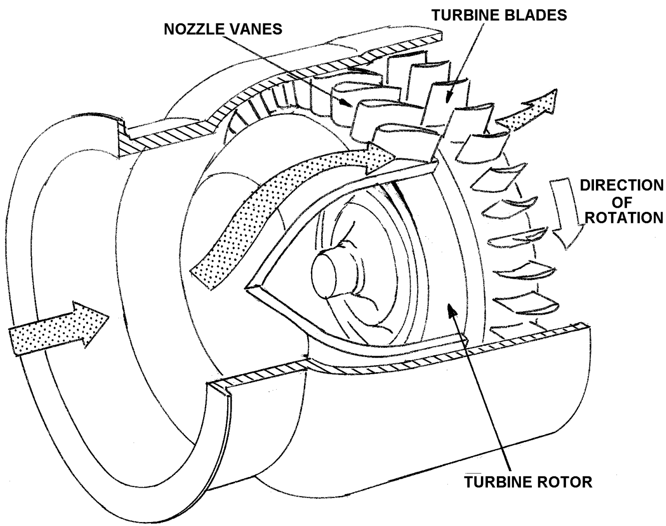
24. The function of the turbine is to transform some of the energy of the hot gases into shaft horsepower (SHP). The expanding gas does work and this process is, ideally, one of adiabatic expansion, in which there is no loss of energy to the surroundings. In turbojet engines the turbine extracts sufficient energy to drive the compressor and accessories (fuel pumps, generators, hydraulic pumps, etc), the remaining energy providing propulsive thrust. In a turbo-prop or turbo-fan engine, the turbine also has to drive the propeller or fan, and consequently much less energy is available to provide thrust in the jet pipe. In many turbo-props the thrust remaining is negligible. In turbo-shaft engines, virtually all of the useful energy is extracted by the turbines and output shaft.

25. In effect, a turbine is a compressor in reverse, since it serves to reduce the pressure and increase the velocity of the gas passing through it. Turbines used in aircraft gas turbine engines are of the axial flow type and the number of turbine stages (rotor/stator sets) varies according to engine power and type.

Single Spool Turbines

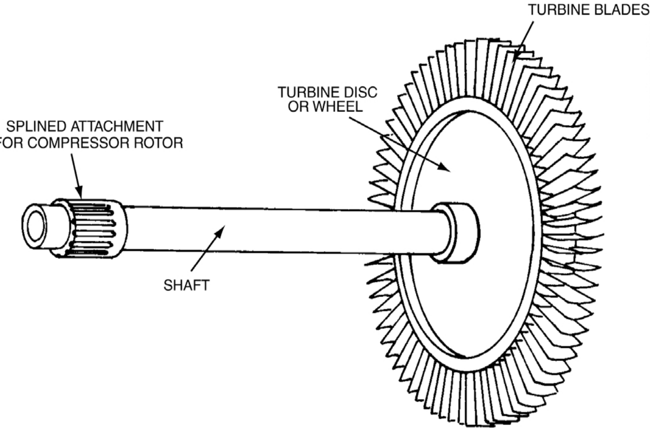
26. In its simplest form the turbine consists of a ring of blades mounted upon a wheel, or disc, attached to the rotor (rotating assembly). High velocity gas from the combustion process is directed onto the turbine blades and the force exerted (mass x acceleration) causes the turbine wheel to rotate. The velocity of the combustion gas is increased (accelerated) by passing it through a convergent passage, or nozzle, thus increasing the force it exerts upon the turbine blades. A simple turbine is illustrated is Figure 21-14.

FIGURE 21-14
Single-Spool Turbine



27. The terminology associated with the turbine rotating assembly is illustrated at Figure 21-15.

FIGURE 21-15
Single -Stage
Turbine Rotor

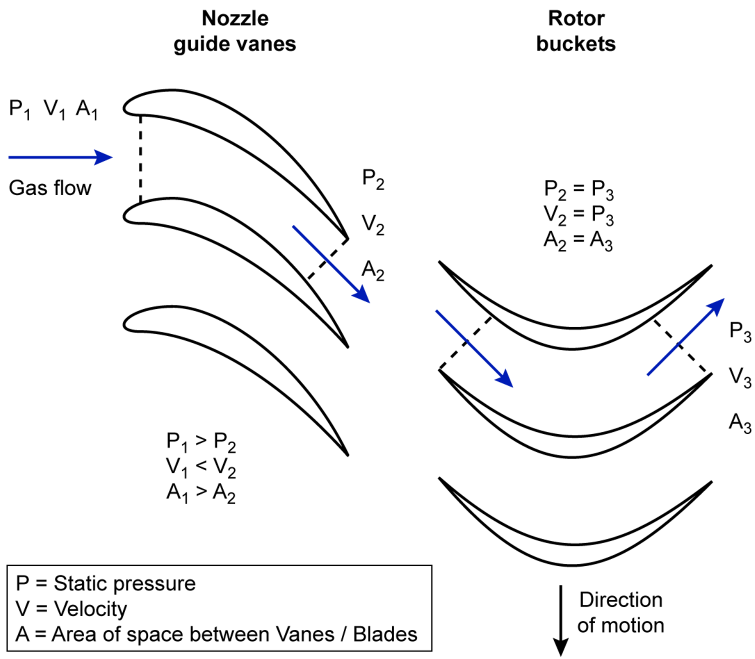


28. The energy transfer in the turbine is achieved by accelerating the combustion gases, by forcing them to pass through a convergent passage (a nozzle), to approximately sonic velocity. The kinetic energy added to the gas during the velocity increase is absorbed upon impact with the turbine blades, causing the turbine to rotate at high speed, providing the power needed to drive the compressor.

29. In a practical gas turbine the nozzling of the combustion gases is achieved by a ring of stationary, aerofoil shaped nozzle guide vanes attached to the engine casing between the combustion chamber outlet and the turbine inlet. The passage between each pair of adjacent vanes forms a convergent duct, and the vanes are angled so as to direct the accelerated gas onto the turbine blades at the appropriate angle.

30. Three forms of turbine are possible. These are impulse, reaction, and combined impulse/reaction. In the pure impulse type of turbine the nozzle guide vanes form convergent ducts through which the direction of the gas flow is accelerated and deflected in the direction of rotation of the engine. The rotor blades are of bucket shaped section such that the space between adjacent blades is constant. Gas passing between the blades is therefore subjected to a change of direction whilst its speed and static pressure remain constant. This change of direction equates to a change in momentum and it is the impulse of this change of momentum that drives the turbine. The impulse type turbine is illustrated in Figure 21-16.

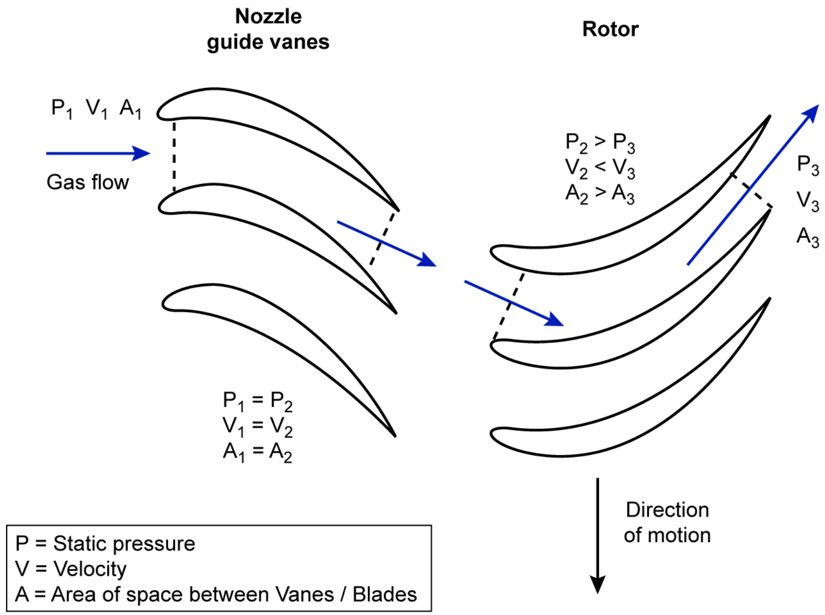
FIGURE 21-16
Impulse Turbine Blades



31. In the reaction type turbine this series of events is reversed. The cross sectional area of the spaces between the nozzle guide vanes is constant and so gas passing through them is subject to a change of direct whilst its speed and static pressure remain constant. The rotor blades are of aerofoil section such that the spaces between them forms convergent ducts. Gas passing through these ducts is accelerated causing its static pressure and temperature to reduce. It is the reaction to this acceleration that drives the turbine as illustrated at Figure 21-17.

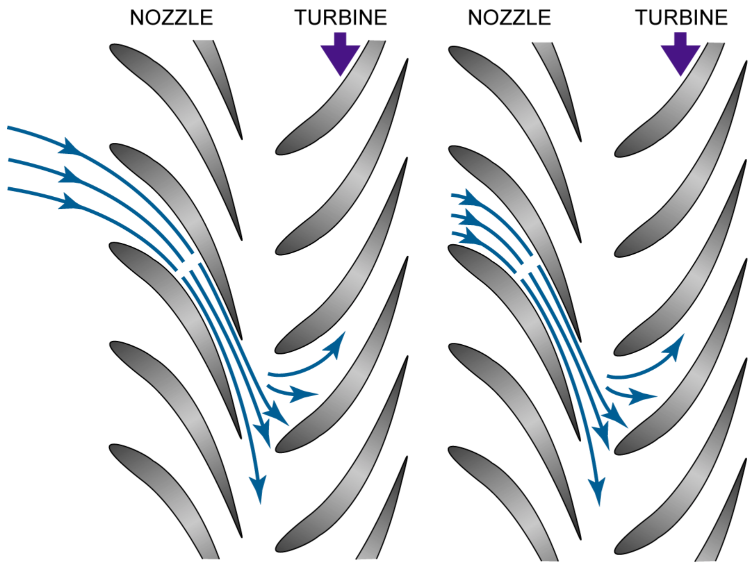
FIGURE 21-17

Reaction Turbine Blades



32. When employed in isolation both the impulse and reaction systems are inefficient and so most practical turbines employ a combination of the two. In the impulse/reaction turbine (Figure 21-18) the blade root sections are predominantly of impulse-type bucket section, gradually changing to reaction-type aerofoil section towards the tip. In the nozzle guide vanes this change of cross section is reversed such that the greatest acceleration occurs close to the roots of the vanes, whilst the greatest deflection in the direction of engine rotation occurs close to the tips. In the rotors the changing cross section of the blades results in mainly impulse-type energy extraction at the roots gradually changing into a mainly reaction-type process at the tips. Gas passing through the rotors is accelerated and deflected such that the rotational velocity imparted by the preceding nozzle guide vanes is negated, leaving a more-or-less axial flow out of the turbine. The general gas flow through an impulse/reaction turbine is illustrated in Figure 21-18, and the changing cross section of turbine blade is illustrated at Figure 21-23.

FIGURE 21-18
Impulse / Reaction Turbine

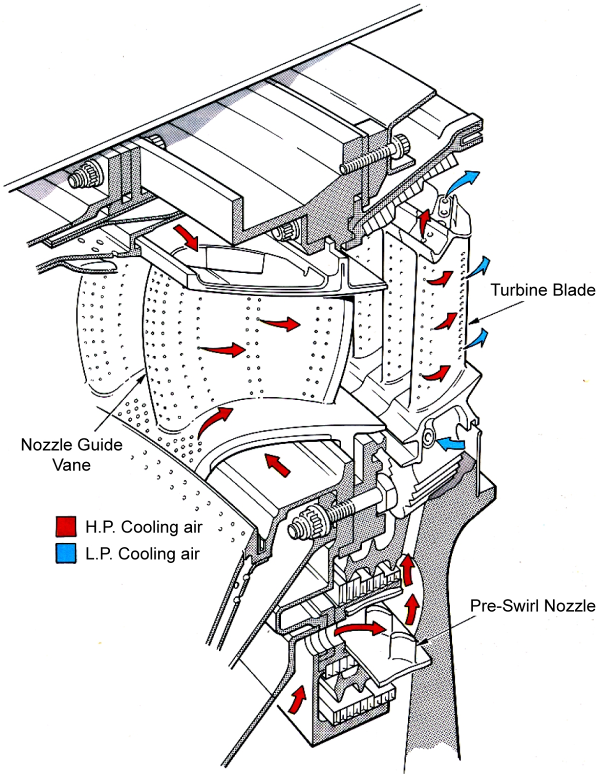


Turbine Construction

33. The function of the combustion process in a gas turbine engine is to expand the gas, forcing it into the nozzle guide vanes. Clearly, the greater the temperature achieved during combustion the greater the expansion and the greater the energy transfer possible in the turbine. However, this is limited by the ability of the nozzle guide vanes and turbine blades to withstand high temperatures.

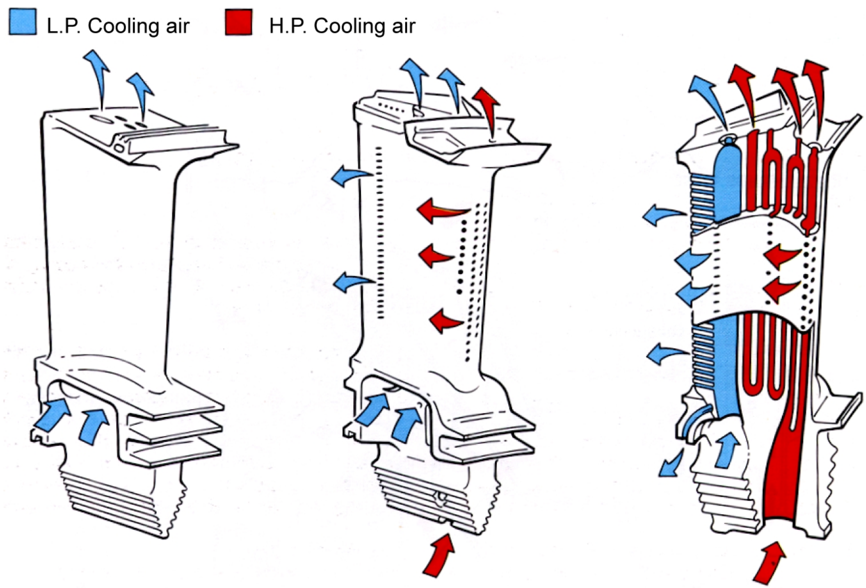
34. The nozzle guide vanes are usually hollow and cooling air from the compressors is passed through them to prevent overheating and damage due to thermal stresses and gas loads. An example of this method of cooling is shown at Figure 21-19. Nozzle guide vanes are usually made of nickel alloy.

FIGURE 21-19


Turbine Blade
Nozzle Guide Vane
H.P. Cooling air
L.P. Cooling air
Pre-Swirl Nozzle

35. Early attempts at turbine blade cooling relied upon heat transfer by convection, using blades with longitudinal passages through which low pressure cooling air is passed. The next stage of development was to introduce external air film cooling of both nozzle guide vanes and rotor blades. In this, high-pressure air is led through internal passages to small holes in the surface of the nozzle vanes and rotor blades. The escaping air forms a cooling film over the surface of the vanes and blades. This is normally supplemented by internal convection cooling of the blades through complex passages that force the cooling air to make several passes along the length of the blade. (See Figure 21-20).

FIGURE 21-20
High Pressure Turbine Blade Cooling



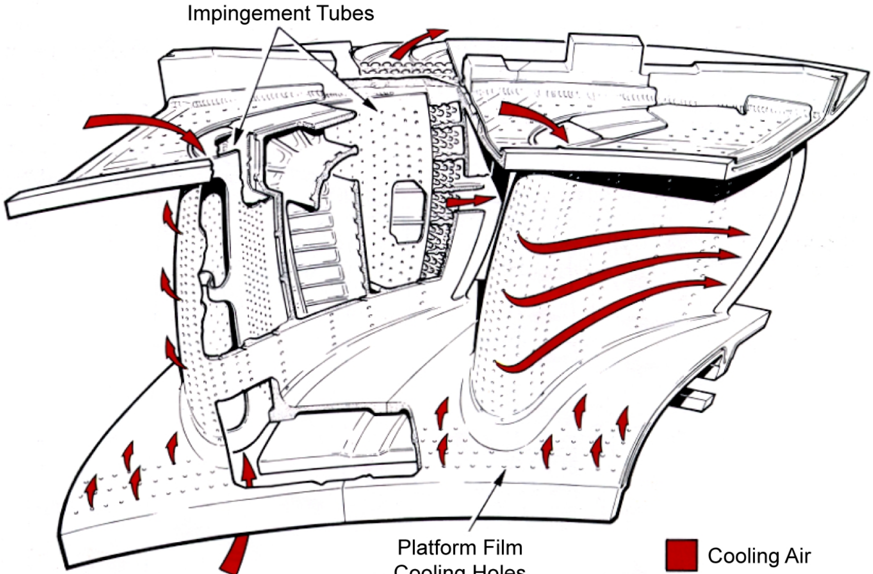
Single Pass Internal Cooling (1960's)

Single Pass, Multi-Feed Internal Cooling with Film Cooling (1970's)

Quintuple Pass, Multi-Feed Internal Cooling with Extensive Film Cooling

FIGURE 21-21

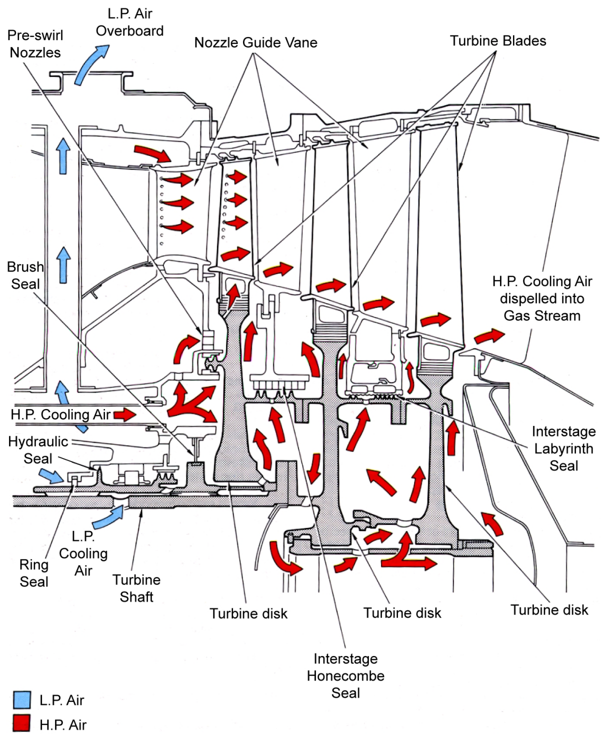
Impingement Cooling of Nozzle Guide Vanes



36. In some cases the nozzle guide vanes contain an internal tube fed with cooling air. Holes in the tube direct air onto the inner surfaces of the hollow nozzle guide vanes, the air finally escaping into the gas flow through holes in the trailing edge of the vane. This method is known as impingement, or jet, cooling. (See Figure 21-21). A few engines have been built with nozzle guide vanes made from a very expensive porous material and cooling is effected by the escape of cooling air through the pores from the hollow interior of the vane. This is known as transpiration cooling.

37. The turbine discs, or wheels upon which the blades are mounted are typically forged from steel and bolted to the rotor shaft. The discs are usually cooled by airflow directed across both sides of the disc. (See Figure 21-22).

FIGURE 21-22
Turbine Cooling and Sealing Arrangements



38. The turbine nozzle guide vanes and rotor blades are twisted so that the stagger angle increases from root to tip. The twist in the nozzle vanes is to ensure that the gas flow from the combustion chamber does equal work at all points along the length of the rotor blade. It achieves this by causing a gas velocity decrease, and pressure increase, from nozzle root to tip to counter the fact that turbine blade rotational velocity is higher at the tip than at the root. The twist in the turbine rotor blades restores the gas flow to uniform velocity and pressure from root to tip so that it enters the exhaust system without creating swirl. This is illustrated at Figure 21-23 and Figure 21-24.

FIGURE 21-23
Change of Stagger Angle from Root to Tip

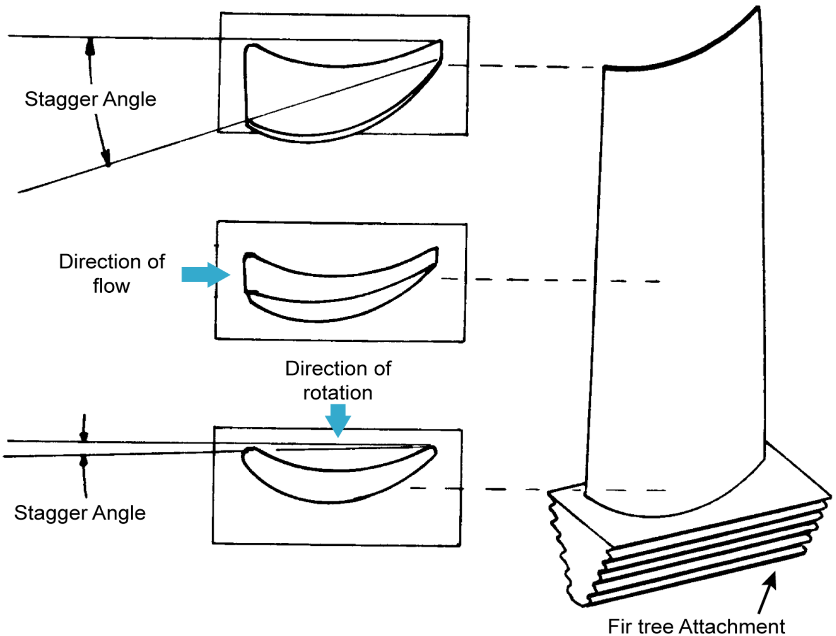
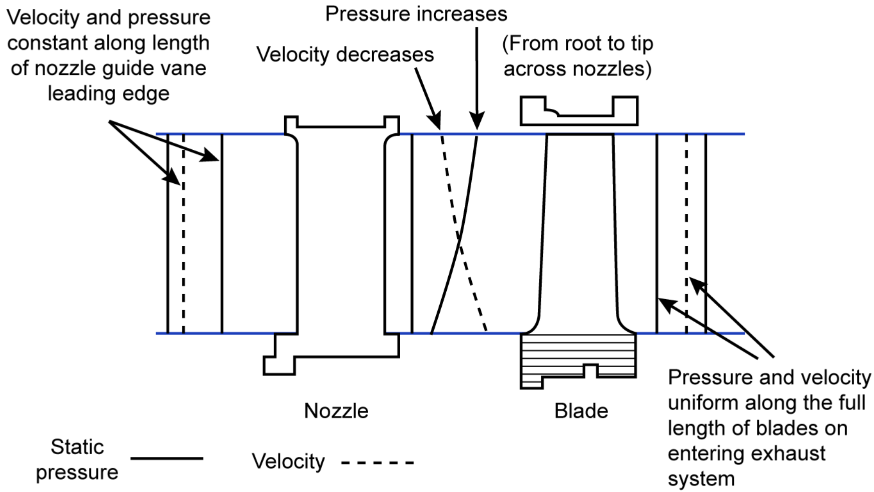


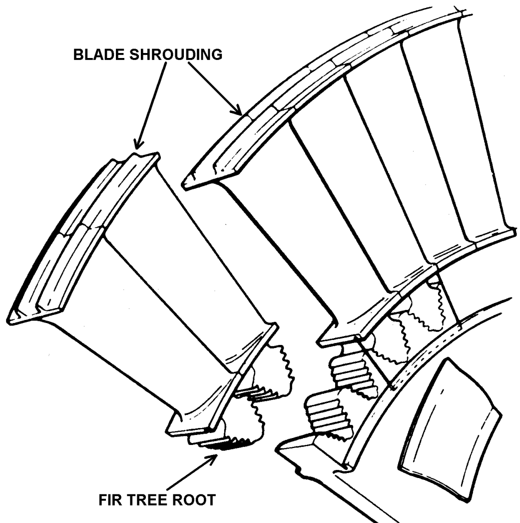
FIGURE 21-24
Incoming Velocity and Pressure Uniform from Root to Tip



39. The turbine blades are the most highly stressed components in the engine. The temperature of the gas flow is sufficient to make them glow red-hot and the centrifugal loads due to rotation often amount to several tons on each blade when the rotor is turning at maximum rpm. The method by which the blades are attached to the turbine disc is consequently of great importance. In most engines the "fir tree" root design is used and this is illustrated at Figure 21-25 . When the turbine is stationary the blade is a loose fit in the peripheral serrations machined in the wheel, so there are no "in-built" stresses. During rotation centrifugal loading stiffens the attachment.

40. Since the function of the blades is to absorb the energy of the gas flow as efficiently as possible it is important that there should be no lost energy due to gas leakage between the tips, or ends, of the blades and the turbine casing. To reduce such leakage to a minimum, shrouding is often fitted to the blade tips, as shown at Figure 21-25. The 'ridges' on the shrouds leave an extremely small clearance between shroud and casing.

FIGURE 21-25
Turbine Blade Shrouding



41. A more effective method of maintaining constant minimum tip clearance over the whole operating range of the engine is known as Active Clearance Control (ACC). Compressor air is used to cool, and actively control the expansion, of the turbine casing to match the radial expansion of the blades. Blade tip shrouding is then unnecessary. A typical un-shrouded turbine blade is shown at Figure 21-23.

42. In addition to the ability to withstand the high centrifugal loads imposed upon them, turbine blades must also be resistant to fatigue, thermal shock and corrosion. During operation the effects of heat and centrifugal loading cause the blades to grow in length, a process known as 'blade creep', which has a finite limit after which 'blade rub' will occur as the blade tips contact the turbine casing. Hence, turbine blades have a finite life, which is determined by the material from which the blades are made. Most turbine blades are cast from nickel-based alloys.

43. Microscopic examination of the blade material reveals that it is made up of many crystals which are normally randomly aligned. By ensuring that the crystals are uniformly aligned the material becomes much more resistant to creep, thermal shock and fatigue, significantly lengthening in-service life. This manufacturing technique is known as 'directional solidification'.

44. A more recent advance is to produce each blade from a single crystal of nickel alloy, which permits a substantial increase in turbine gas temperature. Some modern turbines use blades made from reinforced ceramics, instead of metal, which are capable of withstanding much greater speeds and temperatures.

Multi-Spool Engines

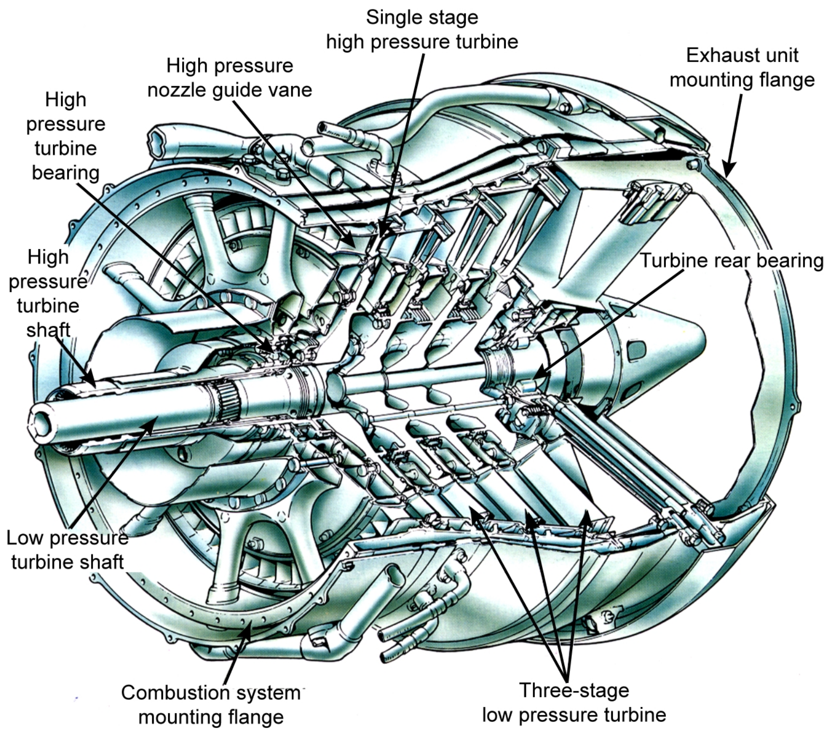
45. In a single rotor (single spool) engine, power is developed by one turbine rotor, which drives the compressor and the engine accessories. The turbine rotor may have one or more stages, or turbine wheels, depending upon the power requirements for the engine.

46. In a multi-spool engine power is developed by two or three independent turbine rotors connected by co-axial, mechanically independent shafts to their respective compressors. Again each turbine rotor may have one or more stages.

47. The first turbine in the sequence (the one into which the combustion chamber outlet is first directed) is called the HP turbine and usually drives the HP compressor and the accessory gearbox. In a twin-spool unit, exhaust gas from the HP turbine is led to the second (LP) turbine. The LP turbine drives the LP compressor. In twin-spool turbo-props, the LP turbine also drives the propeller.

48. A triple-spool engine has a third turbine situated between the HP and LP, known as the intermediate (IP) turbine, which drives the IP compressor. In some turbo-prop and turbo-shaft layouts a free turbine rotor, independent of the HP or LP compressors is used to drive the propeller or shaft. This is called a free/power turbine engine. A twin-spool layout is shown at Figure 21-26.

FIGURE 21-26
Twin-Spool Turbine



49. The LP section runs more slowly than the HP section. Due to the variations in rpm between the two turbine assemblies, the practice is to control and limit the rpm of the HP section through the fuel control unit (FCU). It is also to the HP rotor that the starter is connected. Multi-spool engines are capable of high compression ratios and are much less susceptible to compressor surge and stall than single-spool engines, especially during acceleration. Their improved power-to-weight ratio makes for much-improved specific fuel consumption (sfc). With turbo-prop and turbo-fan engines starting is easier because the HP spool does not drive the power unit (propeller or fan).

Effects of Damage

50. Compressor damage usually occurs to the blades as a result of ingestion of foreign objects. Turbine damage, however, is almost invariably the result of the high gas temperatures, and the stresses arising from high rotational velocity. One of the effects is permanent extension of the turbine blades, or creep. This is only acceptable within certain limits, after which blade rub occurs as already discussed. Another effect is for the turbine blades to untwist due to the forces of the high velocity gas flow acting on the heated blades. The result is reduction of stagger angle, which reduces the efficiency of the blades.

51. Because of the stresses to which turbine blades are subjected, no cracks are acceptable in any part of them and small nicks or indentations are only permitted provided they are well away from the root area of the blade. Burning or distortion is unacceptable in any part of the blades.

52. Engine manufacturers place very stringent limits on the maximum acceptable turbine inlet gas temperature, because of the marked reduction in tensile strength of blade material with increased temperature. The ultimate effect of overheating turbine blades is to cause them to fracture at the shank, where the root joins the blade, due to centrifugal loading.

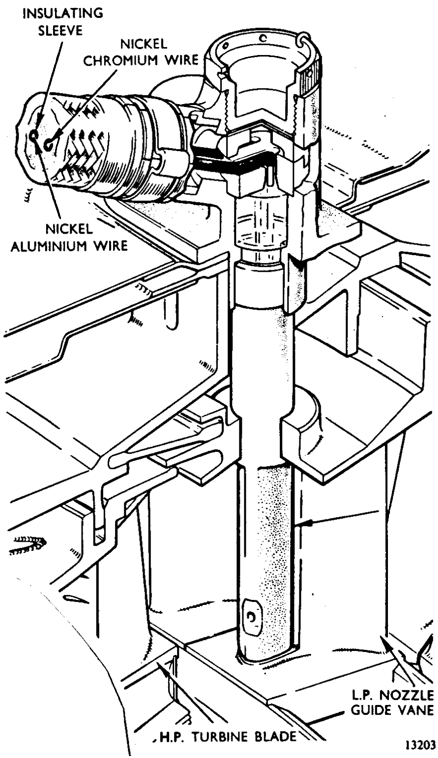
53. Centrifugal force ensures that the fractured blades leave the wheel radially at high velocity, and possibly with sufficient force to penetrate the turbine casing. This is known as an un-contained turbine failure and the departing blades can cause serious damage to adjacent parts of the aircraft structure and systems. Even if the fractured blades are contained within the turbine casing, the damage they cause to the remaining blades is usually sufficient to ruin the engine.

Gas Temperature Monitoring

54. It will be appreciated from the foregoing that monitoring of the turbine gas temperature (TGT) is of prime importance. It is not usually practicable to monitor the turbine inlet temperature (TIT), but since the temperature drop across the turbine stages is calculable it is quite sufficient to measure the temperature of the gases leaving the turbine, the exhaust gas temperature (EGT), and calibrate the display instrument accordingly.

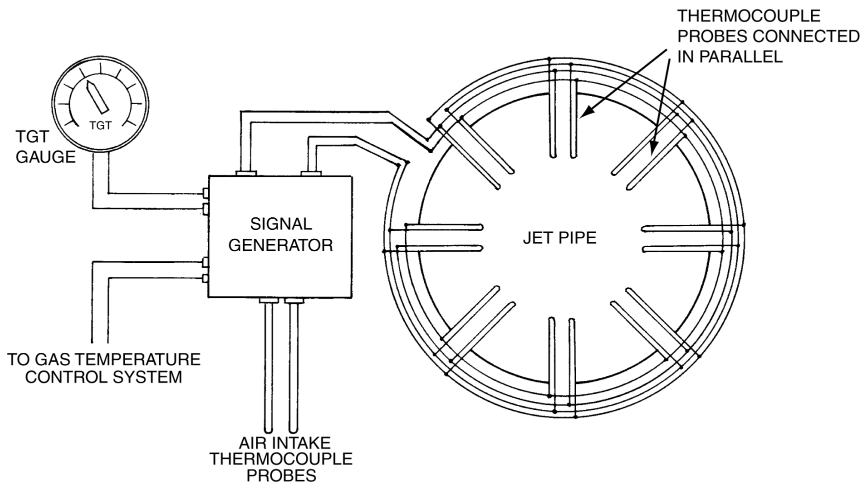
55. Gas temperature measurement is made using thermocouple probes usually located at the exit from the HP turbine. In the case of a multi-spool engine they are often incorporated with the nozzle guide vanes of the successive turbine, as shown at Figure 21-27.

FIGURE 21-27
Thermocouple
Arrangement



56. In the case of single spool engines, the thermocouples are located at the forward end of the jet pipe. In either case a number of thermocouples are dispersed radially and connected in parallel. This has the double advantage of a temperature averaging system, and system redundancy in the event of failure of individual thermocouple probes. The principle is illustrated at Figure 21-28.

FIGURE 21-28
Jet Pipe Multiple Thermocouple System



57. EGT monitoring is of particular importance during engine power changes. In order to increase or decrease power the fuel flow to the engine must be changed, but because of the inertia of the rotating assembly (the HP turbine and its compressor) the engine RPM does not immediately change.

58. During a power increase the fuel flow is increased, which clearly increases the combustion temperature. The engine RPM follows relatively slowly, so there is a period where the airflow from the compressor does not match the increased combustion temperature. This results in a marked temporary rise in EGT. Eventually, as the compressor 'spools up' the increasing airflow reduces the EGT. Care must always be taken when advancing the throttle to avoid excessive peak EGT.

59. Similarly, when reducing power closing the throttle reduces the fuel flow, and combustion temperature, but initially the airflow does not decrease since it takes time for the compressor to "spool down". This results in a marked temporary drop in EGT until the RPM has fallen to match the new throttle setting. Again, care must be exercised in throttle operation to avoid over-rapid cooling of the turbine blades and vanes.

Exhaust and Jet Nozzles

60. The exhaust system of a typical jet engine comprises two parts. These are the exhaust unit and the propelling nozzle. In cases where these parts are a significant distance apart, they are joined by a jet pipe. The functions of the exhaust unit are as follows:

(a) To support the turbine rear bearing.

(b) To provide a streamlined fairing for the rear face of the rear turbine disc.

(c) To remove any residual swirling motion from the gas emerging from the turbine.

(d) To diffuse the exhaust gas, reducing its velocity to reduce friction losses within the jetpipe.

61. The functions of the jet pipe are as follows:

(a) To transfer the exhaust gas from the exhaust unit to the propelling nozzle with minimum energy losses.

(b) To house the reheat system if fitted.

(c) To house temperature and pressure sensors.

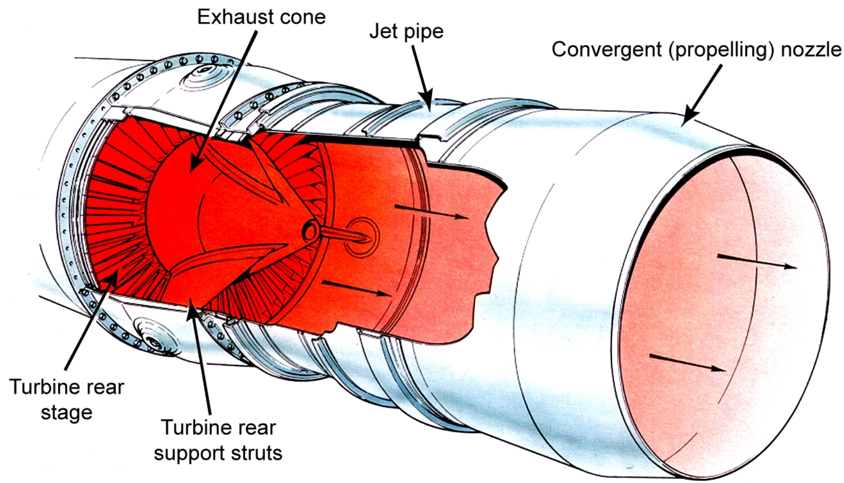
62. The functions of the propelling nozzle are as follows:

(a) To increase the velocity of the exhaust gas converting to kinetic energy as much of the residual pressure energy in the gas as is practicable.

(b) In the case of engines employing reheat systems the cross sectional area of the propelling nozzle must be automatically varied in order to match prevailing mass flow to maximise gas flow acceleration without exceeding the pressure ratio capabilities of the engine.

(c) To house (and comprise part of) reverse thrust systems where appropriate.

FIGURE 21-29
Basic Exhaust Nozzle



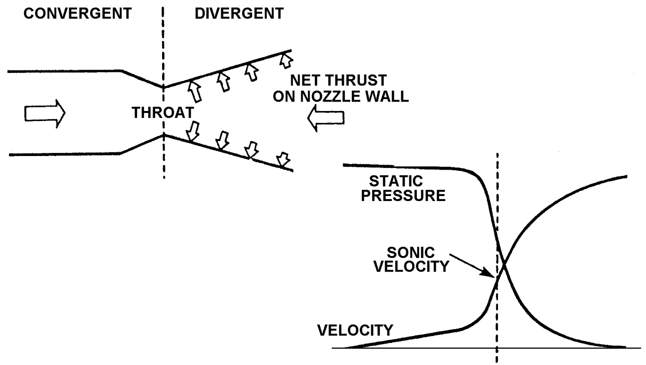
63. As well as the EGT thermocouple probes, a number of pressure-sensing probes are positioned in the exhaust casing to measure the turbine discharge pressure. This pressure may be displayed directly to the pilot, or alternatively compared with the air intake pressure and displayed to the pilot as engine pressure ratio (EPR).

64. Gas from the turbine enters the exhaust casing at velocities of between 750 and 1200 feet per second. Because velocities of this magnitude would produce high friction losses, the speed of the gas flow is reduced by diffusion. The increasing passage area between the exhaust cone and the wall of the jet pipe achieves this. The exhaust cone serves the additional function of preventing the exhaust gases from flowing across the rear face of the turbine wheel. It is normal to limit the speed of the gas flow at outlet from the exhaust case to a velocity of about 950 feet per second (Mach 0.5 at exhaust gas temperatures). It should be noted that this refers to the speed of the gas relative to the inside of the exhaust unit casing.

65. The propelling nozzle is a convergent duct, which accelerates the exhaust gasses before they pass out of the engine to atmosphere. The reaction to this final acceleration makes a major contribution to the overall thrust of the engine. In order to maximise engine thrust it is necessary that the velocity of the gas leaving the propelling nozzle is as high as possible. Provided the gas velocity is lower than the local speed of sound the rate of acceleration through the nozzle is proportional to the pressure drop across it so it. That is to say the acceleration is proportional to the difference between jet pipe pressure and ambient pressure. If jet pipe pressure is increased the velocity at the throat of the nozzle will increase until it equals the local speed of sound. At this point pressure waves can no longer flow upstream and so further increases in jet pipe pressure will not result in further acceleration. Instead the exhaust gas will be discharged to atmosphere at a pressure above ambient. The gas will then expand wastefully reducing the overall efficiency of the engine. Under these conditions the propelling nozzle is said to be choked.

66. In order to increase the gas velocity further it is necessary to increase the local speed of sound. This is achieved by means of a convergent–divergent nozzle which comprises a convergent inlet duct attached to a divergent outlet duct. As jet pipe pressure is increased beyond the choked nozzle condition in an engine employing a convergent-divergent propelling nozzle, the increasing temperature and pressure downstream of the throat causes an increase in the local speed of sound. This restores the relationship between pressure drop and acceleration, causing the gas to continue to accelerate through the divergent section of the nozzle. That is to say the convergent section of the nozzle remains choked with gas velocity equal to the local speed of sound at the throat (the narrowest part of the nozzle). Downstream of the throat the gas velocity continues to increase as the local speed of sound increases in that area. The overall effect is an increase in gas velocity and hence thrust. Figure 21-30 illustrates the pressure and velocity changes of gas flowing through a choked convergent-divergent propelling nozzle.

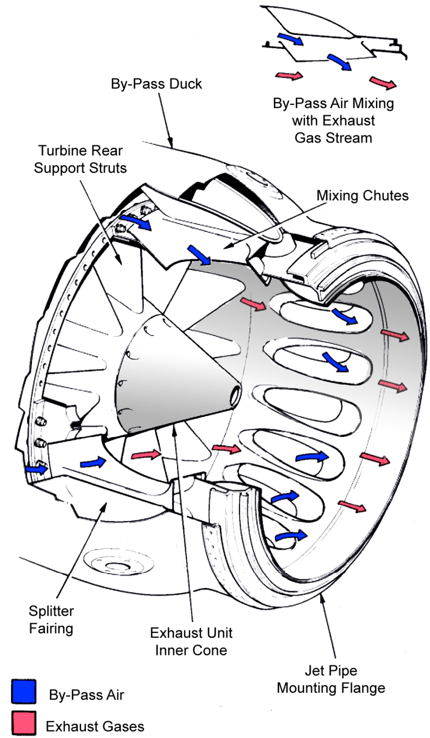
FIGURE 21-30
Gas Flow through a Convergent - Divergent Nozzle



67. The propulsive nozzle is designed to expand the hot combustion gases from the turbine discharge to atmosphere in a manner that will produce the maximum amount of thrust. The nozzle is an orifice, the size of which determines the pressure and, more importantly, the velocity of the gas as it emerges from the engine. The area of this orifice is quite critical and is specified and calibrated at the time of manufacture, and any subsequent change will have a significant effect upon engine performance.

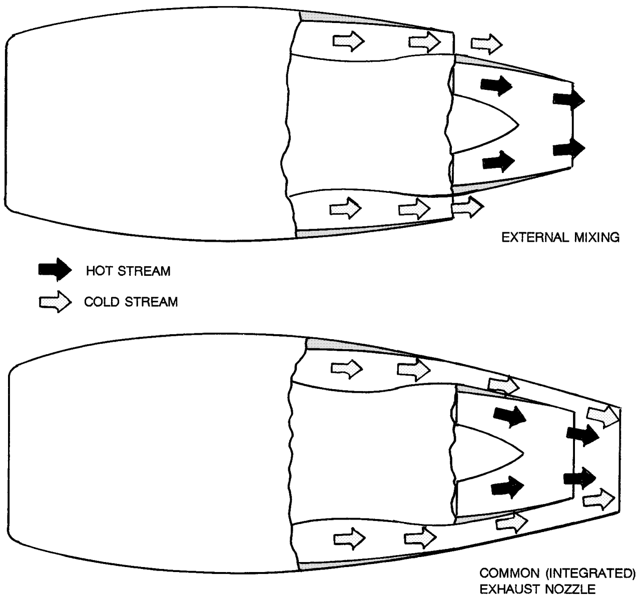
68. In the by-pass engine there are two gas streams to be exhausted, the cold (by-pass) air and the hot exhaust gas from the turbines. In low by-pass ratio engines the two gas streams are combined in a mixer unit before passing through the propelling nozzle. This is illustrated at Figure 21-31.

FIGURE 21-31
Internal Gas Flow Mixing



69. In high by-pass ratio engines the two gas streams are exhausted separately through hot and cold co-axial propelling nozzles. However, it is possible to gain some improvement in thrust if both streams are subsequently ejected through a common, or integrated, exhaust nozzle. These two concepts are illustrated at Figure 21-32.

FIGURE 21-32
Internal / External Gas Flow Mixing



Noise reduction

70. Much of the unacceptable noise produced by a jet engine is that caused by the exhaust jet velocity. Broadly speaking, the higher the exhaust jet velocity the more intense the noise created. It is largely for this reason that the turbo-fan engine is much quieter than the turbo-jet, since the bulk of the exhaust is relatively low speed cold air.

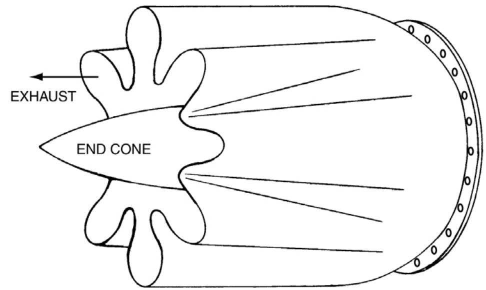
71. The pure jet (no by-pass) engine has the highest exhaust gas velocity and various devices have been introduced to reduce the noise of the jet exhaust. These, when retro-fitted to the engine, are often known as 'hush-kits'.

72. The distance that sound travels in air is dependent upon its frequency. High frequency sound is attenuated much more rapidly than low frequency and it has been found that the more readily the jet exhaust mixes with the surrounding atmospheric air the higher the frequency of the jet noise.

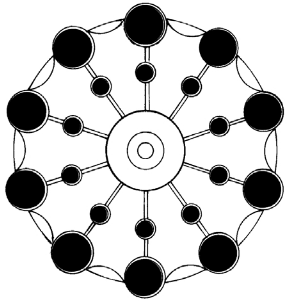
73. One method of encouraging mixing is to pass the exhaust gas through a number of radially mounted nozzles instead of one single nozzle. By reducing the size of the individual gas streams the frequency of the sound is increased. Since high frequency sound is attenuated more rapidly with distance in air than low frequency, so the sound at a given distance from the aircraft is less intense than would be the case with a larger single nozzle.

74. A similar effect to the multiple-tube nozzle is achieved with the corrugated perimeter nozzle. The outer perimeter of the propelling nozzle is 'fluted', or corrugated, to create what is, effectively, a ring of nozzles with the same total outlet area as that necessary for one single nozzle. Examples of multiple-tube and corrugated nozzles are shown at Figure 21-33.

FIGURE 21-33
Multiple-Tube and Corrugated Nozzle Exhausts



(A) CORRUGATED PERIMETER NOZZLE



(B) END VIEW - MULTIPLE TUBE NOZZLE

75. This section covers thrust reversal, thrust augmentation, the effects of air bleeding and the auxiliary gearbox.

Thrust Reversal

76. The use of high performance jet-engined aircraft created a need for alternative and supplementary methods of retarding the aircraft after landing. A simple and effective way of reducing the landing roll was found to be by reversing the direction of the exhaust gas stream, thus using engine power to decelerate through thrust reversal.

77. Though the rpm and fuel flow conditions during reverse thrust operation are similar to take-off power, the reverse thrust obtained is not more than 75% of maximum forward thrust, and is only about 50% in many cases. Use of full reverse power with no aircraft forward speed is inadvisable due to possible ingestion of debris and the possibility of compressor stall due to turbulent entry air.

78. The method of reverse thrust selection is essentially the same with all systems. A reverse thrust lever is interlocked with the engine thrust lever so that reverse thrust cannot be selected until the thrust levers have been moved to the rear of the IDLE setting. Engine thrust cannot be increased to a high setting unless the reversers are in the full reverse thrust position.

79. The reversers are designed for fail-safe operation. In the event of failure of operating hydraulic pressure a mechanical lock holds the reversers in the forward thrust position. Operation of the thrust reverser system is indicated to the flight deck crew to show when the reversers are unlocked, moving to and subsequently in the reverse position.

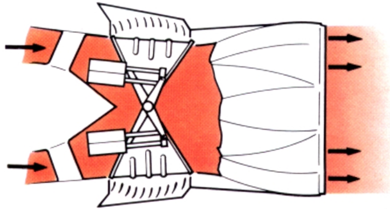
Hot Stream Thrust Reversal

80. In pure jet and low by-pass ratio engines the gas stream from the hot gas propelling nozzle is redirected to produce a forward component of velocity, so that reaction thrust is reversed from the normal, to oppose aircraft forward motion. There are two basic types of hot gas stream thrust reverser, the clamshell door and the bucket-target.

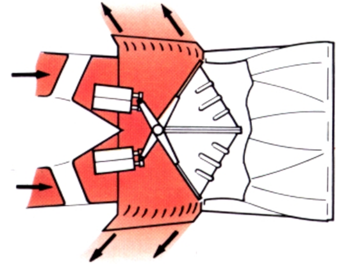
81. Clamshell Door reversers are pneumatically-operated doors which, in their normal (forward thrust) position close the reverse thrust ducts and allow the normal gas stream to flow through the exhaust nozzle. When reverse thrust is selected the doors rotate to block off the normal gas stream and divert the exhaust gas through ducts to cascade vanes, which direct the hot gas forward, creating a jet thrust in opposition to aircraft motion.

82. The pneumatic rams that operate the clamshell doors are designed to exert maximum force when the doors are in the normal, forward thrust, position. This has the effect of holding the doors firmly against the reverse duct seals, preventing gas leakage and loss of forward thrust. A clamshell door thrust reverser is illustrated at Figure 21-34.

FIGURE 21-34
Clamshells Thrust Reversers



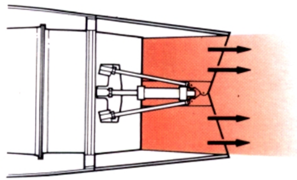
Clamshell doors in forward thrust position



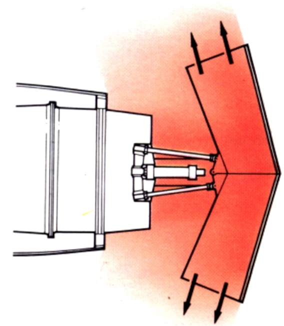
Clamshell doors in reverse thrust position

83. Bucket-Target reversers are hydraulically actuated. In the forward thrust position the bucket shaped doors form a concentric tube around the propelling nozzle, having no effect upon the exhausted gas stream. When reverse thrust is selected hydraulic actuators move the doors, by means of push rods, to deflect the jet stream from the propelling nozzle forwards, to create a reversal of jet thrust. A bucket-target thrust reverser is illustrated at Figure 21-35.

FIGURE 21-35
Bucket-Target
Thrust Reversers



Actuator extended and bucket doors in forward thrust position



Actuator and bucket doors in forward thrust position

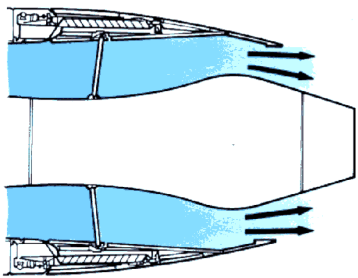
Cold Stream Thrust Reversal

84. In high by-pass ratio turbo-fan engines the greater proportion of the total thrust is produced by the cold (fan) air stream. Consequently it is usual in such engines for the cold stream only to be reversed when thrust reversal is selected. In the normal, forward thrust position the thrust reverser doors block off the gas-reversing cascade vanes.

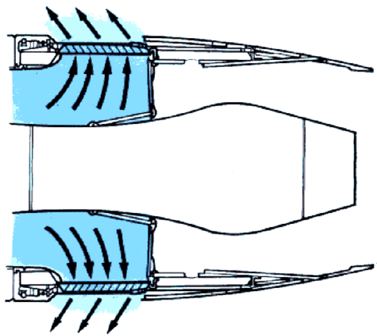
85. When reverse thrust is selected the doors are moved to completely obstruct the rearward flow of fan air to the cold stream propelling nozzle. All the fan discharge air is diverted through cascade vanes in the walls of the air duct. The cascade vanes impart a forward direction to the airflow as it leaves the engine, producing reversal of cold stream propulsive thrust.

86. Cold stream reverser systems are usually operated either by air motors through a system of screwjacks, or by hydraulic rams and push rods. A cold stream thrust reverser is illustrated at Figure 21-36

FIGURE 21-36
Cold Stream Thrust Reversers



Cold stream reverser in forward thrust position



Cold stream reverser in reverse thrust position

87. The main advantage of reverse thrust during a normal landing operation is that its use reduces the amount of wheel braking required and therefore reduces the possibility of loss of directional control during the landing roll. This is particularly useful when operating on a low friction surface such as a wet or icy runway.

88. The disadvantage of reverse thrust is that at low forward speeds the engine may ingest its own exhaust efflux or that from an adjacent engine. This is highly undesirable, as it tends to disrupt intake flow and can lead to surge and other handling problems. At very low forward speeds the jet efflux may throw up spray, dust or loose snow ahead of the aircraft, obscuring or obliterating forward vision. These problems are exacerbated when operating in crosswind conditions. Also, in some aeroplanes selection of reverse thrust creates a pitch up moment, which is clearly undesirable if it occurs whilst the nose wheel is still off the ground.

89. The pilot can reduce these problems by careful selection and de-selection of reverse thrust. In four-engine aircraft with wing mounted podded engines, ingestion of exhaust from a neighbouring engine (typically the outers ingest exhaust from the inners in a swept wing aircraft) occurs at about 80 knots. Consequently, the inners should be throttled back at about that speed. An engine in reverse thrust will be liable to start ingesting its own exhaust at about 50 knots, so the pilot needs to begin throttling back all engines at this speed.

90. The braking effect of reverse thrust is directly proportional to aircraft speed, so it is clearly desirable to select it as early as possible in the landing roll. However, in aircraft types where reverse thrust selection creates a pitch up moment its selection must be delayed until the nose wheel is firmly on the ground.

Thrust Augmentation Systems

91. An increase in altitude or in ambient temperature has an adverse effect on the power output of a gas turbine. Consequently the take-off power available will be reduced if take-off is from a high altitude aerodrome or one where the ambient air temperature is high. Thrust augmentation systems are used to restore lost thrust when operating from such aerodromes.

92. Alternatively, thrust augmentation systems may be used to increase the take-off performance when operating from low altitude and cool aerodromes, but perhaps with a short runway and/or a heavy aeroplane. In commercial aircraft the thrust augmentation method most commonly used is to increase the air mass flow by cooling the air flowing through the engine. This is achieved by introducing either de-mineralised water, or a mixture of de-mineralised water and methanol, into the airflow. The addition of methanol to the water gives anti-freezing properties and provides an additional source of fuel.

93. De-mineralised water (rather than tap water) is used for this purpose in order to prevent the hot section engine component reacting with chemicals in the water.

94. The water or water-methanol mixture may either be sprayed directly into the compressor inlet or, more usually, injected into the combustion chamber.

95. When water-methanol is sprayed into the compressor inlet, cooling increases air density. This increases the mass of the airflow and thus produces increased thrust. The methanol burns in the combustion chamber and thereby maintains turbine inlet temperature (TIT), which would otherwise fall. When water only is sprayed into the compressor inlet, density is restored but additional fuel flow is necessary to maintain TIT and, therefore, engine rpm.

96. With compressor inlet injection the coolant is usually metered from a storage tank by a control unit which distributes the coolant to radial passages in the compressor first stage disc. The coolant is forced outwards by centrifugal action to enter the airflow between the first stage blades.

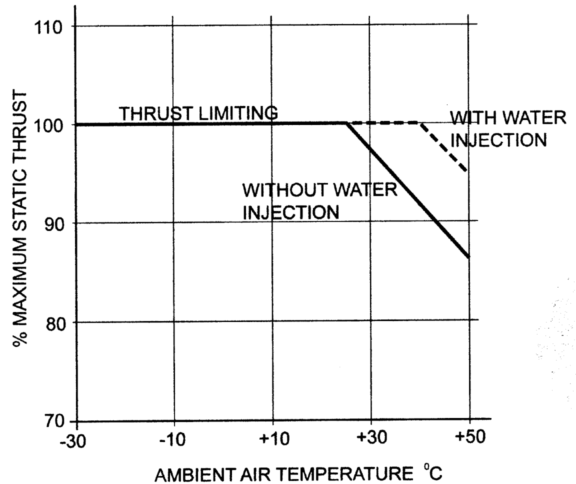
97. Injection of the coolant into the combustion chamber increases the mass flow through the turbine relative to that through the compressor. The consequent reduction of pressure and temperature drop across the turbine gives an increased jet pipe pressure, resulting in additional thrust. When water is injected the reduction in TIT enables the fuel system to schedule an increased fuel flow to increase rpm and thrust. When water-methanol is injected the burning of the methanol helps restore TIT and increased fuel flow is either unnecessary or minimal.

98. Combustion chamber injection typically employs an air-turbine driven pump discharging the coolant to water jets incorporated in the fuel spray nozzles. A sensing unit ensures that coolant can only flow to the spray jets when coolant pump pressure exceeds compressor discharge pressure, and actuates an indicator in the cockpit.

99. The primary advantage of water-methanol over water-only injection is that the combustion of the methanol either reduces or removes entirely the requirement to re-schedule the fuel control unit for thrust augmented take-off. A secondary advantage of mixing methanol with the water is that it acts as antifreeze. Consequently water-methanol can be transported in its own storage tank for use on subsequent occasions without the risk of it freezing.

100. Note that water or water-methanol injection is only used on take-off, when maximum thrust is essential. The graph at Figure 21-37 shows the typical thrust restoration with use of water injection on a turbo-jet engine

FIGURE 21-37
Water Injection Thrust Restoration Graph



101. Military aircraft (and Concorde) frequently employ reheat, or afterburning, to increase maximum thrust. Fuel is burned in the jet pipe, resulting in an increase in jet pipe temperature, pressure, velocity and hence thrust. Variable area jet nozzles are necessary for afterburning, to give a larger area during afterburner operation. This is a highly uneconomical method of increasing thrust and the increased jet pipe velocity makes it extremely noisy. Its main advantage is its ability to increase thrust without significantly increasing the size or weight of the engine. A typical afterburning jet pipe and variable area nozzles are illustrated at Figure 21-38 and Figure 21-39.

FIGURE 21-38
Principle of Afterburning

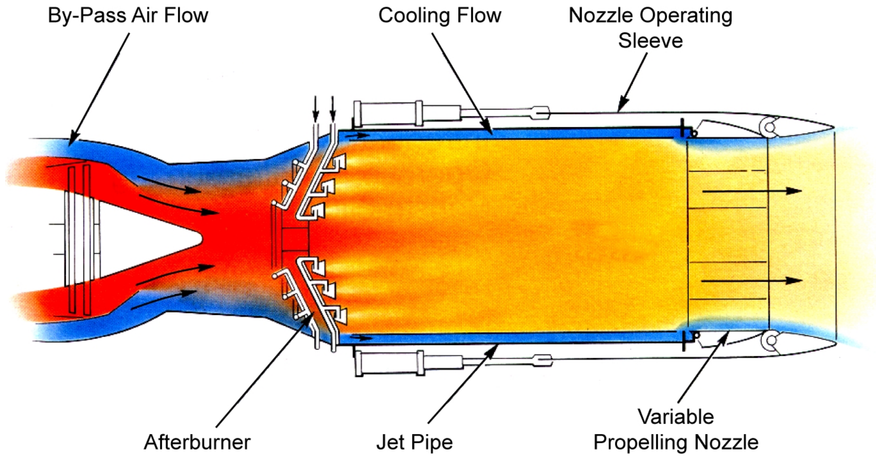
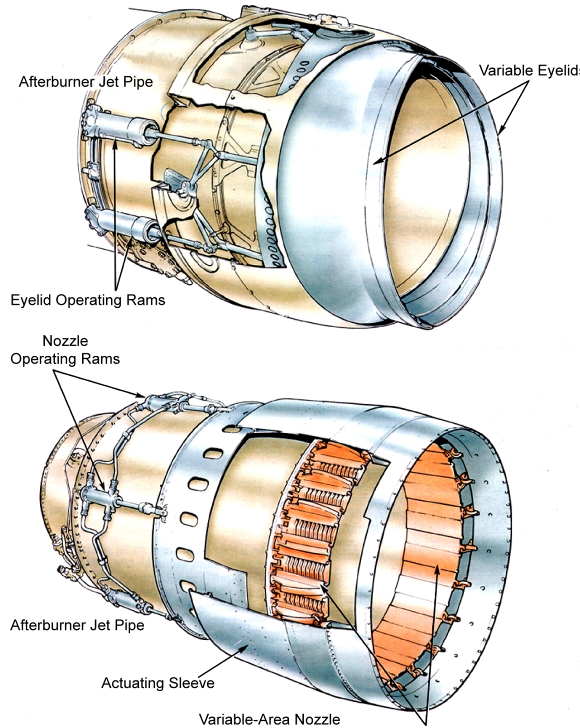


FIGURE 21-39
Afterburning Jet Pipes and Propelling Nozzles



Air Bleeds and Auxillary Drives

Bleed Air

102. Compressed air is bled from the LP and HP compressors for the engine's internal air system, for engine and airframe anti-icing and for the aircraft environmental control system (air conditioning and pressurisation).

103. Removal of air from the engine airflow clearly reduces the total air mass flow and therefore reduces the engine thrust. Thrust is measured and indicated as engine pressure ratio (EPR), which is the ratio of jet pipe pressure to LP compressor inlet pressure. Taking air from the compressors will reduce the mass of air entering the hot section of the engine and therefore reduce the jet pipe pressure, reducing EPR. The resulting loss of thrust increases SFC.

104. The reduced airflow through the combustion system and turbines will also cause an increase in turbine gas temperature (TGT), since the degree of cooling will be less.

105. The reduced air mass flow when air is being bled will usually cause a small increase in engine rpm, since the turbine has less work to do in driving the compressor. Depending upon the type of fuel control unit, the reduction in EPR may also lead to increased fuel scheduling with a resultant rpm increase.

106. The thrust reduction and increased TGT may well limit the use of bleed air during maximum power operations such as take-off and climb-out.

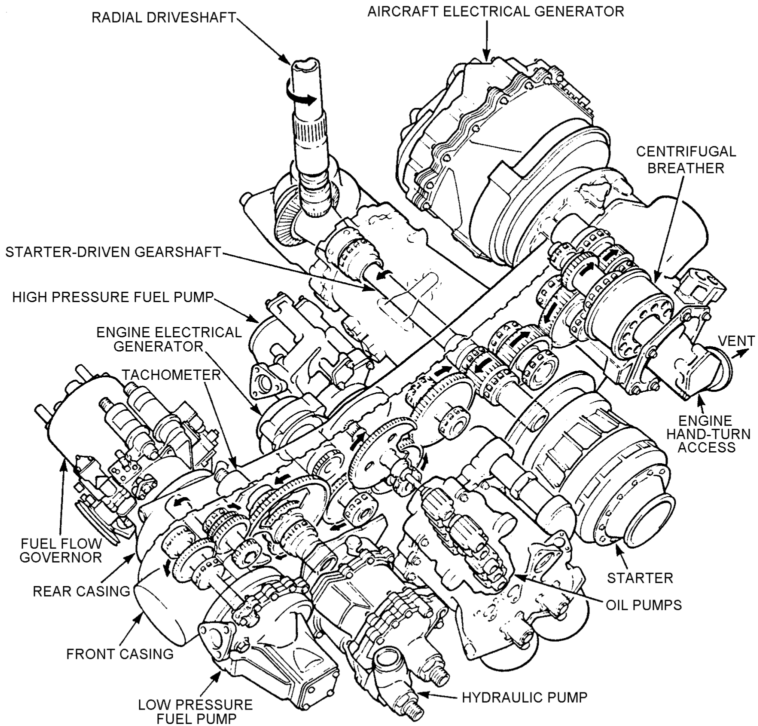
Auxiliary Gearbox

107. The various engine-driven accessories (fuel pumps, governor, lubricating oil pumps, hydraulic pump, and electrical generator) are driven by individual drive shafts from an external gearbox mounted on the engine casing.

108. The external gearbox receives its drive through a radial drive shaft, connected through bevel gearing to one of the gas turbine spools, usually the HP spool. A typical external gearbox is shown at Figure 21-40.

FIGURE 21-40

Jet Engine External Gearbox



109. Lubrication of the accessory drive gears is by engine lubricating oil. To protect the gearbox against damage in the event of accessory failure, some drive shafts, typically generator and hydraulic pump drives, incorporate a deliberately weakened shear section. Should one of these components tend to seize, the 'shear-neck' will fail before damage to gearwheels in the gearbox can occur.

Self Assessed Exercise No. 5

QUESTIONS:

QUESTION 1.

What does Newton's Second Law of Motion state?

QUESTION 2.

How do velocity, pressure and temperature vary when air flows through a divergent duct?

QUESTION 3.

How is thrust generated in a turbo jet engine?

QUESTION 4.

How does air mass flow and acceleration compare in a turbo jet and turbo prop engine?

QUESTION 5.

What form or type of combustion is used in a turbo jet engine?

QUESTION 6.

How and where is static pressure increased in a centrifugal compressor?

QUESTION 7.

What is total head pressure and how does it vary with increasing airspeed?

QUESTION 8.

Define compression ratio in a gas turbine compressor:

QUESTION 9.

State the purpose of variable angle inlet guide vanes in an axial flow compressor:

QUESTION 10.

How and where is static pressure increased in an axial flow compressor?

QUESTION 11.

Why are the blades and air passageways in an axial flow compressor made progressively shorter from front to rear?

QUESTION 12.

Why are the blades of an axial flow compressor twisted?

QUESTION 13.

How do air velocity and static pressure change in each stage of an axial flow compressor?

QUESTION 14.

What would be the consequences of tip clearance in a centrifugal compressor being too small?

QUESTION 15.

What is the main disadvantage of a double-entry impeller in a centrifugal compressor?

QUESTION 16.

What combination of mass air flow and pressure ratio are most likely to cause compressor surge?

QUESTION 17.

What are the outward symptoms of compression surge?

QUESTION 18.

What term is used to describe the values of airflow and pressure ratio at which surge occurs:

QUESTION 19.

What would be the effect of a surge bleed valve stuck in the open position:

QUESTION 20.

What is the purpose of surge bleed valves?

QUESTION 21.

Where in a gas turbine engine do the highest gas velocity and the highest gas pressure occur:

QUESTION 22.

What term is used to describe the air used in the combustion process in a gas turbine engine?

QUESTION 23.

Where in a gas turbine engine does the highest gas temperature occur?

QUESTION 24.

How and why are individual chambers connected in the multiple combustion chamber layout?

QUESTION 25.

What type of combustion engine achieves greatest combustion efficiency in a jet engine?

QUESTION 26.

What air : fuel ratio is required in a jet engine for efficient combustion?

QUESTION 27.

What factors limit the permissible maximum temperature at the combustion chamber outlet in a gas turbine engine?

QUESTION 28.

What is the function of the turbine nozzle guide vanes?

QUESTION 29.

What type of turbine blades are most commonly used in gas turbine engines?

QUESTION 30.

Describe the concept of Active Clearance Control (ACC) as used in some gas turbines:

QUESTION 31.

What is a free power turbine and where is it used?

QUESTION 32.

What is the main reason for shrouding of turbine blades?

QUESTION 33.

How does the gas cause the turbine to rotate in a gas turbine engine?

QUESTION 34.

How do gas, velocity, temperature and static pressure change as it passes through the turbine?

QUESTION 35.

Define the term choked nozzle as applied to exhaust nozzles.

QUESTION 36.

What is the purpose of the exhaust cone in gas turbine engines?

QUESTION 37.

How is the apparent noise reduced in a pure jet or low bypass ratio engine?

QUESTION 38.

How does the air flow through the fan and that through engine core compare in a turbo-fan engine?

QUESTION 39.

What form of power is used to operate clamshell door thrust reversers?

QUESTION 40.

Describe the Bucket type thrust reverse system?

ANSWERS:

ANSWER 1.

Newton's second law of motion states that the rate of change of momentum of a body is proportional to the applied force and takes place in the direction in which the force acts.

ANSWER 2.

When a gas flows through a divergent duct its velocity decreases and its pressure and temperature increase.

ANSWER 3.

To every action there is an equal and opposite reaction. The force resulting from the acceleration of a mass of gas rearwards produces reaction thrust forwards.

ANSWER 4.

Compared to a propeller of similar thrust, a jet engine gives a large acceleration, but to a relatively small mass of air.

ANSWER 5.

In the gas turbine cycle of operations combustion takes place at constant pressure (for a given thrust value).

ANSWER 6.

The impeller vanes form a divergent passage, which causes a rise in pressure as air is thrown outwards. On leaving the impeller the air enters stationary divergent passages (diffuser ducts) which convert the kinetic energy of the air into pressure energy. Hence, pressure rise occurs in both impeller and diffuser.

ANSWER 7.

Total head pressure is the sum of static and dynamic pressure. Dynamic pressure increases with airspeed.

ANSWER 8.

Compressor compression ratio is the ratio of compressor outlet pressure to compressor inlet pressure.

ANSWER 9.

The purpose of inlet guide vanes is to direct the air flow into the first stage rotor blades at an acceptable angle of attack.

ANSWER 10.

The air passage between adjacent blades is slightly divergent, causing a pressure increase. The air passage between the stationary vanes is also slightly divergent and so a further pressure rise occurs.

ANSWER 11.

The compression of the air as it passes through the compressor causes its volume to decrease. If the blades and air passageway were of uniform cross section this volume reduction would cause a reduction in velocity. By gradually reducing blade and air passageway dimensions, a uniform axial velocity is maintained.

ANSWER 12.

In order to maintain uniform axial velocity at all points on the rotor blades they are "twisted" from root to tip.

ANSWER 13.

The velocity gained in the rotor blades is converted into pressure in the following stator vanes. Hence there is ideally no overall velocity increase through a stage (rotor + stator).

ANSWER 14.

Impeller tip clearance is critical, if it is too small aerodynamic buffeting results.

There would only be a danger of seizure were bearing failure to occur. Pressure losses would occur if tip clearance is too large.

ANSWER 15.

In a double-entry compressor the air to the rear side of the impeller tends to be preheated as it passes the discharge from the front side.

Although increased turbine power will be necessary, this is true of any increase in compressor capacity. An advantage of the double entry compressor is that for the same airflow the impeller diameter can be reduced, hence frontal area can be reduced.

ANSWER 16.

A high pressure ratio (excessively adverse pressure gradient) and a low airflow are the ideal conditions for surge (reversal of airflow) to occur. Stall and surge occur at low compressor rpm.

ANSWER 17.

Surge, which is caused by a pressure gradient too great for the airflow to sustain, is evidenced by a combination of a loud "bang", the expulsion of air and combustion gases forward, a marked rise in EGT, a loss of thrust, and/or vibration.

ANSWER 18.

When compressor airflow is plotted against compressor pressure ratio the curve separating unstable (surge) and stable conditions is known as the surge line.

ANSWER 19.

If an anti-surge bleed valve sticks open there will be a loss of air mass flow to atmosphere, therefore less mass flow through the "hot" section and jet pipe. There will therefore be reduced thrust and less cooling of the combustion gases.
An open anti-surge bleed valve will prevent stall and surge.

ANSWER 20.

Anti-surge bleeds prevent surge by maintaining airflow through the early compressor stages at low rpm and mass flow by diverting some of the airflow overboard thus reducing pressure ratio.

ANSWER 21.

The highest gas pressure occurs between the final compressor outlet and the combustion chamber inlet (the diffuser section).
The highest velocity occurs in the propelling nozzle.

ANSWER 22.

The combustion chamber is designed to introduce a small proportion of the air flow (up to about 40%) into the primary or combustion zone. This proportion of the air flow is called the primary air.

ANSWER 23.

The highest gas temperature occurs in the combustion chamber primary zone.

ANSWER 24.

Multiple chambers are interconnected only for purposes of pressure equalisation and to assist flame spread during initial start. Interconnection tubes are used for this purpose

ANSWER 25.

The annular combustion chamber virtually eliminates unburnt fuel, reducing air pollution and increasing thermal efficiency.

ANSWER 26.

Whilst the overall mixture ratio in the combustion chamber may vary between 45:1 and 130:1, in the combustion zone of the chamber air will mix with the fuel at a ratio of about 15:1.

ANSWER 27.

The combustion gas temperature is limited by the maximum temperature that the turbine blades and turbine wheel can safely withstand.

ANSWER 28.

Nozzle guide vanes accelerate the combustion gas and direct it onto the first row rotor blades at the optimum angle.

ANSWER 29.

Impulse/reaction blading is most commonly used in gas turbine engines.

Pure impulse turbines are generally limited to very small engines, such as starters and auxiliary power units. Pure reaction blading is only theoretically possible. In practice there is always some impulse effect.

ANSWER 30.

A method of maintaining constant minimum turbine tip clearance over the whole operating range of the engine is known as Active Clearance Control usually abbreviated to ACC.

ANSWER 31.

In some turbo-prop and turbo-shaft layouts a turbine rotor independent of the HP or LP compressors is used to drive the propeller or output shaft. This is called a free or power turbine.

ANSWER 32.

The main function of turbine blade shrouding is to reduce tip leakage to a minimum.

Compressor stator vane shrouding is also used to reduce vibration.

ANSWER 33.

The rotational force applied to the turbine rotor is a combination of the impulse force of the gas being deflected by the blades and the reaction force due to the gas being accelerated by the convergent ducts formed by the blades.

ANSWER 34.

The turbine converts heat and pressure energy into work, hence both these values will fall. Gas velocity is increased during its passage between both stator vanes and rotor blades.

ANSWER 35.

As the pressure ratio across a convergent duct increases, the gas accelerates through it until the velocity at the throat equals the local speed of sound. Further pressure ratio increases do not result in further acceleration and the nozzle is said to be choked.

ANSWER 36.

The exhaust cone reduces gas velocity by diffusion and prevents gas flowing across the rear face of the turbine wheel.

ANSWER 37.

High frequency sound is attenuated more rapidly with distance in air than low frequency.

The higher the exit gas velocity, the greater the noise. Increasing the gas temperature will increase the choke velocity (that's what reheat does) and increase the noise. Increasing gas pressure simply chokes the nozzle, which is when it is noisiest.

ANSWER 38.

The mass of air accelerated by the fan (the cold stream) is vastly greater than the mass of gas accelerated by the combustion process (the hot stream).

ANSWER 39.

Clamshell Door reversers are pneumatically operated.

ANSWER 40.

With the target bucket system, when reverse thrust is selected the hydraulic actuators move the doors, by means of pushrods, to deflect the jet stream from the propelling nozzle forwards, to create a reversal of jet thrust.

Gas Turbine Engine Systems

Gas Turbine Engine Systems

Introduction

1. This section deals with the engine starting and ignition systems, the fuel systems and fuel control requirements of the various types aircraft gas turbine, the fuels used in aircraft gas turbine engines and the lubrication systems.

Starting Systems

2. Two separate systems are required to ensure that a gas turbine will start satisfactorily. Firstly, provision must be made for the compressor and turbine to be rotated up to a speed at which adequate air passes into the combustion system to mix with the fuel injected by the burners. Secondly, provision must be made for ignition (light-up) of the fuel/air mixture in the combustion system. During engine starting the two systems must operate simultaneously. It must be possible to turn the engine with the starter motor, without ignition, when required for maintenance checks or to blow out surplus fuel after a failed start. Similarly, it must be possible to use ignition alone (without starter) for relight, or prevention of flameout, in flight. The functioning of both these systems is co-ordinated during the start cycle.

3. Gas turbine units are started by rotating the compressor (and, of course, its associated turbine). In the case of dual rotor engines, the HP section is usually the one that is rotated, as this is invariably the hot gas section. The starter motor carries out two functions. Firstly it must rotate and accelerate the compressor up to a particular speed suitable for combustion to take place and secondly, once the light-up has taken place, assist the engine turbine until self-sustaining speed is exceeded.

4. Once self-sustaining speed is, the starter is cut out and the engine accelerates under its own power to idle speed. The torque provided by the starter must be in excess of the torque required to overcome the compressor inertia and the friction loads of the engine.

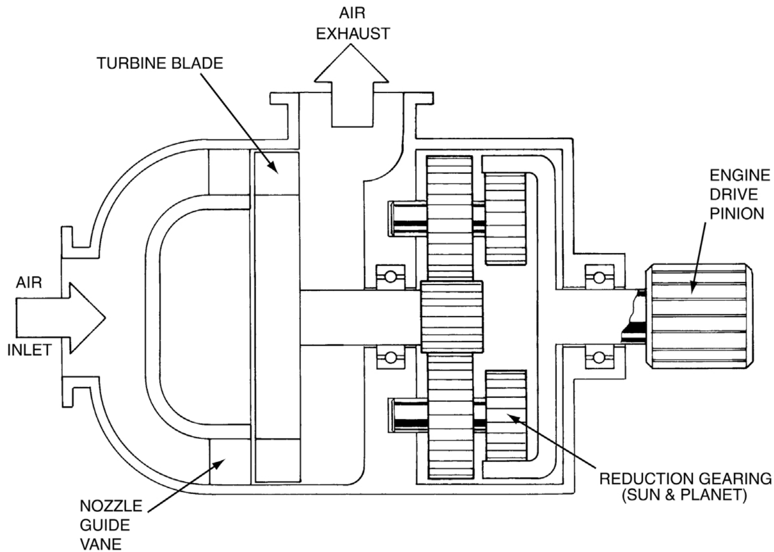
Engine Starting

Types of Starter Motor

Electric Starter Motor. A DC motor driving through reduction gearing. The electrical supply is automatically cancelled when the engine has satisfactorily started or, in the event of failure to start, when a time cycle is completed.

Air Starter Motor. A compressed air driven turbine rotor transmits power through reduction gearing. Air supply for the turbine may be from a ground supply, from an on-board auxiliary power unit, or bleed air from a running engine. This is the commonest type of starter unit used with modern high by-pass engines. A typical air starter motor is illustrated at Figure 22-1.

FIGURE 22-1
Jet Engine Air Starter Motor

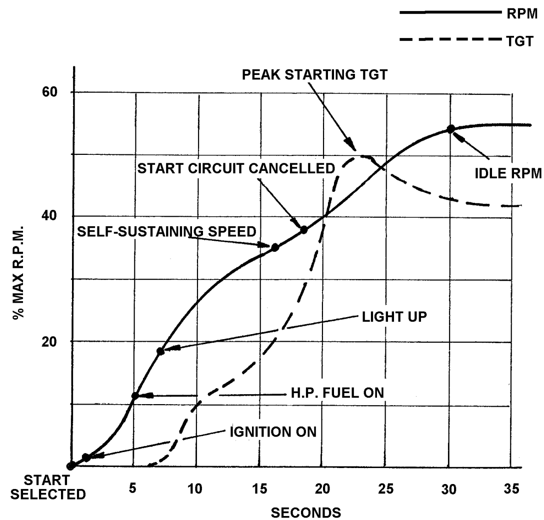


Gas Turbine Starter. A self-contained small gas turbine with a free turbine connected to the engine through reduction gearing.

Starting Sequence

5. A typical starting sequence for a gas turbine is as follows (refer to Figure 22-2). The starter is turned to rotate the compressor by actuating the start control switch. As soon as the starter has accelerated the compressor sufficiently to establish the airflow through the engine, the ignition is turned on and then the fuel flow. Both of these functions are achieved by shifting the start lever from cut-off to idle. This sequence during the starting procedure is important to prevent the danger of spontaneous ignition and over temperature.

FIGURE 22-2
Gas Turbine
Starting System



6. After light up, assisted by the starter motor, the engine accelerates. At this point the fuel flow rate is not sufficient to enable the engine to continue to accelerate satisfactorily on its own. If the starter is disengaged now the engine may not accelerate, may accelerate but far too slowly for the permissible time cycle, or may even decelerate due to insufficient energy. Hence, starter assistance is required after light up, and continues to assist the engine up to slightly above self-sustaining speed. Thereafter the engine can accelerate without the assistance of the starter motor.

7. After self-sustaining speed is exceeded, both the starter and ignition are cut off, either automatically or by releasing the start control switch. The engine now accelerates to, and stabilises at, idle rpm. The higher the rpm before the starter cuts out, the shorter will be the time required until idle rpm is reached. The entire starting cycle is accomplished in between 20 and 90 seconds, depending upon the type and make of engine and the conditions under which the start is attempted.

8. It is important that the pilot is aware of the maximum permitted time for the start cycle events. The pilot must be prepared to discontinue the start when it becomes evident that either the start cycle time will be exceeded (a hung start), or the EGT will exceed the maximum permitted peak starting value (a hot start). In the latter case, the start lever (HP cock) should be closed and the starting sequence cancelled immediately. The engine can subsequently be motored with start lever shut and ignition off, to cool the hot section. If the starter has already been de-energised, follow the manufacturer's recommended procedures.

9. If the engine is slow to accelerate the ignition circuit may be cancelled by a time switch before self-sustaining speed has been exceeded. In this case the engine may well continue to turn, but will not accelerate beyond self-sustaining speed. If this hung-start condition is allowed to persist, the turbine gas temperature is liable to become excessive, since fuel flow will be excessive for the airflow at the low rpm pertaining.

10. If light up does not occur clearly the engine will not accelerate and TGT will stay very low. With fuel being scheduled the combustion chamber(s) will flood and the excess fuel must be allowed to drain off. This is known as a wet start. The procedure to be followed is, ignition off, motor engine on the starter to blow out excess fuel and only attempt light-up (ignition on) when the excess fuel has gone - this may require allowing the engine to stop and the combustion chamber drain valves to open. Failure to get rid of the unburned fuel can result in a spectacular display known as torching, when the residual fuel ignites at the next start attempt and is ejected from the jet pipe as a stream of flaming fuel!

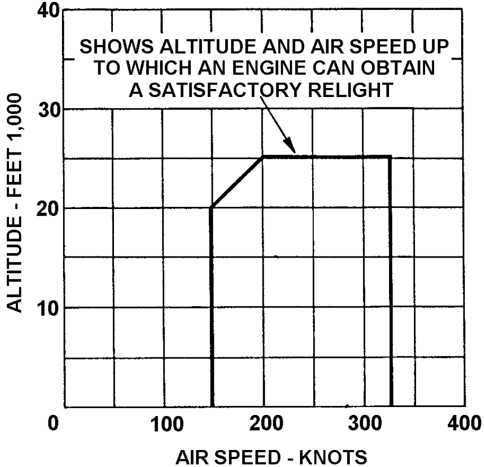
11. The automatic fuel control systems fitted to modern gas turbine engines are designed to prevent the occurrence of hung and hot starts. Self sustaining speed is typically in the order of 30% Nz and ground idle speed approximately 60% Nz or 25% N.

Flight Start or Re-Light

12. When an engine has been shut down in flight, it may be necessary to descend to a specified altitude and/or reduce speed to obtain a positive re-light. (This is referred to as the re-light envelope). If the engine is windmilling at sufficient speed, the starter need not be operated and all that is needed is ignition and fuel. For a flight re-light, the start control switch is selected to flight start, which will start the ignition. Fuel flow is initiated by means of the start lever to open the HP cock, and rising EGT and rpm confirm a satisfactory re-light. The engine will stabilise at idle rpm, which at altitude will be higher than ground idle rpm. The ignition can now be switched off and the thrust lever gradually advanced to increase power.

13. A vital element of the re-light envelope is IAS. Attempting a windmilling re-light with airspeed too low will not result in a satisfactory start. At low IAS the correct drill is to carry out a start-assisted re-light to ensure adequate acceleration of the engine during the starting sequence. A typical re-light envelope is illustrated at Figure 22-3.

FIGURE 22-3
Jet Engine Relighting Envelope



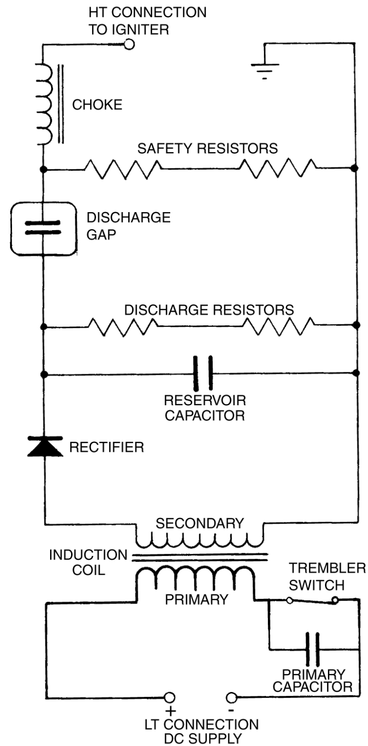
Ignition Systems

14. The gas turbine is a continuous combustion engine where a flame exists as long as fuel is flowing. Hence, an ignition system is only required for initial light up and to stabilise the combustion process. Once the engine has reached self-sustaining speed (the point above which it will accelerate to normal running speed without further assistance) the igniters are de-energised. The light up must take place under many different conditions and hence a high-energy system is used for starting all gas turbine engines. A dual ignition system is frequently fitted.

15. Each ignition system has a high energy ignition unit connected to its own igniter plug (not a sparking plug) and the two igniter plugs of a dual system are located in different positions in the combustion section. The system operates on either a 28v DC or 115v AC supply. Normally the igniters are automatically energised when, during the start cycle, the start lever (or HP cock) is set to deliver fuel to the rotating engine. They are then de-energised when the engine rpm reaches a speed at which combustion becomes self-sustaining.

16. A high-energy, twelve-joule DC ignition unit electrical system is shown at Figure 22-4. A trembler mechanism operates an induction coil, the output of which is rectified to put a high voltage charge on a condenser, or reservoir capacitor. When the voltage stored in the condenser equals the breakdown value of a sealed discharge gap, the condenser discharges to an igniter plug to produce a sequence of "flashovers" across the face of the plug. The function of the choke is to extend the period of discharge at the igniter plug.

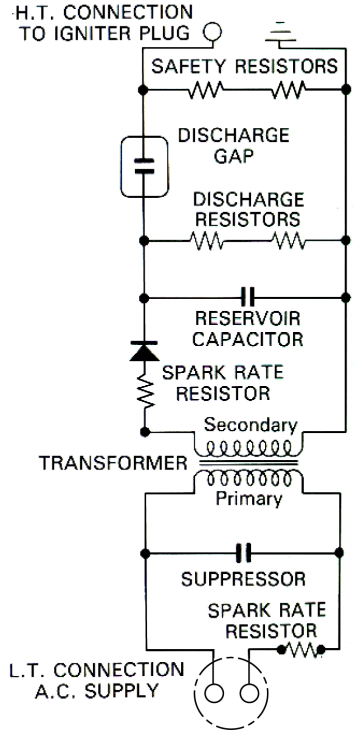
FIGURE 22-4
Jet Engine DC Ignition Unit



17. High-energy ignition systems are particularly necessary in circumstances where ignition of the fuel is likely to prove difficult, such as when re-lighting after a flame out at high altitude. For continuous operation of the ignition system a low-energy, three joule system is adequate and will result in longer life of the igniter plugs. Low energy gas turbine ignition units are normally supplied with AC. Continuous operation of the ignition system is necessary in flight conditions where flame extinction might occur, such as heavy rain, snow or icing.

18. A low-energy, three-joule AC ignition unit electrical system is shown at Figure 22-5. Alternating current is passed through a step-up transformer to charge a capacitor with high voltage. As with the previously described system, when capacitor voltage equals the breakdown value of a sealed discharge gap the stored energy discharges across the face of an igniter plug.

FIGURE 22-5
Jet Engine AC Ignition Unit



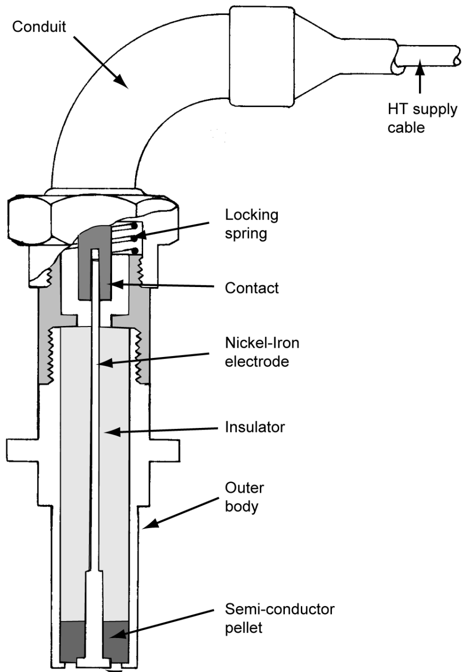
19. Ignition systems normally have a duty cycle specified. This duty cycle (typically 10 minutes of continuous operation of a single system) prevents damage and possible malfunction of the ignition systems. When operating in bad weather with the ignition on, it is simply a matter of remembering to switch from one ignition system to the other every ten minutes, or as otherwise specified.

20. Many gas turbine-powered aircraft employ ignition systems that are automatically energised when the angle of attack indicator senses an incipient stall condition, since compressor stall and surge are likely to add to your problems in this situation.

21. Igniter plugs are usually of the surface discharge type which, whilst energised, produce a high-energy flashover of 60-100 discharges per minute, as opposed to a gap-jumping spark. A typical igniter plug is shown in section at Figure 22-6.

FIGURE 22-6

Typical Igniter Plug



Fuel Systems

22. Regulating the quantity of fuel injected into the combustion system is the means by which the power, or thrust, of a gas turbine engine is controlled. When a higher thrust is required, the throttle is opened by the action of the thrust lever and the pressure of the fuel supply to the burners increases due to the greater fuel flow. This has the effect of increasing the gas temperature in the combustion chamber, which in turn increases the gas flow through the turbine to give a higher engine speed and correspondingly greater airflow. In consequence, greater thrust is produced. The fuel flow is minimum at idle power and maximum at take-off.

23. The main functions of the fuel control system are:

(a) To deliver fuel at the correct pressure and flow rate to the engine burners to produce the power required by the pilot.

(b) To adjust fuel flow as inlet conditions change in order to maintain a selected engine power (automatic barometric control).

(c) To provide extra fuel flow when rapid acceleration is demanded.

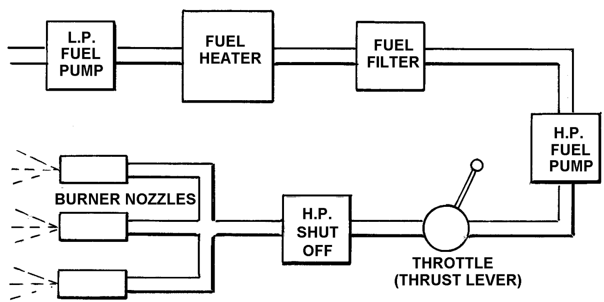
(d) To provide adequate fuel flow during engine starting.

(e) To enable the engine to be stopped by shutting off fuel supply.

24. In addition, the fuel control system must have automatic safety controls to prevent EGT, rpm and compressor delivery pressure from exceeding their design limitations. In order to accomplish the correct fuel flow, the fuel control unit senses the atmospheric and engine conditions, compares them to the throttle position set and delivers the fuel flow necessary to produce the desired engine thrust output. The pilot is thus relieved of normal speed, temperature and air flow/fuel flow adjustments.

25. Control of power or thrust of a gas turbine engine is by means of the throttle (thrust lever), with which fuel flow is manually adjusted for the power setting required. Having set the throttle, fuel flow is controlled automatically to compensate for changes in air density, and therefore air mass flow, due to changes in aircraft altitude, air temperature and aircraft speed. The fuel control unit (FCU), which may be a hydro-mechanical device or an electronic control system, achieves this automatic fuel control. A simplified fuel system for a gas turbine engine is shown at Figure 22-7.

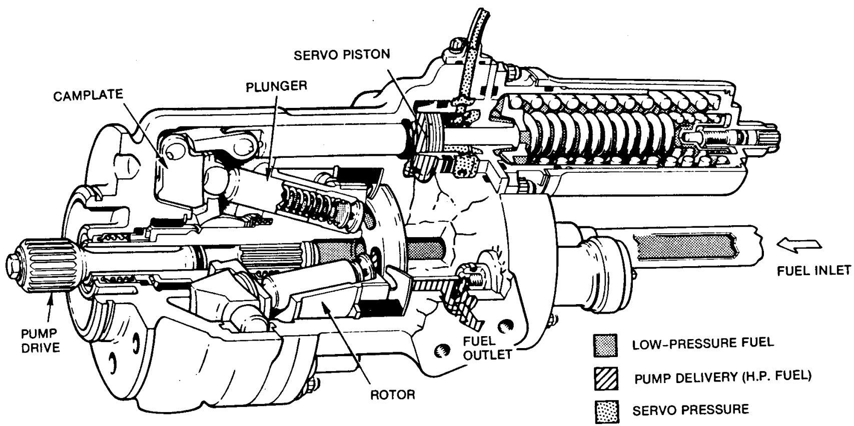
FIGURE 22-7
Simplified Gas Turbine Fuel System



26. Fuel is supplied, from the aircraft fuel tanks, to the high-pressure (HP) pump by a low-pressure (LP) system. The LP fuel pump ensures a constant supply at a suitable pressure to prevent vapour locking and cavitation of the HP pump supply, to ensure satisfactory engine operation. The LP system usually incorporates a fuel heater to prevent the formation of ice crystals which would block the fuel filter.

27. The HP pump is usually of the multi-piston swash plate type, delivering a variable quantity of fuel at constant pressure. Such a pump is illustrated at Figure 22-8.

FIGURE 22-8
HP Fuel Pump



28. Some low-pressure fuel systems use a spur gear HP pump. Fuel quantity delivered to the burners is set by the throttle lever and controlled by the fuel control unit. Fuel supply to the burners can be shut off altogether by the HP shut-off cock.

Fuel Control Systems

29. In most gas turbine fuel control systems altering the output of the HP fuel pump varies the supply of fuel to the burners. In hydro-mechanical fuel control systems the HP pump output is varied by a hydraulic servo system in response to variations of some, or all, of the following:

(a) Throttle position

(b) Ambient air temperature and pressure

(c) Engine rpm

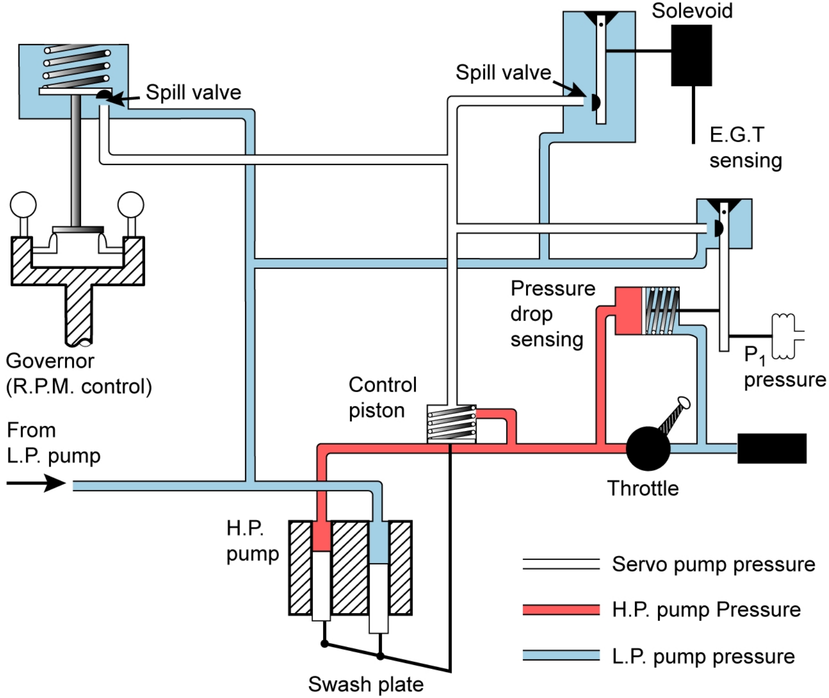
(d) Rate of acceleration or deceleration

(e) Exhaust/Turbine gas temperature

(f) Compressor delivery pressure

30. Figure 22-9 shows a simplified pressure control system designed to adjust fuel supply to the burner spray nozzles in response to changes of throttle position, engine rpm, P_1 (air intake) pressure and exhaust gas temperature.

FIGURE 22-9
Simplified Jet Engine Fuel Pressure Control System



31. The HP fuel pump is of the swash plate type illustrated at Figure 22-8, in which variation of the swash plate angle varies pump output. A control piston sensitive to the pressure in a servo system varies the pump swash plate angle. Control of rate of acceleration/deceleration in such a system is usually incorporated in the throttle valve mechanism. A dashpot system limits the rate at which the throttle valve opens or closes, regardless of how rapidly the throttle lever is moved. The operation and interaction of the component parts of the Pressure Control system shown at Figure 22-9 is discussed in the following paragraphs.

32. HP Pump pressure is maintained at a constant value by the control piston. If pump discharge pressure tends to rise the piston moves up, reducing the pump swash plate angle and hence pump output. If pump discharge pressure tends to fall the control piston moves down, under the influence of the pressure control spring, increasing pump output. HP pump pressure is supplied direct to the throttle valve and also supplies the servo system with a constant supply through a fixed orifice.

33. Servo Pressure is controlled by a number of spill valves, which control the rate at which fuel is spilled back to the LP system. If flow through the spill valves exactly equals supply through the control piston fixed orifice, servo pressure will remain constant. If spill exceeds supply, servo pressure will fall, if spill is less than supply, servo pressure will rise.

Throttle movement. The degree of opening of the throttle valve determines the pressure drop across the valve. Given that the control piston ensures a constant supply pressure to the throttle valve, the more the valve is opened the less the pressure drop across it will be. Opening the throttle increases the supply pressure to the burner nozzles (throttle outlet pressure), throttle inlet pressure is maintained constant by the servo-operated pump control piston. The pressure drop across the throttle valve is sensed by a spring-loaded diaphragm, which controls the position of a spill valve.

34. If the throttle is opened, for more power, pressure drop across the diaphragm decreases and the spring force moves the diaphragm to close the spill valve. Pump servo pressure increases, which acts upon the control piston to increase HP pump output, increasing fuel flow to the burner nozzles. Closing the throttle increases the pressure drop across the diaphragm, opening the spill valve to decrease pump servo pressure, which results in a decreased HP pump output.

Engine rpm is sensed by an engine-driven governor, which positions a spill valve. If rpm increases the governor flyweights move outwards to open the spill valve and reduce pump servo pressure. This causes the control piston to decrease HP pump output and engine rpm falls back to the pre-set value. If rpm drops below this value the governor spring overcomes the force of the flyweights to close the spill valve, increasing servo pressure and HP pump output.

Compressor Air Intake (P_1). Pressure is sensed by a capsule attached to a spill valve. An increase in intake pressure (due to decreased altitude, increased airspeed or decreased air temperature/density) requires a corresponding increase of fuel flow. Increased pressure will expand the pressure-sensing capsule, closing the spill valve to increase pump servo pressure and HP pump output. A fall in air intake pressure will have the reverse effect.

Exhaust Gas Temperature. It is important that the exhaust gas temperature is limited, in order to avoid damage to the turbine blades and inlet guide vanes. EGT is sensed by thermocouples and the electrical output, which is proportional to temperature, activates a solenoid to open a spill valve as temperature increases, reducing pump servo pressure and therefore HP pump output.

35. Pressure Control fuel control systems are quite commonly used for turbo-jet and turbo-propeller engines. Alternative hydro-mechanical fuel control systems are the Flow Control system, the Combined Acceleration and Speed Control system and the Pressure Ratio Control system. The type of system chosen will depend upon engine operating parameters and engine type.

Flow Control System

36. Flow control systems differ from pressure control in that fuel pump delivery pressure is not maintained constant, but is proportional to engine speed. Fuel pump output (fuel flow) is controlled to maintain a constant pressure drop across the throttle valve at constant air intake conditions. The system is better suited to engines requiring large fuel flows.

37. The control system components, with the exception of the HP fuel pump, are contained within a combined fuel control unit. These components are the engine speed governor, an altitude sensing unit, an acceleration control unit and a gas temperature control unit.

Engine Speed Governor. A hydro-mechanical device which senses engine speed and produces a proportional hydraulic servo pressure to control fuel pump stroke (and therefore output).

Altitude Sensing Unit. This is a barometric pressure control, which senses air intake (P_1) pressure and operates a spill valve to control servo pressure (and therefore fuel flow) in conditions below governing speed.

Acceleration Control Unit. This senses compressor delivery pressure and adjusts a spill valve to control servo pressure to match fuel flow to airflow.

Gas Temperature Control. If engine gas temperature exceeds a maximum limit a solenoid-operated proportioning valve is progressively energised to open a spill valve and reduce servo pressure, reducing fuel flow.

Combined Acceleration and Speed Control System

38. This is a mechanical system that does not use spill valves. The controlling unit is a fuel flow regulator, which is engine driven and controls engine speed by controlling fuel flow. It contains two governors, a speed control governor and a pressure-drop control governor. The speed control governor is set by the engine throttle lever and controls fuel flow to the burners by means of a sleeve valve.

39. The pressure-drop governor senses any pressure drop due to an increased fuel flow and maintains fuel flow, by adjustment of a second sleeve valve, at a set value relative to engine rpm. A capsule unit compares compressor inlet and outlet pressures and adjusts fuel flow to match airflow. Engine gas temperature is sensed electrically and a rotary actuator automatically adjusts the throttle mechanism if maximum temperature is reached.

Pressure Ratio Control System

40. This is a mechanical system, similar to speed and acceleration control, which uses the ratio of HP compressor delivery pressure (P_4) to air intake pressure (P_1), to control fuel flow. This system is particularly responsive to conditions of surge or flame extinction (flame out). Either condition will cause the P_4/P_1 pressure ratio to be abnormally low and the system will substantially reduce, or even shut-off, fuel flow to the burners.

Electronic Engine Control

41. Many modern gas turbine engines incorporate electronic controls that monitor engine performance and operate the engine controls to maintain certain parameters within pre-set operating limits. These parameters are, typically, engine spool speeds, exhaust gas temperature and engine pressure ratio (EPR). The system may act simply as a limiter, preventing pre-set parameters from being exceeded, or it may be a supervisory control system, which maintains the thrust condition set by the pilot, regardless of changing atmospheric conditions.

42. Currently the ultimate extension of this concept is full authority digital engine control (FADEC), which replaces most of the hydro-mechanical and pneumatic functions of the fuel control system and virtually takes over all steady state and transient control of the engine conditions. The fuel system retains only sufficient controls (throttle and HP shut-off cock) for safe operation in the event of a major electronic failure.

43. A slightly less sophisticated electronic control system than FADEC, known as full authority fuel control (FAFC) uses an electronically-controlled fuel system, but does not have the same degree of transient condition control of the compressor airflow system.

44. Supervisory systems trim the fuel flow scheduled by the governor to match actual engine power with that demanded by throttle lever position. The engine supervisory control system monitors such parameters as thrust lever angle, engine compressor bleed state, engine pressure ratio (turbine outlet to compressor intake) and altitude, airspeed/Mach No and air temperature information from the air data computer (ADC). The supervisory system computes the LP spool speed (N_1) necessary to achieve the EPR demanded by the thrust lever angle set by the pilot and compares command EPR with actual EPR. Fuel flow is adjusted to achieve the necessary N_1 and trimmed to achieve the command EPR.

45. Throughout the fuel flow/power adjustment the control system continuously monitors N_1, EPR and EGT and operates on a 'lowest wins' precept. That is to say, the system will not allow limiting values of any of those three parameters to be exceeded.

Gas Turbine Fuels

46. Gas turbines require a less volatile fuel than piston engines and that most commonly used is kerosene. Kerosene occupies a narrow band of the fuel distillation process, between gasoline and the heavier fuel oils. Kerosene is denser than gasoline and has a slightly higher calorific value. Its freezing point varies according to its hydrocarbon content, but solid waxy particles that are capable of blocking fuel filters begin to form at temperatures between -40°C and -50°C. The lower volatility of kerosene means that it is much less susceptible to vaporisation in fuel tanks or systems than gasoline.

47. UK commercial aircraft use JET A-1, which is a kerosene type of fuel of relatively low volatility. The equivalent in the USA is known as JET A.

48. In some countries, notably the USA, wide-cut gasoline fuels are available. These fuels are a mix of kerosene and gasoline and have the advantage of a much lower freezing point than pure kerosene, typically about -60°C. Thus they are suitable for use in geographic locations where low temperatures prevail and at extreme altitudes. Wide-cut gasoline fuels are much more volatile than kerosene and produce flammable vapour over a wider range of temperatures, making then more hazardous when fuel tanks or systems are ruptured and more prone to vapour locking.

49. Wide-cut gasoline fuel is designated JET B and it has a lower specific gravity (SG) than JET A-1. If used in an aircraft with a pressure control fuel system the lower SG fuel will result in the governor maintaining a greater engine speed. Consequently, with a change in SG of the fuel, the governor must be reset before the engines are operated. Combined acceleration and speed control fuel systems compensate for changes in SG of the fuel, precluding any need for adjustment upon change of fuel.

Fuel Additives

50. Additives have been devised that reduce the temperature at which the water content of kerosene will freeze. For aircraft operating at extreme altitude or in very cold climates this is usually a preferable alternative to the use of volatile wide-cut gasoline. The principal disadvantage to the use of icing inhibitors is the need to ensure that fuel system components are not susceptible to attack from them. In general, an icing inhibitor works by combining with water in the fuel at low temperatures, to reduce the freezing point of the water/inhibitor mix below that of kerosene.

51. Microorganisms, in the form of fungi or bacteria can live and grow in tanks containing kerosene, provided water is present in the fuel, which it invariably is to some extent. The microorganisms appear as a slimy deposit in the fuel, typically coloured red, brown or black. They produce chemicals that are highly corrosive to metal and which can lead to serious weakening of the wing structure in aircraft with integral fuel tanks, even penetrating the protective coating applied to the inner surface of the tanks.

52. Fuel additives are commercially available that prevent the growth of microorganisms, particularly fungi, and these are essential in gas turbine-powered aircraft whose tanks have not been coated with fungus-resistant material.

53. JET A1 is available with or without fungus suppressant and icing inhibitor. When it contains these additives it is known as JET A1 (FSII).

Water in Fuel

54. If there were no water content in fuel the problems of ice crystal formation at low temperatures and microorganism growth would be virtually non-existent. Unfortunately, fuel has an affinity for water and it is impossible to remove it altogether, although every precaution is taken to exclude it as far as possible. Whilst bulk water is clearly visible and conveniently sinks to the bottom of a fuel container, it is dissolved water (water in solution) that is particularly difficult to detect and remove.

55. The amount of water that kerosene can hold in solution is largely dependent upon temperature, the higher the temperature of the fuel the greater the saturation level. As fuel temperature falls, that water content above the saturation level separates as small water particles, which combine with the fuel to form a frozen gel.

Gas Turbine Lubrication Systems

56. The lubricating oil system of a gas turbine engine provides lubrication and cooling for all parts of the engine where moving parts are in contact either with other moving parts (for example gears) or with stationary parts (for example bearings). In a turbo-prop engine the oil system also provides lubrication for the propeller reduction gearing and oil for the propeller pitch control mechanism.

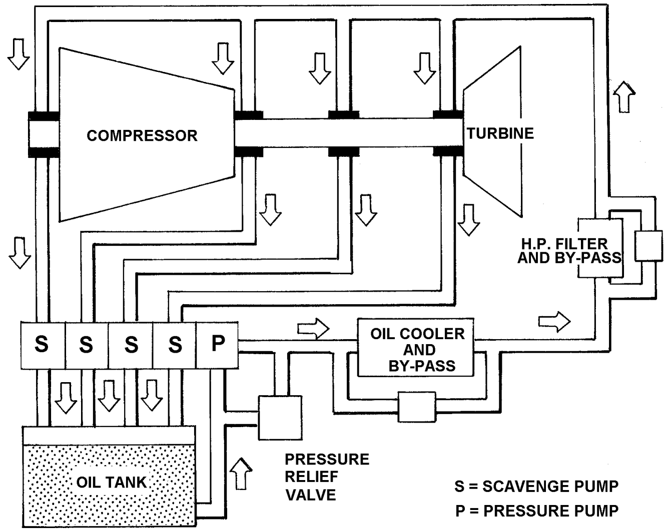
57. The oil used must protect against corrosion and its viscosity must be low enough to permit flow at low starting temperatures, but high enough to withstand and absorb mechanical loads. Because these loads are generally much lower than in a piston engine, gas turbine lubricating oil is usually of lower viscosity. This makes low temperature starting easier and normal starts, without the need to pre-heat the oil are possible at temperatures down to -40°C. Mineral oils are generally not of sufficiently constant viscosity over the wide operating temperature ranges of a gas turbine engine, so synthetic low viscosity oils have been developed.

58. Commercial aircraft gas turbine engines use recirculatory lubrication systems that are self-contained. Oil is contained in an oil tank and distributed around the engine by oil pumps, with sufficient reserve of oil in the tank to cope with normal, minor losses in flight. Some short duration engines employ a total loss lubrication system in which oil is expended overboard after circulation through the engine. There are two basic types of recirculatory system, the pressure relief valve system and the full flow system. In both systems the factors crucial to safe operation are oil temperature and oil pressure and these are invariably indicated on the flight deck.

Pressure Relief Valve System

59. In this system the oil flow to the bearing chambers is controlled by maintaining a constant pressure in the oil supply line. A spring-loaded pressure relief valve on the outlet side of the oil pressure pump opens to return oil to the oil tank or to the suction side of the pump. The valve is set to begin opening at a pressure corresponding to engine idling speed. The faster the engine (and, therefore, pump) speed the more the relief valve opens to maintain constant supply pressure. A schematic diagram of a pressure relief valve system is shown at Figure 22-10.

FIGURE 22-10
Jet Engine Pressure-Relief Oil System



60. An inherent problem with this type of system is the tendency of bearing chamber pressure to increase with increased engine speed. This creates a back-pressure, which reduces the oil flow to the bearing chambers as engine speed increases. Hence, the pressure relief valve system is unsuitable for engines where a high bearing chamber pressure is necessary because of high bearing loads. The problem can be alleviated to some extent by augmenting the pressure relief valve spring force with bearing chamber pressure, but in engines where chamber pressure is necessarily high an alternative lubricating system, the full flow system, is used.

Full Flow System

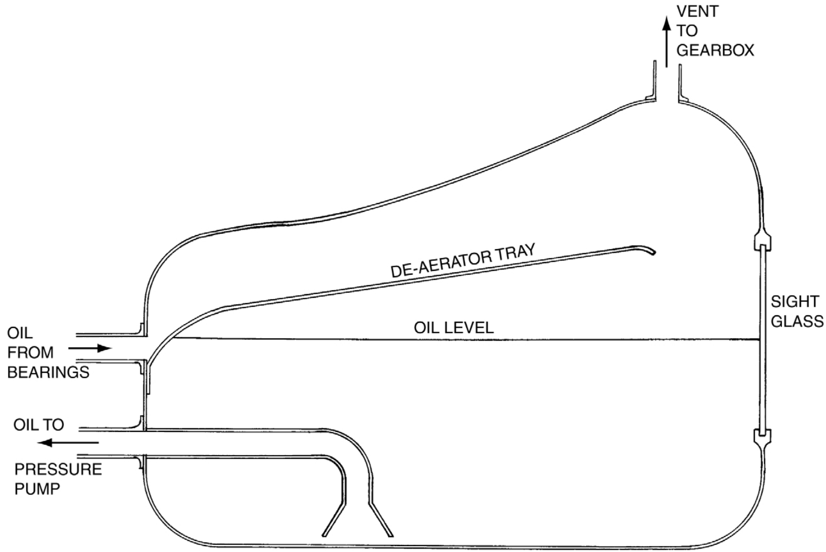
61. This system does not use a pressure relief valve, but instead feeds oil pressure pump output direct to the oil supply line to the bearing chambers. Thus, the higher the engine (and pump) speed the greater the quantity of oil supplied to the bearings. Because all the oil pumped is supplied to the bearings, with none being spilled back by a pressure relief valve, pump sizes can be significantly smaller with this type of system.

62. Because of the higher pressures associated with the full flow system, pressure-limiting by-pass valves are fitted in conjunction with components such as coolers and filters, which could otherwise be damaged. Both the Pressure Relief Valve System and the Full Flow System are dry sump lubrication systems.

Lubricating Oil System Components

63. The oil tank is mounted externally on the engine and incorporates a sight glass or dipstick for checking contents. Oil tank replenishment may be by pressure or gravity filling. The tank has a de-aeration device in the return line to remove air from the oil returning from the bearing chambers. A typical gas turbine engine oil tank is shown at Figure 22-11.

FIGURE 22-11
Typical Gas Turbine Oil Tank

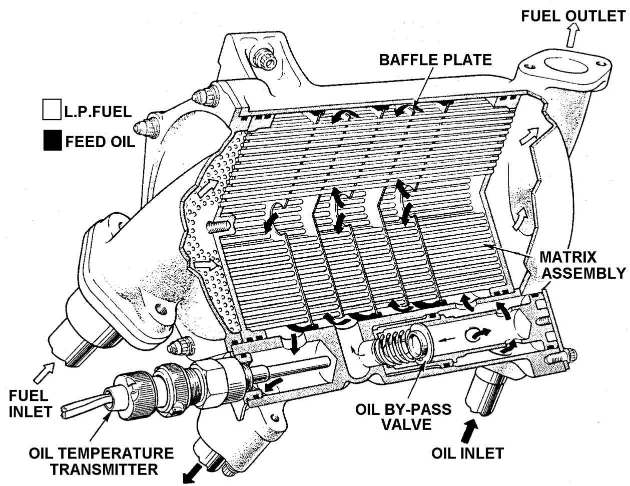


64. The scavenge pumps and pressure oil pump are usually of the spur gear type, although vane type or gerotor (Roots type) pumps are used on some engines. The scavenge pumps and pressure pump are contained in a common casing with a single drive from the accessories gearbox.

65. Oil distribution to gears, bearings, etc is usually by means of a spray from a jet orifice.

66. The oil cooler is a heat exchanger comprising a matrix of tubes through which the cooling medium flows. Oil is directed over the outside of the tubes and heat is transferred by conduction from oil to coolant. The cooling fluid is either fuel or air and in some engines both fuel-cooled and air-cooled oil coolers are used, the latter only being brought into use at high powers when the fuel-cooled cooler cannot cope. An example of a fuel-cooled cooler is shown at Figure 22-12.

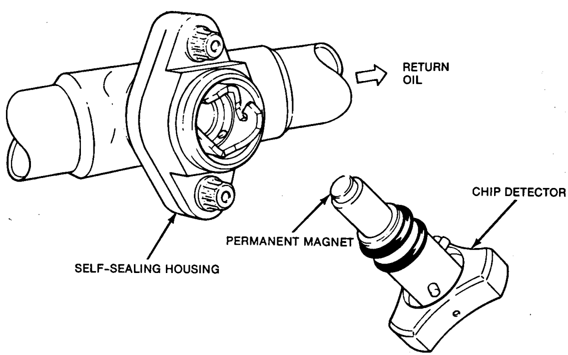
FIGURE 22-12
Fuel-Cooled Oil Cooler



67. A by-pass valve both protects the cooler, and the engine, in event of blockage of the cooler. A pressure-maintaining valve, which maintains oil pressure at a higher value than fuel pressure, is also a common feature. This ensures that, in the event of tube failure, oil leaks into the fuel rather than the reverse, which could be damaging to engine bearings.

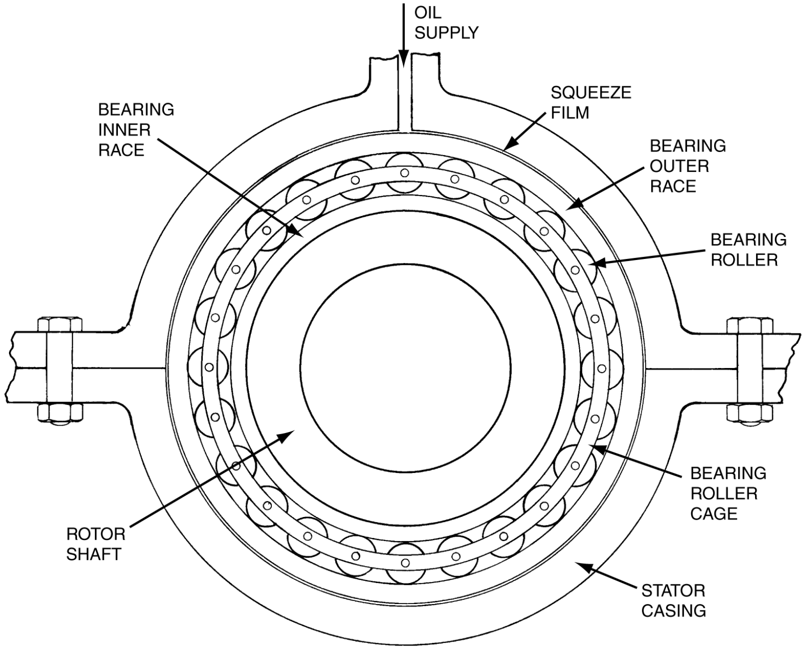
68. Small particles of metallic debris from the bearing chambers and gearboxes are collected by magnetic chip detectors fitted in the scavenge side of the system. Small permanent magnets collect ferritic metal particles that, upon examination, can give warning of potential component failure. Identification of the collected material can often provide identification of precisely which gear or bearing is failing. Magnetic chip detectors (MCD's) are often connected to a flight deck warning system to give in-flight indication of impending component failure. An example of a magnetic chip detector is illustrated at Figure 22-13.

FIGURE 22-13
Magnetic Chip Detector



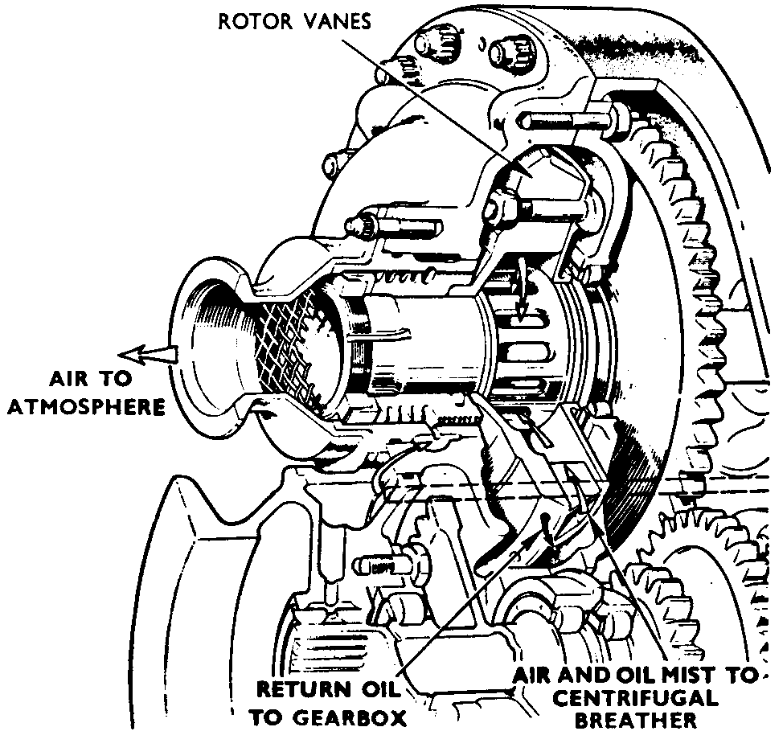
69. Gas turbine engine bearings are of the ball or roller type. In order to reduce the transmission of vibration from the rotating assembly to the bearing housing a squeeze film is often used. Oil from the pressure supply system is fed to a narrow clearance between the outer race (cage) of the bearing and the bearing housing. The film of oil filling the clearance absorbs the radial shock loads of vibration. A squeeze film bearing is illustrated at Figure 22-14.

FIGURE 22-14
Squeeze Film Bearing



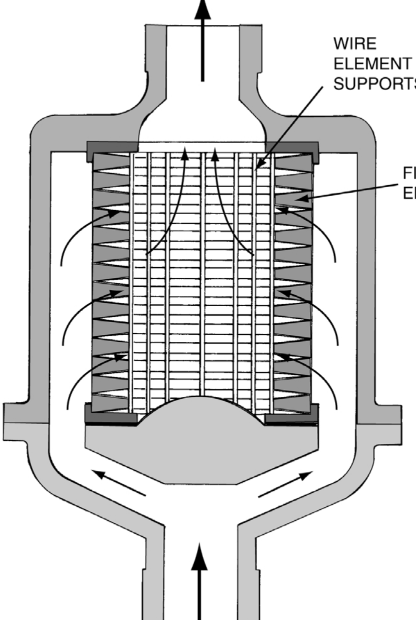
70. Air, which tends to accumulate in the oil at the bearings, is separated and exhausted to atmosphere, before the oil returns to the tank, by a centrifugal breather as shown at Figure 22-15.

FIGURE 22-15
Centrifugal Breather



71. The lubricating oil is filtered at various points in the system to prevent the circulation of debris. Coarse strainers are fitted at the oil tank outlet and/or scavenger pump inlet. Fine, pressure filters are fitted at the pressure pump outlet to prevent blockage of the oil feed jets. In addition, small thread-type filters are often fitted just upstream of the feed jets. An example of a pressure filter, with a number of wire-wound elements providing edge filtration, is at Figure 22-16.

FIGURE 22-16
Jet Engine Pressure Filter



Gas Turbine Engine Performance and Operation

Gas Turbine Engine Performance and Operation

Introduction

1. The performance of a gas turbine engine is dependent upon the mass of air that passes through the engine and the acceleration imparted to that mass by the engine. In the turbo-jet and turbo-fan engine the performance is measured in terms of thrust at the propelling nozzle or nozzles, in the turbo-prop engine it is measured as shaft horsepower (SHP).

2. The mass of air passing through the engine, and therefore the thrust developed, is dependent upon the air density, which varies with temperature and pressure. The forward speed of the aircraft also affects air mass flow through the engine, since there is a ram effect at the air intake when the aircraft is moving.

3. Static thrust, or gross thrust, is the amount of thrust produced by the engine when the aircraft is stationary on the ground or the engine is on a test bench. Net thrust is the thrust produced by the engine when the aircraft is in flight. The static thrust produced by an engine is largely the product of the mass of air passing through the engine and the velocity of the exhaust gas at the propelling nozzle. To calculate net thrust it is necessary to take into account the speed of the aircraft.

The Thrust Formula

4. The mass of air passing through the engine is the weight of air (w_a) divided by acceleration due to gravity (g). The thrust produced is the product of this and the velocity of the jet efflux (v_j). The static thrust force is given by the formula:

$$F = \frac{Wa \times Vj}{g}$$

5. To calculate the net thrust it is necessary to include the forward speed of the aircraft in the equation, because the thrust developed will depend upon the difference between the exhaust gas velocity (V_2) and the initial gas velocity (V_1). The initial gas velocity is, of course, the same as the aircraft velocity. Hence, v_j in the static thrust formula can be substituted with $V_2 - V_1$ and the formula for calculating static or net thrust becomes:

$$F = \frac{Wa}{g}(V_2 - V_1)$$

6. When calculating static thrust, V_1 will be zero.

7. As described in the section dealing with propelling nozzles in (section 3.37), a turbo-jet operating under choked conditions derives additional thrust from the pressure drop across the propelling nozzle. This added thrust is the product of the pressure difference across the nozzle and the area of the nozzle throat and is given by:

$$F = (P - P_0) \times a$$

Where: p = static pressure at the nozzle throat

p_0 = ambient atmospheric pressure

a = area of the propelling nozzle throat

8. Consequently, the static thrust of a turbo-jet engine operating with the propelling nozzle in a choked condition is given by the formula:

$$F = \frac{Wa}{g}(V_2 - V_1) + (P - P_0) \times a$$

9. When considering a turbo-fan engine there are two gas streams to be taken into account, the hot gas stream from the jet pipe and the cold gas stream from the fan. Given that the hot gas exhaust is normally un-choked in such an engine, the formula for calculating the total thrust may be given as:

$$F = \frac{Wa}{g}(V_2 - V_1)(\text{Jet}) + \frac{Wa}{g}(V_2 - V_1)(\text{Fan})$$

Turboprops

10. The output of a turboprop is measured normally as shaft horsepower (SHP) or thrust equivalent shaft horsepower (TESHP). TESHP takes account of the residual jet thrust of a turboprop exhaust as shaft horsepower.

Factors Affecting Power Output

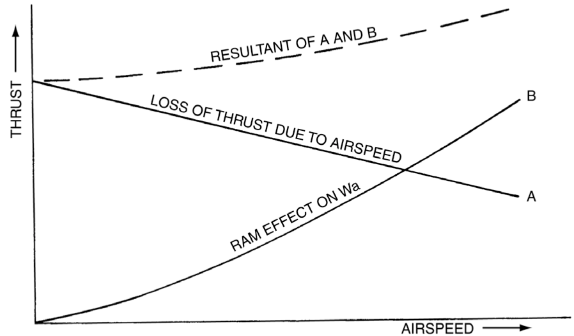
Effect of Airspeed

11. From the foregoing it appears that the net thrust of the engine must decrease as airspeed increases since, at any given altitude, V_2 is constant and V_1 is increasing. The greater the value of V_1, the lower the net thrust. This is indicated by curve A on the graph at Figure 23-1.

12. However, the ram effects at the compressor air intake increase with increasing forward speed, forcing air into the engine at greater velocity. The design of the intake converts the increased velocity of the intake air into pressure, increasing the density, thereby increasing the engine thrust. The increased thrust due to ram effect is shown in curve B of the graph at Figure 23-1.

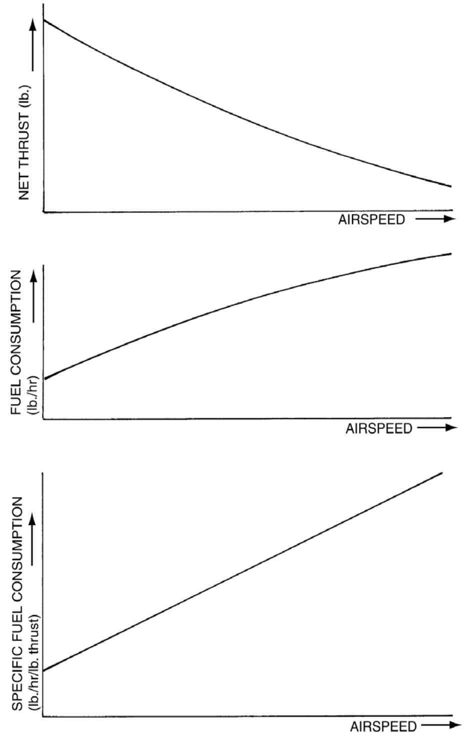
13. In terms of the thrust formula, the loss of thrust due to the reduced differential between V_2 and V_1, as airspeed increases, is largely offset by the increase in w_a due to increased density, because of ram effect. At supersonic airspeeds ram effect takes on great importance in increasing engine thrust. The overall effect of airspeed upon thrust is shown in curve C of the graph at Figure 23-1.

FIGURE 23-1
Ram Effect



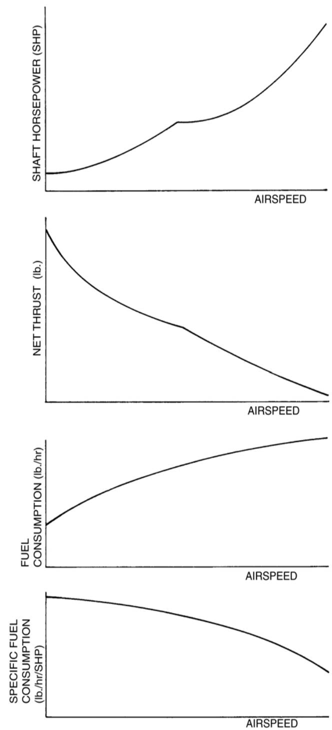
14. The increased airflow, due to ram effect, must be matched by an increased fuel flow, as discussed in the section covering fuel control systems. At subsonic airspeeds, net thrust decreases slightly with increased airspeed and fuel consumption increases as shown in the upper two graphs at Figure 23-2. From this it follows that specific fuel consumption (sfc) of a turbo-jet engine must increase with increasing airspeed, since specific fuel consumption is defined as pounds of fuel burned per hour per pound of thrust developed (lb/hr/lb.thrust). This is shown in the lower graph at Figure 23-2.

FIGURE 23-2
Effect of Airspeed
- Turbojet



15. In turbo-prop aircraft sfc is defined as lb.fuel/hr/SHP. Because SHP (shaft horsepower) increases with increased airspeed, sfc shows an increase in turbo-prop aircraft up to the maximum propeller efficiency airspeed (about 350 mph). This is illustrated at Figure 23-3.

FIGURE 23-3
Effect of Airspeed
- Turboprop



Effect of Altitude

16. As altitude increases the ambient atmospheric pressure and temperature decrease. The reduction in pressure decreases the air density but, up to the tropopause this effect is partly compensated by the reduction in temperature, which limits the density decrease with altitude. Any reduction in air density must, however, result in a reduced air mass entering the compressor and therefore a reduced mass flow through the engine. Consequently thrust of a gas turbine engine decreases with increased altitude.

17. The effect becomes most marked above the tropopause (36,089 feet) where temperature remains constant up to 65,617 feet but pressure continues to fall. With no compensating effect from temperature, air density due to pressure reduction falls at a much more rapid rate above the tropopause, with a consequent rapid decrease in thrust. It is for this reason that the optimum cruising altitude for long-range cruising is 36,000 feet.

Effect of Temperature

18. At lower levels, outside air temperature has a marked effect upon engine performance for a given altitude. This is of particular importance to take-off performance of gas turbine engines.

19. When air temperature is lower than standard the increased density will increase the mass of air entering the compressor so, for a given engine rpm the thrust is greater. However, the compressor will require greater power to maintain the same speed in the denser air so the engine will require more fuel. Alternatively, if the fuel supply is not increased, engine speed will decrease.

20. When air temperature is higher than standard the reduced density will result in a decrease in thrust for a given rpm. The engine cannot be safely operated much beyond 100% rpm so, if the mass flow for take-off thrust is to be achieved, the air density must be restored. This means that some form of thrust augmentation must be used, such as water or water-methanol injection, as discussed in section 3.38.4.

Effect of Engine RPM

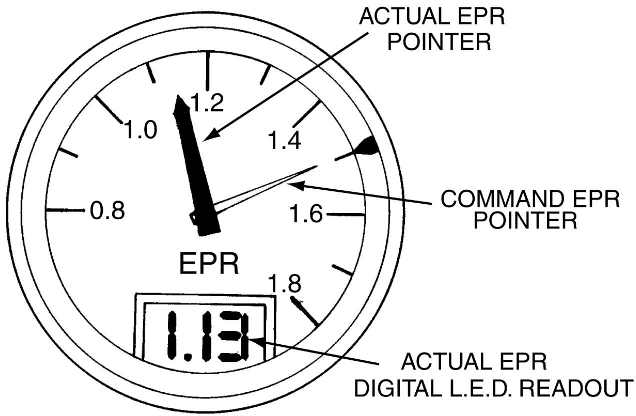
21. The air mass flow produced by the gas turbine engine compressors is dependent upon their speed of rotation, the higher the rpm the greater the mass flow. Further, the change of thrust (mass flow) for a given change in engine rpm is most marked at high engine speeds. For this reason, cruising rpm is usually 85% to 90% of maximum rpm. It is also for this reason that it is usually desirable to maintain high engine rpm during the approach to land, especially as gas turbine engines are inherently slow to 'spool up' (increase rpm)

Flat Rated Thrust

22. It is clear from the foregoing that the thrust of a gas turbine engine is directly related to the density of the air in which it is operating. Consequently, if the throttle (thrust) lever is set for a given fuel flow then any change in air density will result in a change of engine thrust. For example, if the aircraft climbs with the throttle lever set, the engine thrust will decrease with altitude because of the decreasing air density.

23. In many modern engines, with full authority electronic engine control systems, this is overcome by automatic compensation that adjusts engine speed to maintain thrust at a value demanded by the pilot's throttle lever angle. Engine thrust is displayed as engine pressure ratio (EPR) on an EPR gauge, an example of which is illustrated at Figure 23-4.

FIGURE 23-4
EPR Gauge



24. To select the required thrust (EPR), the pilot adjusts the throttle lever angle until the command EPR needle on the gauge is aligned with the reference bug set by the thrust management computer in the electronic engine control system. The control system will automatically adjust fuel flow, and thus engine speed, to maintain the required thrust with varying altitude, airspeed and inlet air temperature. In other words, thrust is maintained at a constant (flat) value regardless of changes in air density.

25. Such a system is known as a flat rated thrust system and engines equipped with it are known as flat rated engines.

Gas Turbine Operation and Monitoring

26. Control of a gas turbine engine is usually by means of a single lever, known as the throttle or, more correctly, power lever. This is used to select the desired thrust. Engine rpm is normally governed within certain limits.

27. The characteristics of the gas turbine engine differ in a number of respects from the piston engine and these affect the way in which it must be operated, both on the ground and in the air.

28. The much greater air mass flow through the engine means that there is a considerable influx of air at the intake, even when the engine is operating at idle rpm. On the ground great care must be taken to ensure that foreign objects are not drawn into the intake, since even quite small solid objects can cause significant damage to fan and compressor blades. Small nicks in these blades are sufficient to create damaging out-of-balance forces at high rpm.

29. The jet efflux from a turbo-jet or turbo-fan engine is of sufficiently high velocity to be hazardous to personnel and ground equipment to a considerable distance behind the engine. Personnel and equipment should be kept clear of the hazard areas forward of and behind the engines. It is not unknown for ground personnel to have been ingested by a running aircraft turbine engine.

30. It is important to monitor the turbine gas temperature, particularly during engine start, since the turbine nozzle guide vanes and first stage blades are highly susceptible to damage from overheating.

31. The gas turbine compressor only operates close to maximum efficiency at high rpm, typically above 85% of maximum rpm. This must be borne in mind when the engines are being operated at low rpm in flight, as a turbine engine accelerates much more slowly than a piston engine. Consequently demands for an increase in power take considerably longer to be met. The fuel scheduling system will almost certainly be 'slugged' to limit the rate at which fuel flow increases, to avoid compressor stall and surge.

32. The high air mass flow of the engine also means that it is susceptible to the ingestion of water in sufficient quantity to extinguish the flame in the combustion chambers when flying in heavy rain or snow. In these conditions continuous ignition should be selected.

33. In any aircraft, encountering a bird when in flight can be damaging to the airframe. Gas turbine compressor and fan blades are particularly prone to damage from bird ingestion and many engines have been wrecked in such encounters. Great care must be exercised when flying at low level, as during approach and take-off.

Instrumentation

34. The engine operating parameters indicated to the pilot are listed below.

Thrust. Generally indicated in the form of engine pressure ratio (EPR). Usually the pressure ratio between jet pipe pressure and LP compressor inlet pressure, although on large fan engines it is often an integration of fan discharge/turbine outlet pressure to LP compressor inlet pressure.

Torque. The power output of a turboprop engine is usually measured by a torquemeter. Power assessment is made by comparison between the torquemeter reading and a reference value.

RPM. Engine speed is measured usually as a percentage of maximum rpm. On multi-spool engines HP spool rpm (N_2) is always measured, LP spool or fan speed (N_1) is usually indicated as well.

TGT. Turbine gas temperature, also referred to as jet pipe temperature (JPT), exhaust gas temperature (EGT), turbine inlet/entry temperature (TIT or TET) is the most critical of engine temperatures. It is measured by a system of thermocouples.

Lubricating Oil Temperature and Pressure. At the high rotational speeds of gas turbines, monitoring of oil temperature and pressure is essential to safe operation.

Fuel Temperature and Pressure. Indication of the LP fuel supply is provided.

Fuel Flow. An indication of the fuel flow is given for each engine, since it provides a valuable indication of unit performance. Frequently a fuel-used indicator is included in the engine instrumentation.

Vibration Monitoring. Gas turbine engines have very low levels of vibration in normal operation so vibration is an indication of incipient potential failure. Typical causes of vibration are damaged fan, compressor or turbine blades - each of which could lead to catastrophic engine failure. Vibration monitors transmit a signal of relative amplitude of vibration within critical frequency ranges appropriate to the engine and its components.

35. The various powerplant-monitoring instruments and systems, and their operation, are dealt with in detail under Powerplant and System Monitoring Instruments, at the end of this book.

Auxiliary Power Units

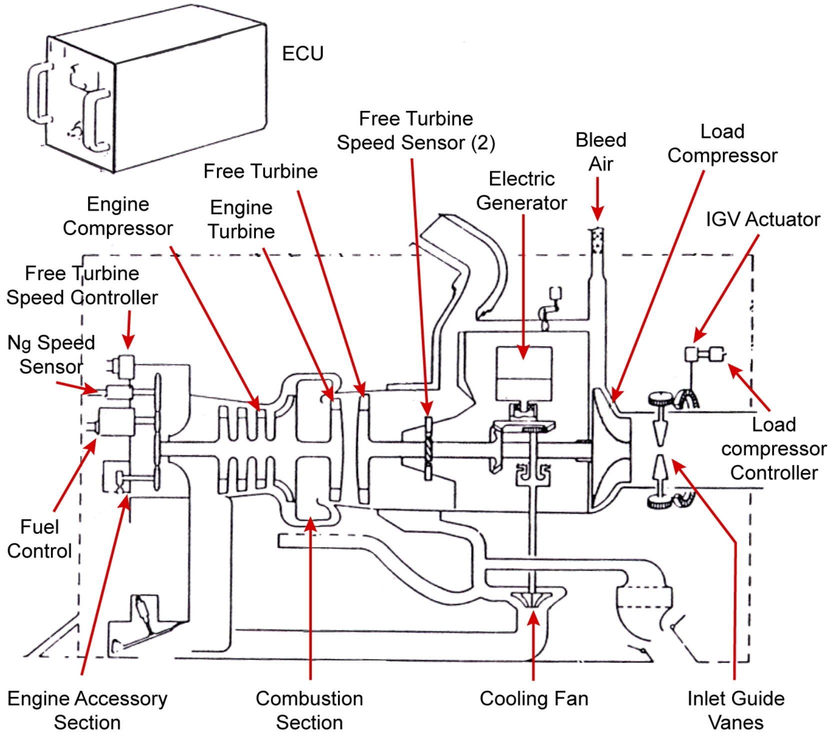
Auxiliary Power Units

1. Auxiliary power units (APU's) are fitted in the majority of transport aircraft to supply electrical power and compressed air when the aircraft is on the ground with its main engines shut down, in order to reduce the need for ground support equipment. Normally the APU is operated only when the aircraft is on the ground, but many modern transport aircraft can utilise the APU in flight as an alternative source of electrical power, if needed. Operation in flight is usually only possible within specified maximum airspeed and altitude limitations.

2. The APU is typically a self-contained unit comprising a constant speed gas turbine coupled to a gearbox, from which a generator of similar type and rating to the aircraft's main generators is driven. The APU gearbox also drives the gas turbine accessories (fuel pump, lubricating oil pump, tachometer and a centrifugal switch controlling the starter, ignition, speed governing and overspeed protection circuits).

3. A typical Auxiliary Power Unit is illustrated in section at Figure 24-1. A load compressor mounted on the same shaft as the APU power section and sharing the same air supply as the power section compressor, supplies air to the aircraft pneumatic system. The gearbox containing the generator and ancillary drives is shown at the end of the APU rotating assembly

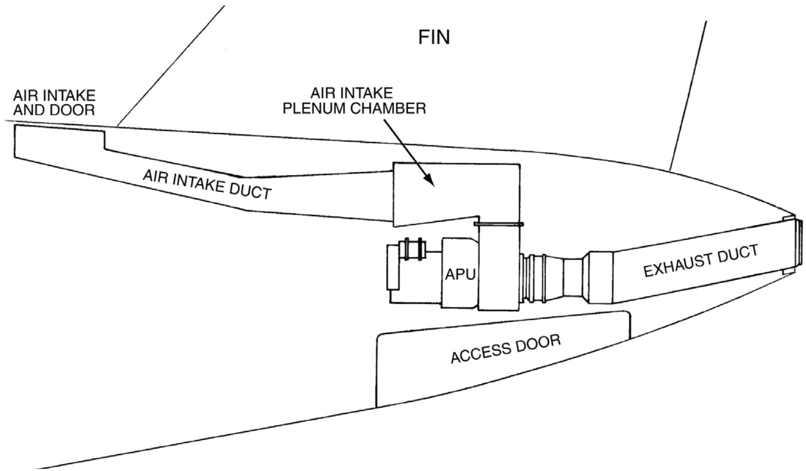
FIGURE 24-1
A Typical Auxillary Power Unit



Location

4. The APU is located in an un-pressurised compartment, usually in the rear fuselage and separated from the remainder of the aircraft by a firewall. Access to the compartment is external. Figure 24-2 shows the location of the APU in the Boeing 757.

FIGURE 24-2
Location of Auxillary Power Unit



5. Air supply for the gas turbine compressor is admitted via ducted intakes. These normally take the form of doors, which open automatically at the beginning of the start sequence and close when the APU is shut down. The APU intake door positions are often indicated at the APU control panel in the flight compartment.

6. The APU gas turbine compressor discharges to a plenum chamber, from which air for combustion is supplied to the combustion chambers of the APU gas turbine. In some units, a regulated supply of bleed air is ducted from the plenum chamber for the aircraft's cabin air conditioning and main engine air-starting systems.

Fuel System

7. The APU fuel is supplied from one of the aircraft main fuel tanks via a remotely operated (solenoid) valve. A fuel control unit (FCU) regulates fuel supply. The FCU controls acceleration during starting and maintains constant speed, under varying load conditions, when the unit is running.

Starting and Ignition

8. The APU utilises an electric starter motor, which drives through the unit gearbox and is usually powered from the aircraft batteries. In some cases a separate APU starter battery is fitted. The ignition unit is of the high-energy type and the starting sequence is basically the same for all types of APU.

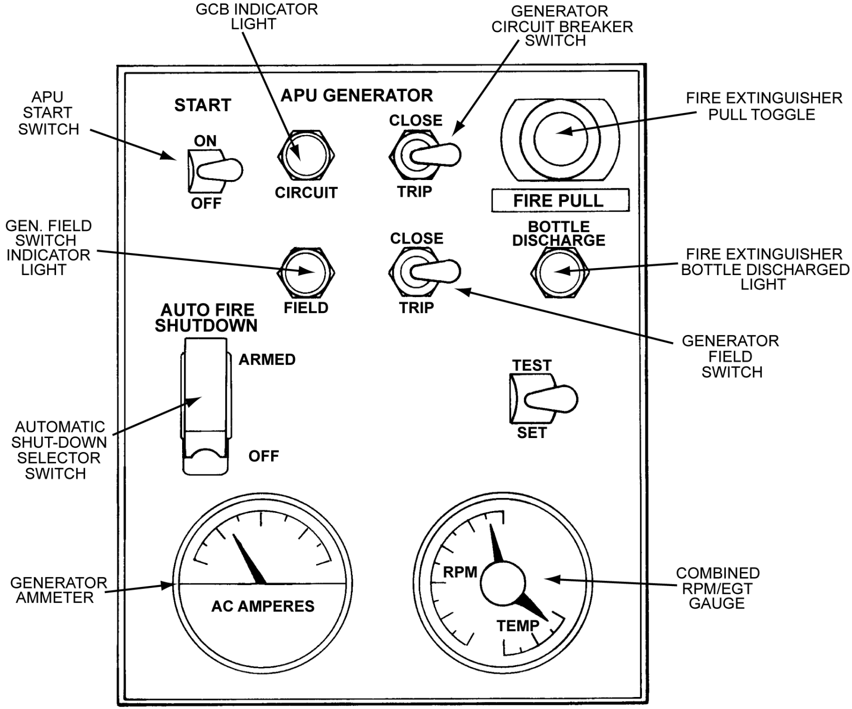
9. In some systems the APU starter motor is isolated in flight (via one of the landing gear squat switches), and in-flight starts are achieved using the windmill start principle. The in-flight start envelope for the APU will be narrower (a lower altitude and a higher airspeed being required than with a motor-assisted air-start) with this system. Furthermore, the APU inlet duct door arrangement needs to be more complex in order to provide increased ram air effect to windmill the engine up to an adequate rotational speed for the introduction of fuel and the energising of the igniter circuits.

10. However, in the event of a double generator failure (on a two-engined aircraft) the available emergency source of electrical power (the aircraft batteries) is not depleted by what may be an unsuccessful attempt to start the APU in order to provide an alternative source of generated electrical power. The only electrical power required being that involved in sequencing the APU inlet duct doors.

11. APU's are notoriously difficult to start following a prolonged period of flight at high altitude, which has resulted in well and truly cold-soaked lubricating oil.

12. The main APU control and indication panel is located in the aircraft flight deck. The starting sequence is initiated by closing the master control switch (toggle or push type). This opens the air intake doors and the starter motor then motors up the APU. The fuel supply and ignition controls are activated as and when the appropriate rotational speeds are achieved. A typical APU control panel is shown at Figure 24-3.

FIGURE 24-3
Typical Auxillary Power Unit Control Panel



13. After light up has taken place the APU engine accelerates, with the continued aid of the starter motor, typically to between 35% and 50% of governed rpm. At this point the starter motor circuit is de-energised and the engine accelerates to governed speed under its own turbine power. At about 95% governed speed the ignition circuit is cancelled by the centrifugal switch and combustion becomes self-sustaining.

14. It is good practice to allow the unit to run at no load for a minute or so, to allow all parts to reach normal working temperatures, before selecting electrical and pneumatic system air loads. An overspeed sensing system will automatically shut down the APU if governed speed is exceeded, typically when speed reaches 110% of governed speed.

15. During starting the APU must be monitored to ensure that light up and governed speed are achieved within specified time limits, and that overheating does not occur. The number of re-start attempts permissible is also specified in the aircraft operating and maintenance manuals.

Fire Protection and Cooling Systems

16. The APU compartment is fitted with its own continuous-wire fire detection system and its own single-shot fire extinguishing system. In addition to activating visual and aural warnings, the detection circuit is normally arranged so that it will automatically activate the APU shut down. The APU fire extinguisher is activated by manually operated switches on the APU control panel. In some APU's the fire extinguisher is discharged automatically when the detection circuit is activated.

17. A fan, driven from the accessories gearbox, provides cooling and ventilation of the APU compartment and cooling air for the generator and lubricating oil cooler.

Shutting Down

18. The APU is normally shut down by selecting the OFF (or STOP) position on the master control switch, after first allowing the unit to run on no-load for about two minutes to assist with even cooling of the engine.

19. Automatic shut down of the APU will typically occur as a result of any or all of the following conditions:

(a) Overspeed (typically 110% of governed rpm).

(b) Excess exhaust gas temperature (EGT).

(c) Loss of EGT signal to the control system.

(d) Low lubricating oil pressure.

(e) High lubricating oil temperature.

(f) APU compartment fire detection system operation.

(g) Excess APU bleed air outlet temperature.

(h) When specified airspeed or altitude limitations are exceeded.

20. Following an automatic shut down the master control switch should be selected OFF (or STOP).

21. Depending upon the complexity of the APU monitoring and automatic shut-down system provided, a second APU control panel may or may not be provided externally under a panel in the aircraft skin (normally adjacent to the APU itself), which will permit APU shut-down to be initiated from outside the aircraft. Associated with this external panel will be an APU fire warning klaxon. Appreciate that it is common for the APU to be left running when the flight deck is unoccupied.

22. Operating envelopes for APU's vary according to type or requirement. Some APU's have a restriction on the maximum altitude for starting, which may be governed by associated equipment limitations i.e. battery or by the starting envelope. The altitude limit may be lower than the aircraft's maximum ceiling. Whilst an APU can be operated at higher altitudes its output may be restricted ie. a generator may go from 100% output at 25,000 ft to 60% at 39,000 ft.

23. Should the demand on systems provided by the APU (pneumatic/electric or both) exceed the designed APU load, automatic control devices will be activated to protect the APU from exceeding engine operating temperature limits (EGT). Higher altitudes could be more critical. In conditions of high electrical load and air bleed the limiting function of the APU EGT will reduce the bleed air supply but permit the APU to maintain generator output.

Ram Air Turbine

24. Many commercial transport aircraft are fitted with an air-driven turbine that drives an emergency alternator, for use in the event of failure of all the engine-driven alternators.

25. A typical unit comprises a single stage turbine with direct drive to the alternator, which is usually of lower output capacity to the engine driven alternators and capable of meeting essential electrical requirements. The unit is normally stowed in a compartment closed by a hinged door.

26. When activated, the door opens and the ram air turbine is deployed into the air stream, where it is rotated by the airflow passing through the turbine blades. Variable incidence inlet guide vanes control the airflow into the turbine wheel. Constant rotary speed is maintained by variation of the inlet guide vanes under the influence of a flyweight type governor.

27. In some instances a variable pitch two-bladed propeller is used as the driving unit and certain aircraft have a ram air turbine driving an emergency hydraulic pump.

28. Typically, the unit is located in the underside of the wing root fairing and is deployed mechanically by spring action when a release catch is activated from the flight deck.

Emergency Equipment-Aeroplane

Emergency Equipment-Aeroplane

Doors and Emergency Exits

1. Statutory regulations require that all passenger-carrying aircraft must be equipped with adequate crew and passenger exits for normal and emergency situations.

Passenger Emergency Exits

2. In most large passenger transport aircraft normal exit/entry doors are situated at the front and rear of the passenger cabin. Cabin length and passenger capacity will determine the total number of doors located either side of the cabin. All doors are designed to be capable for an emergency evacuation – some aircraft may have an over-wing exit in the cabin which is usually a removable window. All doors/window are clearly marked to indicate operation for normal and emergency use.

Crew Emergency Exits

3. In the event of an emergency evacuation, the flight deck crew would use the passenger cabin exits as their primary means of escape. If circumstances prevent the flight deck crew from reaching the cabin they are provided with an alternative means of exit from the flight deck. This can take the form of either an opening side window or a hatch in the roof of the fight deck. Because of the long drop from this position each crew member will have available an inertia reel or rope system which, when grasped, will allow the crew member to descend at a controlled rate.

Accessibility

4. To ensure swift evacuation from the aircraft in an emergency situation it is paramount that the route to all emergency exits is completely clear and unimpeded. The manufacturer will design the interior fixtures and fittings to ensure a clear path that does not slow down the flow of passengers proceeding to the exit(s).

5. It is then left to the operator to maintain this and to ensure that any items brought on by passengers are not placed where they would impede the required pathways. This would mean controlling the items carried on and ensuring that they are stored in the spaces provided once on board the aircraft. The provision of an unimpeded pathway to all exits is a serious and mandatory requirement and must not be ignored.

Normal And Emergency Operation

6. There are a variety of aircraft door types, opening either inward or outward from the passenger cabin. In modern passenger aircraft the doors are of the plug type, in which the cabin differential pressure keeps the door closed and sealed. Opening of this type of door is only achieved when differential press is zero. The doors can be opened either manually or electrically from outside or inside. Manual operation is achieved by moving a handle, releasing the mechanical locks, before pushing/pulling the door open. Electrical operation is achieved by a motor supplied from the aircraft electrical system, backed up by the standby electrical system in event of normal system failure.

7. Method of operation is clearly displayed on the inside and outside of the door.

8. Some large, heavy doors have a pneumatic assist system to power the door open when operated in the auto emergency situation.

9. Emergency operation of the door is achieved by first placing the closed door into the 'arm' position. This action operates a bar which engages with lugs in the cabin floor. When the door handle is operated to open, the locked bar will trigger off the inflation mechanism of the emergency escape chute, or slide. Some slides have a dual role where they combine the use of the slide and function as a life raft. The slide will inflate and deploy to allow occupants to vacate the aircraft. If the slide inflation bottle fails to operate automatically a manual means of operation is provided. This is usually a handle or lanyard which is pulled by hand and sets off the bottle firing mechanism. If the bottle fails to function the slide cannot be used.

10. Irrespective of the position of the door arming lever, the system will be disarmed if the door is opened from the outside of the cabin. If the overwing exit is to be used a lever must be operated to unlock a removable window section (usually pulled inwards to the cabin). Exit is then through the aperture and onto the wing surface. Door slides are normally armed at commencement of taxi-out and disarmed just prior to reaching parking area.

Door Markings

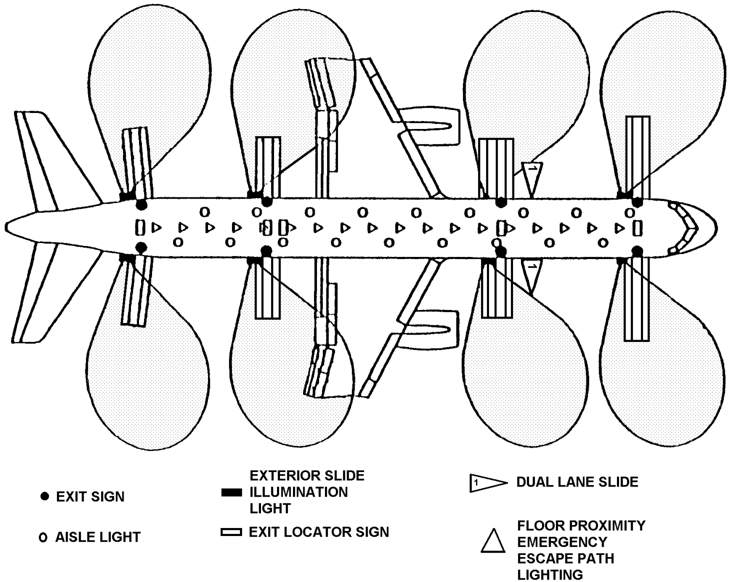
11. All doors and exit positions are clearly marked on the inside and outside to indicate operation. All doors and windows used for emergency evacuation have an exit sign adjacent to the facility. The sign may be illuminated or it may have a light reflective quality. Operating instructions for normal and emergency use are printed boldly and are accompanied with pictorial guides i.e. arrows indicating direction of handles/switches.

12. Indication that a door is in the auto armed position can be visually displayed by a protuding pin or the position of the arming lever/switch.

13. Evacuation routes must be capable of enabling all passengers to evacuate the aircraft in 90 seconds.

14. Figure 25-1 shows the location of exits, slides and emergency lighting in the B757.

FIGURE 25-1
Doors, Exits & Emergency Equipment



JAR-OPS Requirements - Doors and Emergency Exits

The JAR-OPS requirement for door and emergency exits are as follows;

(a) Each cabin must have at least one easily accessible external door

(b) Each external door must be safeguarded from opening in flight (by person or failure of a single structural element). Each door must be capable of being opened from both inside and outside, even if persons are crowded against the door on the inside of the aeroplane. Doors can be either inward or outward opening. The means of opening must be simple and obvious. The marked instructions must be easily read by day or night.

(c) Each external door must be free from jamming as a result of fuselage deformation in a minor crash.

(d) On propeller driven aeroplanes, exits must be so located that persons using them are not endangered by the propellers.

(e) Any doors, whose initial opening movement is **not inward** must have provision for direct visual inspection of its locking mechanism to determine if its fully closed and locked. A visual warning system must be provided to warn the flight crew of any door not fully closed and locked. For doors with initial opening movement, **not inward**, the indication should be totally free of any **erroneous** closed and locked indication.

(f) If an external door is not fully closed or locked it should be such that any attempt of initiation of pressurisation is impossible. It must also be shown that inadvertant opening is extremely improbable.

JAR-OPS Requirements - Doors and Emergency Exits

(g) Each passenger entry door in the side of the fuselage must be either a Type A, Type 1 or Type 2 passenger emergency exit. Each emergency exit must be capable of being opened (when there is no fuselage deformation) with either the aeroplane in the normal ground mode or one or more of the landing gears collapsed, within 10 seconds from door actuation to fully open.

Means for Emergency Evacuation

(a) An operator shall not operate an aeroplane with passenger emergency exit sill heights:

(i) Which are more than 1.83 metres (6 ft) above the ground with the aeroplane on the ground and the landing gear extended; or

(ii) Which would be more than 1.83 metres (6 ft) above the ground after the collapse of, or failure to extend of, one or more legs of the landing gear and for which a Type Certificate was first applied for on or after 1 April 2000.

unless it has equipment or devices available at each exit, where sub-paragraphs (i) or (ii) apply, to enable passengers and crew to reach the ground safely in an emergency.

(b) Such equipment or devices need not be provided at overwing exits if the designated place on the aeroplane structure at which the escape route terminates is less than 1.83 metres (6 ft) from the landing gear extended, and the flaps in the take off or landing position, whichever flap position is higher from the ground.

(c) In aeroplanes required to have a separate emergency exit for the flight crew and:

JAR-OPS Requirements - Doors and Emergency Exits

(i) For which the lowest point of the emergency exit is more than 1.83 metres (6 ft) above the ground with the landing gear extended; or

(ii) For which a Type Certificate was first applied for on or after 1 April 2000, would be more than 1.83 metres (6 ft) above the ground after the collapse of, or failure to extend of, one or more legs of the landing gear,

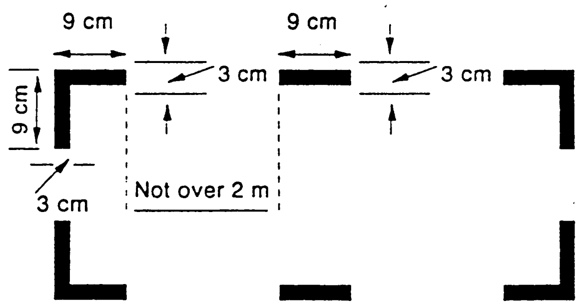
there must be a device to assist all members of the flight crew in descending to reach the ground safely in an emergency.

Cut-In Areas

15. Access to a crashed aircraft by rescue services could be fatally delayed by distorted and jammed doors. To allow for prompt access, rectangular break-in areas are marked on the outside of the fuselage by right-angled corner markings and the words 'Cut Here in Emergency' in capital letters across the centre. These indicate areas of structure which can be cut through with relative ease, to provide 'door-sized' holes in the fuselage. The size, colour and design of cut-in panels is controlled by law and is shown in Figure 25-2.

16. An operator shall ensure that, if designated areas of the fuselage suitable for break-in by rescue crews in emergency are available on an aeroplane, such areas shall be marked as shown below. The colour of the markings shall be red or yellow, and if necessary they shall be outlined in white to contrast with the background. If the corner markings are more than 2 metres apart, intermediate lines 9 cm x 3 cm shall be inserted so that there is no more than 2 metres between adjacent marks.

FIGURE 25-2
Cut-In Areas



Smoke Detection

JAR-OPS Requirement Related to Smoke Detection

(a) **Lavatory.** For aeroplanes with a passenger capacity of 26 or more.

(i) Each lavatory must be equipped with a smoke detector system that provides a warning on the flight deck or provides a visual or audible warning in the cabin that would be readily detected by the cabin crew.

(b) **Cargo compartment.** For aircraft certified with cargo compartment fire detection the following must be met for each compartment.

(i) The detection system must provide a visual indication to the flight crew within one minute after the start of a fire.

JAR-OPS Requirement Related to Smoke Detection

(ii)	The system must be capable of detecting a fire at a temperature significantly below that at which the structural integrity of the aeroplane is substantially decreased.
(iii)	There must be a means to allow the crew to test the functioning of each fire detection circuit.
(iv)	The effectiveness of the detection system must be shown for all approved operating configurations and conditions.

17. Freight holds, baggage compartments and equipment bays are fitted with smoke detectors which sample the air in the compartment and activate an alarm if certain parameters are exceeded.

18. Smoke detectors are fitted in the toilet compartments of most passenger aircraft. The detector provides an aural warning to alert the cabin crew, it is fully automatic and operates from the aircraft's 28v DC power supply. The detector unit displays a green light to indicate that power is being supplied to it and a large red display illuminates in conjunction with the aural warning when smoke is detected. A reset switch enables cancellation of the warning, but the alarm will sound again if smoke is still present. Smoke detectors are of four main types:

(a) **Photoelectric cells.** These detect the diffusion of a beam of light which occurs when the beam is interrupted by smoke. The scattering of the light increases the conductance of the cell and its output is amplified to operate a warning circuit.

(b) **Alpha particle detectors.** These are ionisation chambers which measure alpha radiation from radium. Alpha particles are absorbed by smoke, which reduces the ionisation current of the device, to operate an alarm.

(c) **Visual smoke indicators.** These are usually only fitted as alarm verification devices.

(d) **Carbon monoxide detectors.** Found mainly in aircraft of American manufacture, these devices detect concentrations of CO and activate a warning system.

19. Activation of a smoke detector is indicated on the flight deck by illuminating a master fire warning light and a red warning light for the affected area. In the case of multiple areas for detection a panel with a specific light for each area will be provided. Some installations will incorporate an aural warning in addition to the visual indications. A functional test facility will be provided to enable the crew to check the system at any time. Smoke detection systems are sometimes subject to false warnings. This can occur if the detector is exposed to contamination other than smoke in the sampled air e.g. dust, sand or impurities given off by certain categories of freight, especially in all cargo aircraft. (Live animals).

Fire Protection Systems

JAR-OPS for Fire Extinguishing Equipment

A fire extinguishing system, the quantity of the extinguishing agent, the rate of discharge and the discharge distribution must be adequate to extinguish the fire in the designated zone. For most firezones, two discharges must be provided of adequate agent concentration.

Each extinguishing agent container must have a pressure relief to prevent bursting of the container. There must be a means for each fire extinguishing agent container to indicate that the container has discharged or that the charging pressure is below the minimum necessary for operation. If any toxic agent is used, harmful concentration of liquids must be prevented from entering any personnel compartments as a result of the extinguisher discharging deliberately or by defect.

20. All gas turbine engines and their associated installation systems incorporate a fire protection system for the detection and rapid extinguishing of fire. A detection system is provided on the engine to sense an overheat condition or the occurrence of a fire and give warning on the flight deck. The detector system can consist of either a number of detector units located in strategic positions, or a continuous sensing element. The latter is discussed in greater detail at the end of this section.

21. The occurrence of a fire is indicated on the flight deck by a steady RED warning light and an alarm bell. The red light will usually indicate the location of the fire and the alarm bell can be silenced by a cut-out switch, leaving the light remaining. Each engine is covered by its own warning system and is provided with an extinguishing system, which is often doubled up, either by the use of more than one fire bottle or the optional use of another engine's fire bottle.

22. The system is controlled from the pilot's fire control panel by means of discharge switches. Figure 25-3 shows a typical schematic layout for a twin-engine aircraft fire extinguishing system. Depending on aircraft type/engine type, when the engine fire switch is pulled, certain automatic actions are triggered, i.e:

(a) Closure of the hydraulic shut-off valve

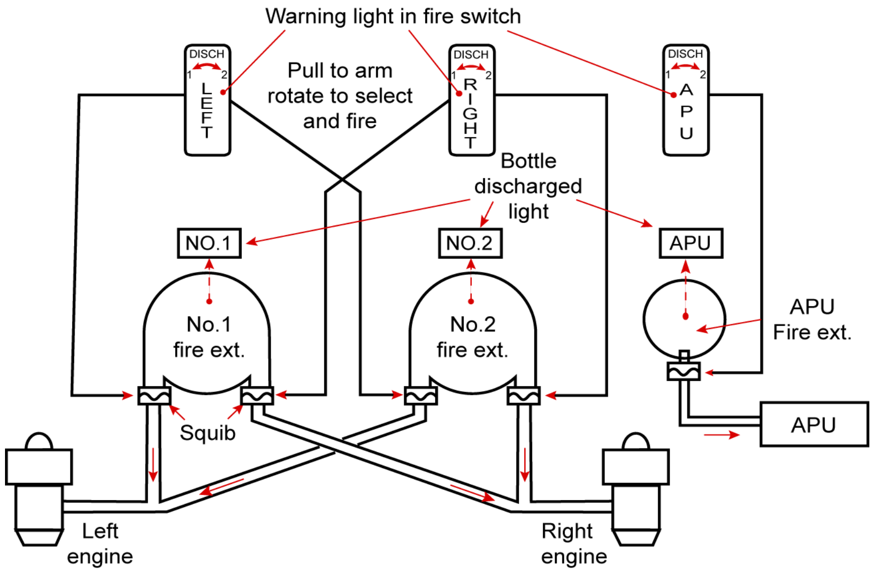
(b) Closure of the pneumatic system supply bleed air valves

(c) The fire bottle squibs are armed

(d) The generator field control relay is tripped

(e) The fuel system tank shut-off valve is closed.

FIGURE 25-3
Twin Engine Fire Extinguishing System



23. A discharge device, often called a squib, is fitted between the fire bottle and its discharge line. The squib consists of a breakable disc and a small explosive charge which is electrically detonated to break the disc and discharge the contents of the bottle.

24. Two fire extinguisher bottles are installed. Let us suppose a fire is detected in the number one engine. Pulling the LEFT fire switch arms the left engine squib in each fire bottle. Rotating the switch anti-clockwise selects the number one fire extinguisher and fires its left squib, discharging the bottle into the left engine. Subsequently rotating the switch clockwise will fire the number two fire extinguisher into the left engine. Alternatively, if the number two extinguisher is not used it remains available for use on the right engine. Pulling the RIGHT fire switch will arm the right engine bottle squibs and turning it clockwise will fire its right squib and discharge the number two bottle into the right engine.

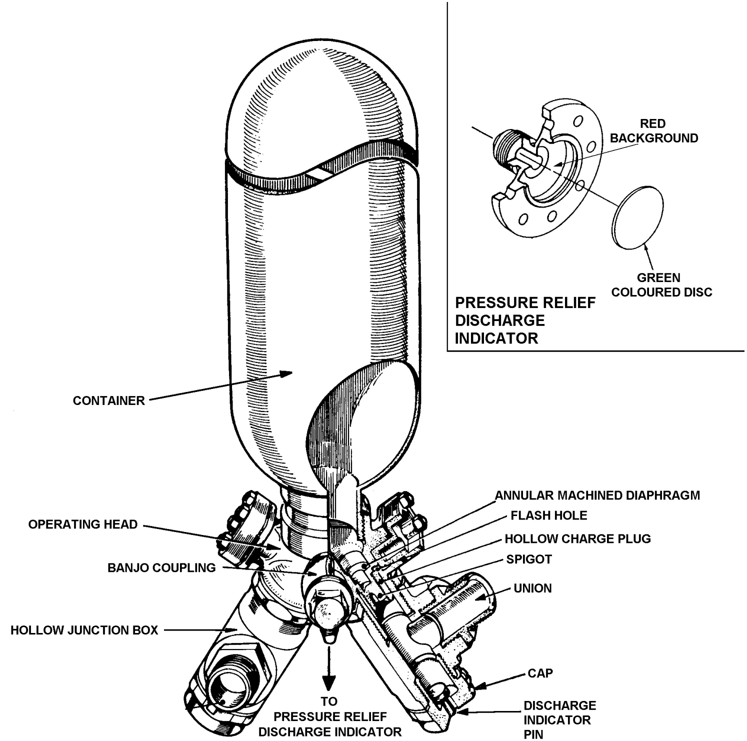
25. A single fire extinguisher bottle is provided for the APU. Pulling the APU fire switch arms the squib, which is fired by rotating the switch in either direction. It is normal to wait 30 seconds before contemplating the use of a second shot. In the case of the dual head system it must be borne in mind that the use of the second charge will deprive the pilot of fire protection for the other engine.

26. Power plants and APU's use fixed fire extinguishing installations consisting of pressurised extinguishant containers, distribution piping and operating controls. The types of extinguishant are usually toxic or semi-toxic Freon compounds such as bromochlorodiflouromethane (BCF) and bromotriflouro-methane (BTM).

27. Fixed extinguishers normally consist of a steel or copper bottle and a discharge head through which the extinguishant is discharged to the distribution system by remote operation of electrically initiated cartridge units in the discharge head. The firing circuit is controlled by switches or tee handles on the flight deck. In some aircraft types firing may be automatically initiated in the event of a crash landing. The discharge head contains a cartridge unit which, when fired, ruptures a diaphragm and allows the pressure of the extinguishant to force a plug to the end of a tube, connecting the bottle to the distribution system.

28. At the end of the discharge tube is a pin which is normally flush with the tube end cap. The movement of the plug causes the pin to protrude from the end cap once the cartridge has been fired, providing a visual indication that the extinguisher has been discharged. In some aircraft the extinguisher bottle incorporates a pressure gauge to indicate the state of charge. Figure 25-4 shows a fixed fire extinguisher.

FIGURE 25-4
Fixed Fire Extinguisher



29. In the event of a build-up of excess pressure in the bottle, due for example to overheating, protection against bursting is provided by a disc which either fuses under high temperature or bursts under excess pressure. The disc is situated in the discharge head and discharges overboard through a pressure relief line. An indicator located in a visible position, typically on the outside of the engine nacelle, will be exposed by the pressure in the relief line. The indicators are usually in the form of a coloured disc, red in most cases, but occasionally green or yellow. Alternatively, it may be arranged for the disc to be blown out when the bottle discharges due to excess pressure. With this arrangement, the absence of the disc warns of a depleted fire bottle.

30. Some aircraft incorporate electrical indicators on the flight deck to show when extinguisher cartridge units have been fired or that extinguishant pressure is present in the distribution system. These may be magnetic indicators, or warning lights.

Fire Detection Systems

JAR-OPS Requirements for a Fire Detector System

JAR-OPS states that there must be approved, quick acting fire or overheat detectors in each designated fire zone, and in the combustion, turbine, and tailpipe sections of turbine engine installations. They must be in sufficient numbers and locations ensuring prompt detection of fire in those zones. Each fire detector system must be constructed and installed so that:

(a) It will withstand vibration, inertia and other loads during operation.

(b) There is provision for the crew to be warned of any severance of a detection system where it would render the system inoperative.

(c) There is provision for the crew to be warned of any short circuit of a detection system where it would render the system inoperative.

JAR-OPS Requirements for a Fire Detector System

(d) A detector is not affected by oil, water or other fluid or fumes present in their location

(e) The crew can carry out a functional check of the system at any time.

(f) Wiring components of any detector system in the fire zone must be fire resistant.

(g) No fire or overheat detector system component may pass through another fire zone unless;

- (i) It is protected against possibilty of false warnings resulting from fires in zones through which it passes or
- (ii) Each zone involved is simultaneously protected by the same detector and extinguishing system

(h) Each fire detection system must not exceed the approved alarm actuation time.

31. On civil transport aircraft the engine compartments are usually divided into fire zones. Those in which the likelihood of fire is greatest are protected by warning and extinguisher systems. Others, such as jet pipe surrounds, may only be fitted with overheat warning systems.

32. Equipment bays and baggage compartments are usually protected by smoke detection equipment and areas adjacent to hot air ducts usually contain excess temperature detectors.

33. Auxiliary Power Units (APUs) have similar fire detection and extinguishing equipment to the main engines, but usually incorporating automatic shut-down.

34. Fire detector signals activate warning lamps and/or captions on the flight deck and often audible warnings also. Fire warning lamps conventionally give a steady red indication. All detection systems include functional test circuits and many are of a sophisticated type which monitor temperature trends in engine bays. There is one warning lamp for each engine, but the warning bell will be activated by any detection circuit.

35. Detection equipment falls into two main categories, unit and continuous types.

36. Unit type detectors usually employ either thermocouples or switches which are operated by differential expansion of metals. Unit detectors are used to monitor specific points where excessive temperatures might occur, continuous detectors are routed around a potential fire zone to provide maximum coverage.

37. Continuous wire detectors consist of a co-axial cable in which the central conducting core is insulated from the outer, earthed, sheath by a temperature sensitive material. These detectors may be of either the capacitive or resistive variety.

38. Resistive continuous detectors make use of the decrease in resistance of the insulation with increasing temperature, which will eventually allow current to flow from core to sheath and activate a warning circuit. The disadvantage of these detectors is that a short-circuit between core and sheath due to crushing or chafing will cause them to initiate a spurious fire warning.

39. Capacitive continuous detectors use the increase of capacitance which occurs with increased temperature. The increase of stored charge, and therefore discharge, with increased temperature creates a back emf and current which eventually is sufficient to activate the warning circuit. If a capacitive detector is short-circuited it may cease to act as a capacitor, but does not produce a spurious fire warning.

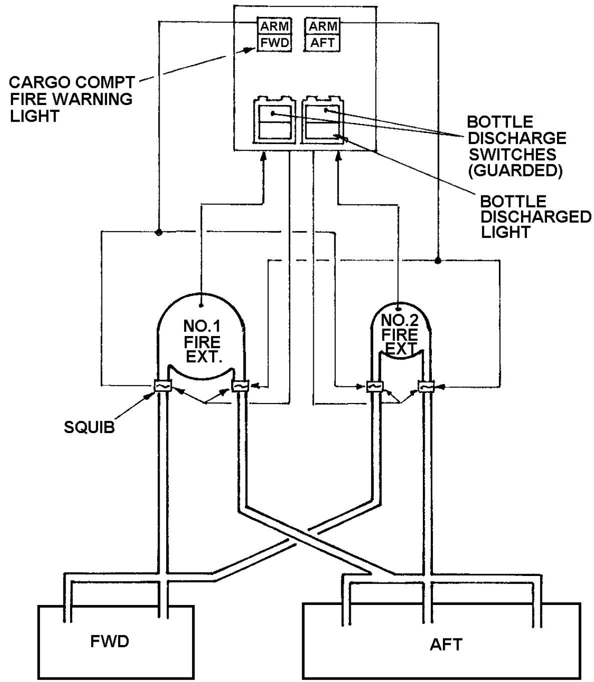
Automatic Toilet Fire Extinguishers

40. Most modern passenger transport aircraft are fitted with automatic fire extinguishers located under the wash basin in the toilet compartments. The system is designed to discharge when a pre-set temperature is exceeded. The extinguisher discharges non-toxic freon gas through one, or both, of two heat-activated nozzles. One nozzle discharges toward the towel disposal container, the other directly under the wash basin. The colour of the nozzle tips changes to aluminium when the extinguisher is discharged. A temperature indicator placard is located on the inside of an access door below each wash basin. White dots on the placard will turn black when exposed to high temperature. If an indicator has turned black or a nozzle tip has changed colour it should be assumed that the extinguisher has discharged and extinguisher and placard renewed.

Cargo Compartment Fire Extinguishing

41. In the system shown at Figure 25-5, two fire extinguisher bottles are provided for fire control of the forward and aft cargo compartments. When an overheat is detected in either compartment the associated fire warning light illuminates. Pressing the corresponding arming switch arms the selected squib (FWD or AFT) in each of the two extinguisher bottles. The number one fire extinguisher bottle is the larger of the two and is discharged first, into the compartment with the overheat warning, by pressing the number one bottle discharge switch. The number one cargo bottle discharged warning light illuminates and a CARGO BTL 1 caption is displayed to indicate low bottle pressure. The number two bottle may be manually discharged at a later stage, whereupon the discharge light and advisory message appears for the second bottle. A squib test switch is provided for checking the electrical continuity of the squib firing circuits

FIGURE 25-5
Cargo Fire Extinguishing System



CARGO COMPARTMENT FIRE EXTINGUISHING SCHEMATIC

Oxygen Systems

JAR-OPS Requirement for Aeroplane Oxygen Equipment

The oxygen system must be free from hazards in itself, in its method of operation and its effect upon other components e.g. no material should be used which, in direct contact with oxygen may give off noxious or toxic gases. Any couplings or connectors must be protected agains incorrect assembly and if required for connection, disconnection a gloved hand should be used. Where electrical connections are combined, the oxygen circuit must be complete and sealed before electrical connection is attempted. When oxygen is supplied to both crew and passengers the distribution system must be designed for either

(i) a source of supply for the flight crew on duty and a separate supply for passengers and other crew members or

(ii) a common source of supply with means to separately reserve the minimum supply required by the flight crew on duty.

(iii) Portable walk around oxygen units of continuous flow, diluter demand or straight demand may be used to meet the crew or passenger requirements

JAR-OPS Requirement for Aeroplane Oxygen Equipment

Each crew member on flight deck duty must be provided with demand equipment and a quick donning mask connected to an oxygen supply. The mask should be able to be placed on the face from its stowed position, secured, sealed and supplying oxygen with one hand within 5 seconds (without disturbing eyeglasses or causing delays in emergency duties. Normal communication duties must be able with mask donned. Portable oxygen equipment must be immediately available for each cabin attendant. There must be a means to allow the crew to determine whether oxygen is being delivered to the dispensing equipment (flow indicator). Pressure limiting devices (relief valves) and protective devices (rupture disc) should be fitted to protect the system from excessive pressure. Venting should be overboard.

JAR-OPS Equipment Standards for Oxygen Dispensing Units

If oxygen dispensing units are installed, the following apply:

(a) There must be an individual dispensing unit for each occupant for whom supplemental oxygen is to be supplied. Units must be designed to cover the nose and mouth and must be equipped with a suitable means to retain the unit in position on the face. Flight crew masks for supplemental oxygen must have provisions for the use of communication equipment.

JAR-OPS Equipment Standards for Oxygen Dispensing Units

(b) If certification for operation up to and including 25,000 ft is requested, an oxygen supply terminal, **either a supply terminal with the unit of oxygen dispensing equipment already connected or a connection which ensures that the oxygen is immediately available**, must be within easy reach of each crew member. For any other occupants the supply terminals and dispensing equipment must be located to allow use of oxygen as required by the **applicable National Operational Regulations.**

(c) **Except as specified in National Operational Regulations**, if certification for operation above 25,000 ft is requested, there must be oxygen dispensing equipment meeting the following requirements:

(i) There must be an oxygen dispensing unit connected to oxygen supply terminals immediately available to each occupant, wherever seated. If certification for operation above 30,000 ft is requested, the dispensing units providing the required oxygen flow must be automatically presented to the occupants before the cabin pressure altitude exceeds 15,000 ft and the crew must be provided with a manual means to make the dispensing units immediately available in the event of failure of the automatic system. The total number of dispensing units and outlets must exceed the number of seats by at least 10%. The extra units must be as uniformly distributed throughout the cabin as practicable.

42. Most civil transport aircraft are pressurised to maintain conditions inside the cabin equal to an altitude of approximately 8000 feet, regardless of actual aircraft altitude above this figure. Under these conditions oxygen is not normally needed for passengers and crew, but oxygen equipment is installed for emergency use in the event of pressurisation system failure.

43. In passenger aircraft without pressurisation, oxygen equipment may be installed for the use of passengers and crew when it is necessary for the aircraft to fly above 10,000 feet. Where no oxygen system is fitted, portable oxygen sets are provided, these are often also provided in large transport aircraft for therapeutic purposes and for use by cabin attendants during pressurisation emergencies.

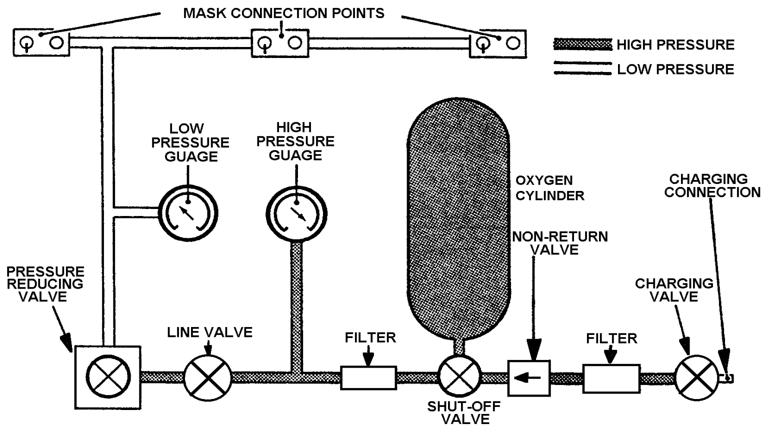
44. Gaseous oxygen systems may be of the continuous flow or diluter demand type, or a combination of the two, especially on the larger transport aircraft types. Gaseous oxygen is stored in cylinders at approximately 1800 lb/in^2. Oxygen storage cylinders are provided with an excess pressure rupture disc, fitted to the shut-off valve body and venting the cylinder contents to the outside of the aircraft in the event of a dangerous pressure rise in the cylinder. In most cases an indicator is fitted which will show that discharge has occurred due to excess cylinder pressure.

45. In many transport aircraft chemical oxygen generators are used for passenger oxygen supplies, since these obviate the need for storage of large quantities of pressurised oxygen.

Continuous Flow Oxygen Systems

46. A continuous flow oxygen system is shown at Figure 25-6. Gaseous oxygen is supplied to the pressure reducing valve from the storage cylinder and thence at a reduced pressure to the distribution system and mask connection points. These may be of the plug in or the drop out type.

FIGURE 25-6
Continuous Flow
Oxygen System



47. With plug in systems the points at which the oxygen mask supply tubes are connected to the system usually provide for 'normal' or 'high' flow rates. Some embody a selector lever for this purpose, whilst others have two alternative socket points for each mask connection, one giving normal flow and the other high flow. The socket points incorporate self-closing shut-off valves which are spring-loaded to the closed position and are opened by plugging in the mask connecting tube.

48. With drop out systems the mask is automatically ejected from the stowed position to the "half hang" position by an aneroid controlled release mechanism, activated when cabin altitude exceeds about 14,000 feet, and over-ridden, if necessary, by a switch on the flight deck. When the mask is further pulled towards the passenger's face a lanyard-operated valve opens to supply oxygen to the mask.

49. A simpler version of the continuous flow system is often used in light aircraft which only carry a pilot and perhaps five passengers. In this type of system the functions of shut-off valve and pressure regulator are combined and mask connections are of the plug-in variety.

Diluter Demand Oxygen Systems

50. A diluter demand oxygen system is illustrated at Figure 25-7. Oxygen is diluted with air and the mixture is supplied as demanded by the user's respiration cycle, that is to say only when the user inhales. Diluter demand systems are provided only for crew use and are additional to the passenger oxygen system. There is a mask connection point for each crew member.

51. The diluter demand regulator incorporates a four-position control lever, which controls the oxygen flow to the crew masks as follows:

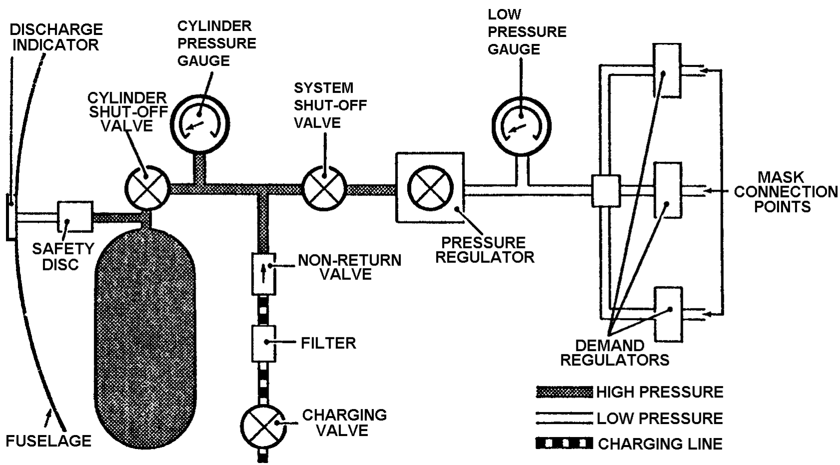
Normal oxygen. Diluted oxygen is supplied to the crew member's mask when the user inhales. The proportion of oxygen in the mixture is automatically increased as cabin altitude increases until, at a cabin altitude of 32,000 feet, approximately 100% oxygen is supplied.

100% oxygen. The regulator air valve is closed and 100% oxygen is supplied when the user inhales, regardless of cabin altitude.

Emergency. Used to provide protection against smoke or other noxious fumes in the cabin atmosphere. When selected, a flow of 100% oxygen is supplied at a pressure greater than cabin pressure, to prevent any leakage into the mask. In some systems the flow is continuous (as opposed to demand) when the emergency selection is made.

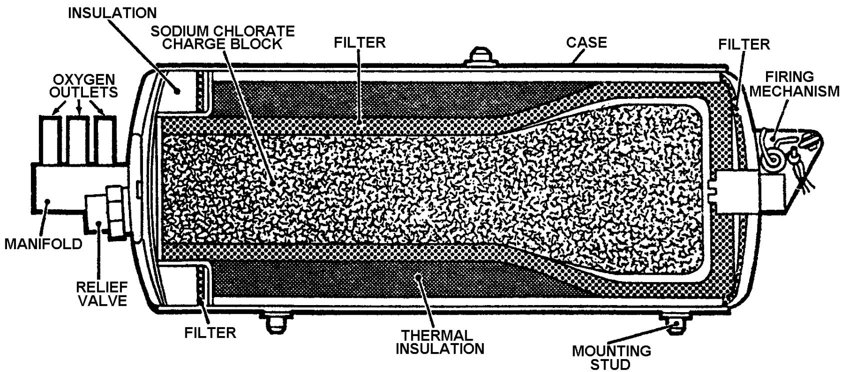
Test mask. Oxygen is supplied at a higher pressure than that when the emergency position is selected. This is used to test the mask for facial fit and to test the mask and equipment for leakage.

FIGURE 25-7
Diluter Demand Oxygen System



Chemical Oxygen Generators

FIGURE 25-8
Chemical Oxygen Generator



52. Figure 25-8 shows a diagram of a chemical oxygen generator.

53. The unit consists of (typically) three drop out masks barometrically ejected when cabin altitude exceeds a preset value (usually about 14,000 feet). When a mask is pulled towards the user a lanyard trips the electrical firing mechanism, which ignites a sodium chlorate and iron powder charge block. As the temperature of the block rises, a chemical reaction creates a flow of low pressure oxygen through a filter to the mask. Oxygen flow will normally be maintained for about 15 minutes and, despite the very high temperatures generated, the oxygen itself is at a comfortable temperature. In some sets the electrical firing circuit is initiated mechanically. Chemical oxygen generators have a shelf/service life of ten years.

Masks

54. Crew masks fit snugly to the user's face, with minimum leakage around the mask, and incorporate a microphone and jack-plug for connection to the aircraft communication system. Passenger masks are usually cup-shaped rubber mouldings which deploy automatically (drop-out masks). They may be provided with a simple elastic head strap, or may need to be physically held in place. In passenger systems which do not embody automatic deployment the masks may be plastic bags with a head strap.

Indications

55. Storage cylinder and supply line pressures are indicated, usually on a cockpit overhead panel. Cylinder pressures are usually displayed on pressure gauges, line pressure warning lights illuminate when a system is in use. In some systems cylinder contents are displayed instead of cylinder pressure.

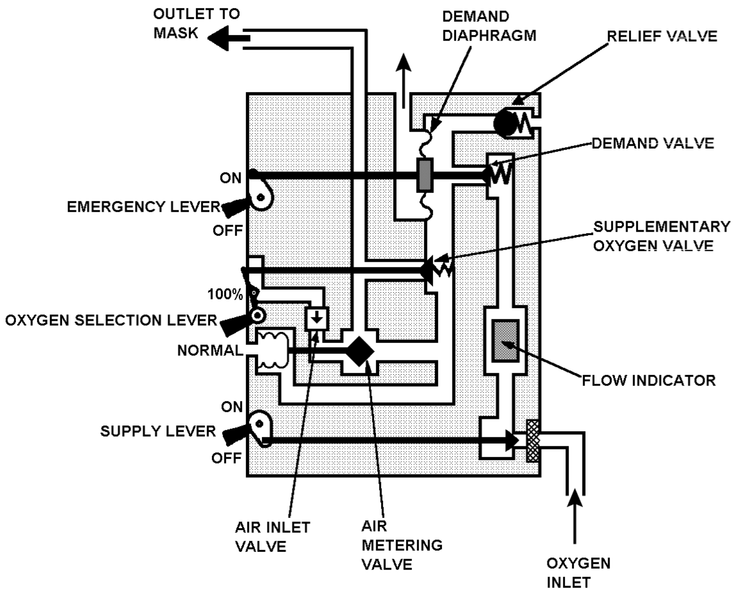
Cylinder Charge Pressure

56. Oxygen cylinders are charged to a nominal 1800 lb/in^2 at 21°C. At higher or lower temperatures the cylinder pressure will be higher or lower. A charging table is provided, giving the pressure to which the cylinder must be charged at ambient temperature.

Diluter Demand Regulator

57. Figure 25-9 shows a schematic layout of a narrow panel system diluter demand oxygen regulator.

FIGURE 25-9
Diluter Demand Regulator



Normal Operation

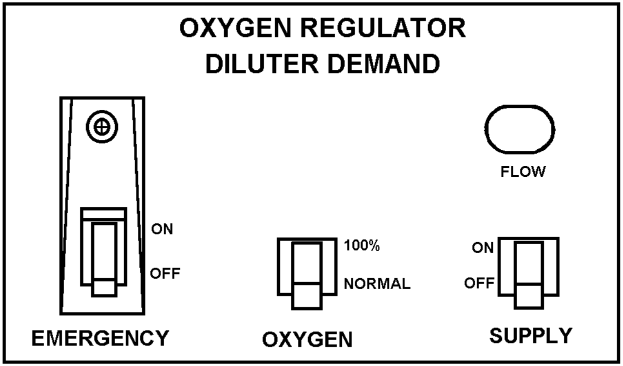
For normal operation the Emergency Lever is set to OFF, the Oxygen Selection Lever to NORMAL, and the Supply Lever to ON. When the user inhales this creates a pressure differential across the demand diaphragm, causing the demand valve to open partially, and admit oxygen from the open supply valve to mix with air from the air inlet valve. The aneroid-controlled air metering valve determines the oxygen/air ratio of the mixture. As altitude is increased the reducing barometric pressure causes the aneroid capsule to contract, reducing the air supply and increasing the oxygen supply to the outlet. The air inlet valve is a non-return valve, set to open when the user inhales and creates a pressure differential across it.

100% Oxygen. With the Oxygen Selection Lever set to 100% the air inlet port is closed and the supplementary oxygen valve is held open, connecting undiluted oxygen to the outlet whenever the user inhales.

Emergency. With the Emergency Lever set to the ON position, the demand valve is held open and the demand diaphragm is mechanically loaded and no longer responds to differential pressure when the user inhales. Undiluted oxygen flows continuously to the outlet and oxygen supply pressure holds the air inlet valve closed.

58. Figure 25-10 shows a typical control panel for a narrow panel oxygen regulator.

FIGURE 25-10
Oxygen Regulator Panel



59. The flow indicator is in the form of a float suspended in the oxygen flow and visible through a sight window. When no oxygen is flowing the float drops out of view.

Emergency Equipment

Portable Oxygen Sets

60. These consist of a cylinder, charged to 1800 ib/in^2 and containing, usually, 120 litres of oxygen. They are provided for use by cabin crew in dealing with depressurisation and paramedical incidents. The cylinder is contained in a fabric carrying bag, with straps so that it may be worn by the bearer, and a mask connected to the cylinder regulating valve by a flexible hose.

61. It is usually possible to select 'normal', 'high' and (in some cases) 'emergency' flow rates from these sets, corresponding to 2, 4 and 10 litres per minute. The endurance of a 120 litre portable oxygen set at these flow rates would therefore be 60, 30 and 12 minutes respectively.

Portable Fire Extinguishers

62. Portable extinguishers in aircraft use either CO_2 WATER or BCF as the extinguishant. The type of extinguisher chosen for a particular location will contain an extinguishant suitable for the type of fire to be expected in that compartment. Extinguishers should be easily accessible and are therefore retained in brackets by quick-release fittings.

63. CO_2 extinguishers are particularly suitable for electrical fires. The released CO_2, gas excludes two of the three components required for combustion, namely heat and oxygen (the third being fuel). Because of its rapid cooling effect, CO_2 can cause damage if used on an engine fire.

64. BCF extinguishers are suitable for all types of fires and are therefore widely used in aircraft. The disadvantages of BCF are that, once heated by the fire that it is tackling, it gives off noxious gases. Also, BCF does not cool the fire-affected area, and it is therefore necessary to employ a follow-up action to cool the area and prevent re-ignition of the fire, either with a water extinguisher or a convenient coffeepot (as we know, airline coffee is invariably only luke warm).

65. WATER extinguishers are suitable only for use on dry materials (furnishings, paper, wood and so on). They should never be used on electrical equipment fires, since the water jet can conduct electricity and may lead to electrocution of the fire-fighter. Water is also unsuitable for use on metal fires, since the high temperature of burning metal may cause the water to break down into hydrogen (a fuel) and oxygen. Liquid fuel fires are another type of fire where water extinguishers are unsuitable, since the burning fuel could well float on the water and result in the spread of the fire.

JAR Requirements

JAR–OPS 1.790 Hand fire Extinguishers

An operator shall not operate an aeroplane unless hand fire extinguishers are provided for use in crew, passenger and, as applicable, cargo compartments and galleys in accordance with the following:

(a) The type and quantity of extinguishing agent must be suitable for the kinds of fires likely to occur in the compartment where the extinguisher is intended to be used and, for personnel compartments, must minimise the hazards of toxic gas concentration;

(b) At least one hand fire extinguisher, containing Halon 1211 (bromochlorodifluoromethane, $CBrClF_2$), or equivalent as the extinguishing agent, must be conveniently located on the flight deck for use by the flight crew;

(c) At least one hand fire extinguisher must be located in, or readily accessible for use in, each galley not located on the main passenger deck;

(d) At least one readily accessible hand fire extinguisher must be available for use in each Class A or Class B cargo or baggage compartment and in each Class E cargo compartment that is accessible to crew members in flight; and

JAR–OPS 1.790 Hand fire Extinguishers

(e) At least the following number of hand fire extinguishers must be conveniently located in the passenger compartment(s):

Passenger Capacity	Number of Extinguishers
7 to 30	1
31 to 60	2
61 to 200	3
201 to 300	4
301 to 400	5
401 to 500	6
501 to 600	7
601 to 700	8

(f) When two or more extinguishers are required, they must be evenly distributed in the passenger compartment.

(g) At least one of the required fire extinguishers located in the passenger compartment of an aeroplane with a maximum approved passenger seating configuration of at least 31, and not more than 60, and at least two of the fire extinguishers located in the passenger compartment of an aeroplane with a maximum approved passenger seating configuration of 61 or more must contain Halon 1211 (bromochlorodifluoromethane, $CBrClF_2$), or equivalent as the extinguishing agent.

AMC Requirements

AMC OPS 1.790 Hand Fire Extinguishers

(a) The number and location of hand fire extinguishers should be such as to provide adequate availability for use, account being taken of the number and size of the passenger compartments, the need to minimise the hazard of toxic gas concentrations and the location of toilets, galleys etc.

(b) These considerations may result in the number being greater than the minimum prescribed.

(c) There should be at least one fire extinguisher suitable for both flammable fluid and electrical equipment fires installed on the flight deck. Additional extinguishers may be required for the protection of other compartments accessible to the crew in flight. Dry chemical fire extinguishers should not be used on the flight deck, or in any compartment not separated by a partition from the flight deck, because of the adverse effect on vision during discharge and, if non-conductive interference with electrical contacts by the chemical residues.

(d) Where only one hand fire extinguisher is required in the passenger compartments it should be located near the cabin crew member's station, where provided.

(e) Where two or more hand fire extinguishers are required in the passenger compartments and their location is not otherwise dictated by consideration of paragraph 1 above, an extinguisher should be located near each end of the cabin with the remainder distributed throughout the cabin as evenly as is practicable.

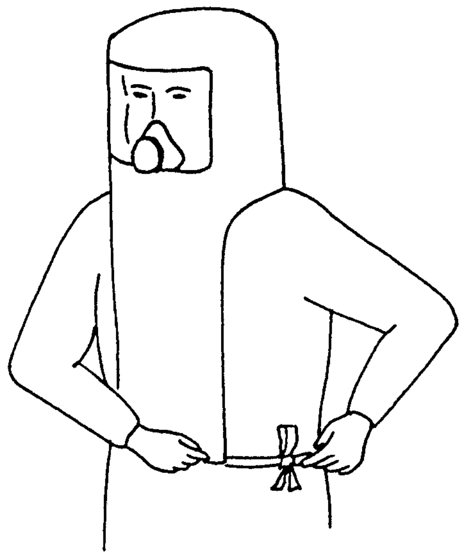
(f) Unless an extinguisher is clearly visible, its location should be indicated by a placard or sign. Appropriate symbols may be used to supplement such a placard or sign.

Smoke Hoods

66. The description which follows is of the Drager Oxycrew smoke hood as provided in many aircraft. It is reasonably representative of most smoke hoods in current usage and is a light-weight portable breathing device, designed to provide the wearer with respiratory and eye protection in oxygen deficient, smoke-filled or toxic atmospheres.

67. A Drager mask is illustrated at Figure 25-11 and comprises a single size hood which completely covers the head. An apron extending down over the chest contains a life support pack in the form of a chemical oxygen canister filled with potassium superoxide. This chemical reacts with water vapour and carbon dioxide in the wearer's exhaled air, to produce pure oxygen. The chemical action can be initiated by simply breathing into the mask, but this takes a few seconds to start and these may not be available in an emergency situation. Consequently a quick-start toggle is located at the bottom of the apron. When pulled this immediately provides oxygen for the wearer until the potassium superoxide reaction commences.

FIGURE 25-11
Smoke Hood



JAR Requirements

JAR-OPS 1.780 Crew Protective Breathing Equipment

An operator shall not operate a pressurised aeroplane or, after 1 April 2000, an unpressurised aeroplane with a maximum certificated take-off mass exceeding 5700 kg or having a maximum approved seating configuration of more than 19 seats unless:

(a) It has equipment to protect the eyes, nose and mouth of each flight crew member while on flight deck duty and to provide oxygen for a period of not less than 15 minutes. The supply for Protective Breathing Equipment (PBE) may be provided by the supplemental oxygen required by JAR-OPS 1.770(b)(1) or JAR-OPS 1.775 (b)(1). In addition, when the flight crew is more than one and a cabin crew member is not carried, portable PBE must be carried to protect the eyes, nose and mouth of one member of the flight crew and to provide breathing gas for a period of not less than 15 minutes; and

(b) It has sufficient portable PBE to protect the eyes, nose and mouth of all required cabin crew members and to provide breathing gas for a period of not less than 15 minutes.

PBE intended for flight crew use must be conveniently located on the flight deck and be easily accessible for immediate use by each required flight crew member at their assigned duty station.

PBE intended for cabin crew use must be installed adjacent to each required cabin crew member duty station.

JAR-OPS 1.780 Crew Protective Breathing Equipment

An additional, easily accessible portable PBE must be provided and located at or adjacent to the hand fire extinguishers required by JAR-OPS 1.790 (c) and (d) except that, where the fire extinguisher is located inside a cargo compartment, the PBE must be stowed outside but adjacent to the entrance to that compartment.

PBE while in use must not prevent communication where required by JAR-OPS 1.685, JAR-OPS 1.690, JAR-OPS 1.810 and JAR-OPS 1.850.

Emergency Locator Beacon/Transmitter (ELT)

JAR Requirements

JAR-OPS 1.820 Automatic Emergency Locator Transmitter

An operator shall not operate an aeroplane unless it is equipped with an automatic Emergency Locator Transmitter (ELT) attached to the aeroplane in such a manner that, in the event of a crash, the probability of the ELT transmitting a detectable signal is maximised and the possibility of the ELT transmitting at any other time is minimised.

An operator must ensure that the ELT is capable of transmitting on the distress frequencies

121. 5 MHz and 243 MHz for a period of 48 hrs at an operating temperature of -20ºC.

IEM Requirements

IEM OPS 1.820 Automatic Emergency Locator Transmitter

Types of automatic Emergency Locator Transmitters are defined as follows:

(a) **Automatic Fixed (ELT (AF)).** This type of ELT is intended to be permanently attached to the aeroplane before and after a crash and is designed to aid SAR teams in locating a crash site;

(b) **Automatic Portable (ELT (AP)).** This type of ELT is intended to be rigidly attached to the aeroplane before a crash, but readily removable from the aeroplane after a crash. It functions as an ELT during the crash sequence. If the ELT does not employ an integral antenna, the aircraft-mounted antenna may be disconnected and an auxiliary antenna (stored on the ELT case) attached to the ELT. The ELT can be tethered to a survivor or a lift-raft. This type of ELT is intended to aide SAR teams in locating the crash site or survivor(s);

(c) **Automatic Deployable (ELT (AD)).** This type of ELT is intended to be rigidly attached to the aeroplane before the crash and automatically ejected and deployed after the crash sensor has determined that a crash has occurred. This type of ELT should float in water and is intended to aid SAR teams in locating the crash site.

To minimise the possibility of damage in the event of crash impact, the Automatic Emergency Locator Transmitter should be rigidly fixed to the aeroplane structure as far aft as practicable with its antenna and connections so arranged as to maximise the probability of the signal being radiated after a crash.

Life Jackets

68. Passenger lift jackets are normally stored beneath the passenger seats and crew lift jackets in easily accessible locations. Although there are several types of lift jacket in general use, they are all basically similar. Inflation is by means of CO_2, stored under pressure in a small cylinder incorporated in the life jacket and activated manually by a pull-cord. A stand-by mouth-operated inflation valve is provided for emergency inflation and to maintain inflation over extended periods. If the CO_2 inflation is inadvertently activated the life jacket can be deflated. Subsequent re-inflation can only be effected through mouth-operation. To aid in rescue operations, life jackets are equipped with a battery-operated light and a mouth-operated whistle. The battery is activated by contact with water. Some types of life jacket also incorporate fluorescent dye marker and shark repellent compounds, which stain the surrounding water when immersed. Passenger life jackets are generally coloured yellow and crew life jackets are coloured dayglow.

JAR Requirements

JAR-OPS 1.825 Life Jackets

Land aeroplanes. An operator shall not operate a land aeroplane: When flying over water and at a distance of more than 50 nautical miles from the shore; or

When taking off or landing at an aerodrome where the take-off or approach path is so disposed over water that in the event of a mishap there would be likelihood of a ditching, unless it is equipped with life jackets equipped with a survivor locator light, for each person on board. Each life jacket must be stowed in a position easily accessible from the seat or berth of the person for whose use it is provided. Life jackets for infants may be substituted by other approved flotation devices equipped with a survivor locator light.

JAR-OPS 1.825 Life Jackets

> **Seaplanes and amphibians.** An operator shall not operate a seaplane or an amphibian on water unless it is equipped with life jackets equipped with a survivor locator light, for each person on board. Each life jacket must be stowed in a position easily accessible from the seat or berth of the person for whose use it is provided. Life jackets for infants may be substituted by other approved flotation devices equipped with a survivor locator light.

IEM Requirements

IEM OPS 1.825 Life Jackets

> For the purpose of JAR-OPS 1.825, seat cushions are not considered to be flotation devices.

Life Rafts

Jar Requirements

JAR-OPS 1.830 Life-rafts and Survival ELTs for Extended Overwater Flights

> On overwater flights, an operator shall not operate an aeroplane at a distance away from land, which is suitable for making an emergency landing, greater than that corresponding to:
>
> (a) 120 minutes at cruising speed or 400 nautical miles, whichever is the lesser, for aeroplanes capable of continuing the flight to an aerodrome with the critical power unit(s) becoming inoperative at any point along the route or planned diversions; or

JAR-OPS 1.830 Life-rafts and Survival ELTs for Extended Overwater Flights

(b) 30 minutes at cruising speed or 100 nautical miles, whichever is the lesser, for all other aeroplanes,

(c) unless the equipment specified in sub-paragraphs below is carried.

Sufficient life-rafts to carry all persons on board. Unless excess rafts of enough capacity are provided, the buoyancy and seating capacity beyond the rated capacity of the rafts must accommodate all occupants of the aeroplane in the event of a loss of one raft of the largest rated capacity. The life-rafts shall be equipped with:

(a) A survivor locator light; and

(b) Life saving equipment including means of sustaining life as appropriate to the flight to be undertaken (see AMC OPS 1.830 (b) (2); and

(c) At least two survival Emergency Locator Transmitters (ELT). (See IEM OPS 1.820.)

AMC Requirements

AMC OPS 1.830 (b) (2) Life-rafts and ELT for Extended Overwater Flights

The following should be included in each life-raft:

(a) Means for maintaining buoyancy;

(b) A sea anchor;

(c) Life-lines, and means of attaching one life-raft to another;

(d) Paddles for life-rafts with a capacity of 6 or less;

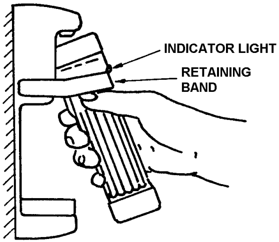
AMC OPS 1.830 (b) (2) Life-rafts and ELT for Extended Overwater Flights

(e)	Means of protecting the occupants from the elements;
(f)	A water resistant torch;
(g)	Signalling equipment to make the pyrotechnical distress signals described in ICAO Annex 2;
(h)	For each 4, or fraction of 4, persons which the life-raft is designed to carry:
(i)	100 g glucose tablets;
(j)	First-aid equipment
(k)	The above three items, inclusive, should be contained in a pack.

Emergency Torches

69. The emergency torches are self-powered, they are automatically switched ON when removed from their stowage and OFF when refitted. The state of charge of the torch battery is indicated by a red light on the torch body which flashes every 3 to 4 seconds as long as the battery has an adequate reserve of power. The rate of flashing becomes slower as the battery power diminishes and when it reaches one flash every 10 seconds the battery is in need of replacement. The red light also monitors the bulb filament, the red light will cease to flash when the filament fails. Once removed from its stowage, refitment of an emergency torch is a maintenance procedure, since the wall bracket retaining band which holds the torch in place must be broken to remove the torch. A typical emergency torch stowage is illustrated at Figure 25-12.

FIGURE 25-12
Emergency Torch



Emergency Lighting

70. Emergency lighting and exit signs must operate when all of the main electrical circuits on the aircraft are rendered inoperative. Cabin emergency and escape path lighting will illuminate automatically and will be supplied from the battery bus or an essential/vital/standby bus or from its own dedicated battery. The lighting must be capable of illumination for a minimum of 10 minutes.

JAR Requirements

JAR-OPS 1.815 Emergency Lighting

An operator shall not operate a passenger carrying aeroplane which has a maximum approved passenger seating configuration of more than 9 unless it is provided with an emergency lighting system having an independent power supply to facilitate the evacuation of the aeroplane. The emergency lighting system must include:

For aeroplanes which have a maximum approved passenger seating configuration of more than 19:

(a) Sources of general cabin illumination;

(b) Internal lighting in floor level emergency exit areas; and

(c) Illuminated emergency exit marking and locating signs.

For aeroplanes for which the application for the type certificate or equivalent was filed in a JAA Member State or elsewhere before 1 May 1972, and when flying by night, exterior emergency lighting at all overwing exits, and at exits where descent assist means are required.

For aeroplanes for which the application for the type certificate or equivalent was filed in a JAA Member State or elsewhere on or after 1 May 1972, and when flying by night, exterior emergency lighting at all passenger emergency exits.

For aeroplanes for which the type certificate was first issued in a JAA Member State or elsewhere on or after 1 January 1958, floor proximity emergency escape path marking system in the passenger compartment(s).

For aeroplanes which have a maximum approved passenger seating configuration of 19 or less and are certificated to JAR –23 or JAR-25:

JAR-OPS 1.815 Emergency Lighting

(a) Sources of general cabin illumination;

(b) Internal lighting in emergency exit areas; and

(c) Illuminated emergency exit marking and locating signs.

For aeroplanes which have a maximum approved passenger seating configuration of 19 or less and are not certificated to JAR-23 or JAR-25, sources of general cabin illumination.

After 1 April 1998 an operator shall not, by night, operate a passenger carrying aeroplane which has a maximum approved passenger seating configuration of 9 or less unless it is provided with a source of general cabin illumination to facilitate the evacuation of the aeroplane. The system may use dome lights or other sources of illumination already fitted on the aeroplane and which are capable of remaining operative after the aeroplane's battery has been switched off.

Megaphone

JAR Requirements

JAR-OPS 1.810 Megaphones

An operator shall not operate an aeroplane with a maximum approved passenger seating configuration of more than 60 and carrying one or more passengers unless it is equipped with portable battery-powered megaphones readily accessible for use by crew members during an emergency evacuation, to the following scales:

JAR-OPS 1.810 Megaphones

(a) For each passenger deck:

Passenger Seating Configuration	Number of Megaphones Required
61 - 99	1
100 or more	2

For aeroplanes with more than one passenger deck, in all cases when the total passenger seating configuration is more than 60, at least 1 megaphone is required.

AMC Requirements

AMC OPS 1.810 Megaphones

Where one megaphone is required, it should be readily accessible from a cabin crew member's assigned seat. Where two or more megaphones are required, they should be suitably distributed in the passenger cabin(s) and readily accessible to crew members assigned to direct emergency evacuations. This does not necessarily require megaphones to be positioned such that they can be reached by a crew member when strapped in a cabin crew member's seat.

Crash Axe

JAR Requirements

JAR-OPS 1.795 Crash Axes and Crowbars

> An operator shall not operate an aeroplane with a maximum certificated take-off mass exceeding 5700 kg or having a maximum approved passenger seating configuration of more than 9 seats unless it is equipped with at least one crash axe or crowbar located on the flight deck. If the maximum approved passenger seating configuration is more than 200 an additional crash axe or crowbar must be carried and located in or near the most rearward galley area.
>
> Crash axes and crowbars located in the passenger compartment must not be visible to passengers.

Fireproof Gloves

71. Most commercial aircraft carry at least one set of fireproof gloves. Current designs are made in Kevlar. Such gloves are normally located alongside the portable fire extinguishers.

Emergency Equipment – Typical Inventory

72. The table at Figure 25-13 shown the quantity of emergency equipment typically fitted in a Boeing 757 aircraft. The slide rafts will typically support about 25 persons and the life rafts about 40 persons. Emergency locator beacons are usually stowed adjacent to cabin attendants' door seats. Additional cabin emergency equipment consists of jemmies (stowed in the cockpit), torches and megaphones for use by cabin staff.

FIGURE 25-13
Emergency Equipment B757

Cabin Emergency Equipment

8	Doors
6	Inflatable Slides
2	Inflatable Slides - Dual Lane
8	BCF Fire Extinguishers
8	Kelvar Gloves
14	Portable Oxygen Sets
8	Smoke Hoods
228	Passenger Life Jackets
8	Crew Life Jackets
23	Infant Life Jackets
23	Child Seat - Belts
8	Emergency Torches
3	First Aid Kits
3	Megaphones
4	Infant Flotation Cots
4	Life Rafts (EROPS Aircraft)
2	Sabre Emergency Location Beacons (EROPS)

4	Slide Rafts
4	Survival Packs

Flight Deck Emergency Equipment

2	BCF Fire Extinguishers
4	Smoke Goggles
1	Crew Oxygen Set with Full Face Mask
1	Jemmy
2	Windows with Escape Straps
4	Crew Life Jackets
4	Emergency Torches
1	Kelvar Gloves

First-Aid Kits

The following should be included in the First-Aid Kits:	
	Bandages (unspecified)
	Burns dressings (unspecified)
	Wound dressings, large and small

	Adhesive tape, safety pins and scissors
	Small adhesive dressings
	Antiseptic wound cleaner
	Adhesive wound closures
	Adhesive tape
	Disposable resuscitation aid
	Simple analgesic e.g. paracetamol
	Antiemetic e.g. cinnarizine
	Nasal decongestant
	First-Aid handbook
	Splints, suitable for upper and lower limbs
	Gastrointestinal Antacid +
	Anti-diarrhoeal medication e.g. Loperamide +
	Ground/Air visual signal code for use by survivors
	Disposable Gloves

A list of contents in at least 2 languages (English and one other). This should include information on the effects and side effects of drugs carried.

NOTE: An eye irrigator whilst not required to be carried in the first-aid kit should, where possible, be available for use on the ground.

+ For aeroplanes with more than 9 passenger seats installed.

Emergency Medical-Kit

The following should be included in the emergency medical kit carried in the aeroplane:	
	Sphygmomanometer - non mercury
	Stethoscope
	Syringes and needles
	Oropharyngeal airways (2 sizes)
	Tourniquet
	Coronary vasodilator e.g. nitro-glycerine
	Anti-smasmodic e.g. hyascene
	Epinephrine 1 : 1000
	Adrenocortical steroid e.g. hydrocortisone
	Major analgesic e.g. nalbuphine
	Diuretic e.g. fursemide
	Antihistamine e.g. diphenhydramine hydrochloride
	Sedative/anticonvulsant e.g. diazepam
	Medication for Hypoglycaemia e.g. hypertonic glucose
	Antiemetic e.g. metoclopramide
	Atropine
	Digoxin

	Uterine contractant e.g. Ergometrine/Oxytocin
	Disposable Gloves
	Bronchial Dilator - including an injectable form

First-Aid Kits

(a) An operator shall not operate an aeroplane unless it is equipped with first-aid kits, readily accessible for use, to the following scale:

Number of passenger seats installed	Number of First-Aid Kits required
0 to 99	1
100 to 199	2
200 to 299	3
300 and more	4

(b) An operator shall ensure that first-aid kits are:

(i) Inspected periodically to confirm, to the extent possible, that contents are maintained in the condition necessary for their intended use; and

(ii) Replenished at regular intervals, in accordance with instructions contained on their labels, or as circumstances warrant.

Emergency Medical-Kit

(a) An operator shall not operate an aeroplane with a maximum approved passenger seating configuration of more than 30 seats unless it is equipped with an emergency medical kit if any point on the planned route is more than 60 minutes flying time (at normal cruising speed) from an aerodrome at which qualified medical assistance could be expected to be available.