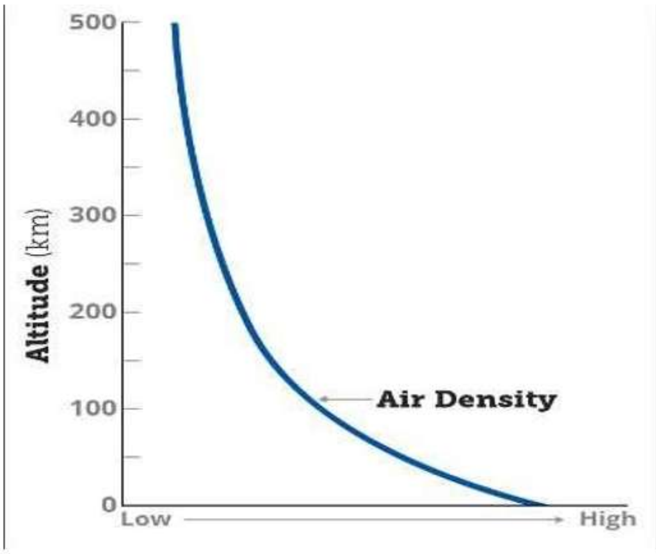


Methods of Measuring Pressure

Pressure, which is defined as force per unit area, may be measured directly either by balancing it against that produced by a column of liquid of known density, or it may be permitted to act over a known area and then measured in terms of the force produced. The former method is the one utilized in simple U-tube manometers, while the second enables us to measure the force by balancing it against a known weight, or by the strain it produces in an elastic material. In connection with pressure measurements, we are concerned with the following terms:

Absolute Pressure

The absolute pressure of a fluid is the difference between the pressure of the fluid and the absolute zero of pressure, the latter being the pressure in a complete vacuum. Thus, in using a gauge to measure the fluid pressure, the absolute pressure of the fluid would be equal to the sum of the gauge pressure and the atmospheric pressure.

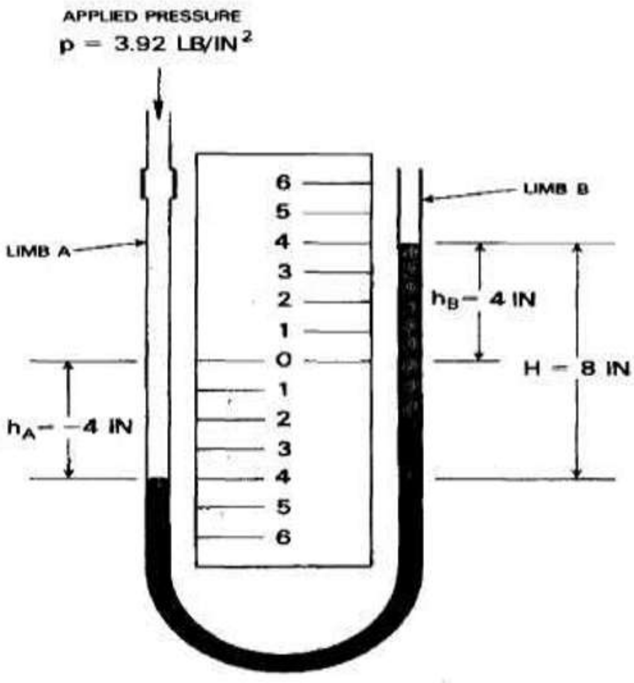
Gauge Pressure

Most pressure gauges measure the difference between the absolute pressure of a fluid and the atmospheric pressure. Such measurement is called the gauge pressure, and is equal to the absolute pressure minus the atmospheric pressure. Gauge pressure is either positive or negative, depending on its level above or below the atmospheric pressure reference.

U-Tube Manometer

The simple U-tube manometer consists of a glass tube partially filled with a liquid, usually water or mercury, which finds its own level at a point 0 within the open-ended limbs of the U. If a low- pressure source is connected to the limb A, then a force equal to the applied pressure multiplied by the area of the bore will act on the surface of the liquid, forcing it down limb A. At the same time the liquid is forced up the bore of limb B until a state of equilibrium exists and the levels of the liquid stand at the same distance above and below the zero point. By taking into account the area of the tube bore and the density of the liquid it is possible to calculate the pressure from the difference in liquid levels, as the following example shows.

Let us assume that the manometer is of the mercury type having a bore area of 3 in2, and that a pressure is applied to limb A such that at equilibriu m the mercury levels are 4 in below and 4 in above zero. The difference in levels is H and its



value is obtained by subtracting the lower level from the higher one; thus, H= hB-h, =4-(- 4) = 8 in.
Now, we must know the weight of the mercury column being supported, and this is calculated from volume multiplied by density. The volume in this case is 3H and the density of mercury is usually taken as 0.49 lbf/in3 . Thus, the weight of the column is 3H X 0.49 = 1.47 X 8 = 11.76 lb, and as the pressure balancing this is weight divided by area, then 1 1.7613 = 3.92 ibf/in2 is the pressure being applied to limb A and corresponding to a difference in mercury levels of 8 in. In the same manner, other pressures can be calculated from the corresponding values of level difference H. In practice, manometers are used for checking the calibration of pressure gauges, and so it is usually more convenient to graduate the manometer scale directly in pounds per square inch. If 3.92 lbf/in2 is represented by 8 in, then, for the mercury manometer we have considered, 1 lbf/in2 is equal to 8/3.92, or 2.04 in, and so a scale can be graduated with marks spaced this distance apart, each representing an increment of 1 lbf/in2. The equivalent value 2.04 in. Hg to I lbf/in2 is standard and results of calculations for differing bore areas will show that they are independent of the areas. If the water is used in the manometer the foregoing principles also apply, but as water has a much lower density than mercury, then for a given pressure the difference in level H for a water manometer will be much greater than that of a mercury manometer (2.04 in. Hg = 27.7 in.,H20 very nearly).
Elastic Pressure-Sensing Elements
For pressure measurements in aircraft, it is obviously impracticable to equip in the cockpit with U-tube manometers and dead-weight testers. It is the practice, therefore, to use elastic pressure-sensing elements, in which forces can be produced by applied pressures and made to actuate mechanical and/or electrical indicating elements. The sensing elements commonly used are Bourdon tubes, diaphragms, capsules and bellows.

Bourdon Tube
The Bourdon tube is about the oldest of the pressure-sensing elements. It was developed and patented in 1850 by a Parisian watchmaker (whose name it bears) and has been in general use ever since, particularly in applications where the measurement of high pressure is necessary. The element is essentially a length of metal tube, specially extruded to give it an elliptical cross-section, and shaped into the form of a letter C. The ratio between the major and minor axes depends on the sensitivity required, a larger ratio providing greater sensitivity. The material from which the tube is made may be either phosphor-bronze, beryllium-

bronze or beryllium-copper. One end of the tube, the 'free-end', is sealed, while the other end is left open and fixed into a boss so that it may be connected to a source of pressure and form a closed system.
When pressure is applied to the interior of the tube there is a tendency for the tube to change from an elliptical cross-section to a circular one, and also to straighten out as it becomes more circular. In other words, it tends to assume its original shape. This is not such a simple as it might appear and many theories have been advanced to explain it. However, a practical explanation sufficient for our purpose is as follows. Firstly, a tube of elliptical crosssection has a smaller volume than a circular one of the same length and perimeter. This being the case, an elliptical tube when connected to a pressure source is made to accommodate more of the liquid, or gas, than it can normally hold. In consequence, forces are set up which change the shape and thereby increase the volume. The second point concerns the straightening out of the tube as a result of its change in cross-section. Since the tube is formed in a C-shape then it can be considered as having an inner wall and an outer wall, and under 'no pressure' conditions they are each at a definite radius from the centre of the C. When pressure is applied and the tube starts changing shape, the inner wall is forced towards the centre, decreasing the radius, and the outer wall is forced away from the centre thus increasing the radius. Now, along any section of the curved tube the effects of the changing radii are to compress the inner wall and to stretch the outer wall, but as the walls are joined as a common tube, reactions are set up opposite to the compressive and stretching forces so that a complete section is displaced from the centre of the C. Since this takes place at all sections along the tube and increases towards the more flexible portions, then the resultant of all the reactions will produce maximum displacement at the free end. Within close limits the change in angle subtended at the centre by a tube is proportional to the change of internal pressure, and within the limit of proportionality of the material employed, the displacement of the free end is proportional to the applied pressure. The displacement of the free end is only small; therefore, in order to transmit this in terms of pressure, a quadrant and magnifying system is employed as the coupling element between tube and pointer.

Diaphragms

Diaphragms in the form of corrugated circular metal discs, owing to their sensitivity, are usually employed for the measurement of low pressures. They are always arranged so that they are exposed at one side to the pressure to be measured, their deflections being transmitted to pointer mechanisms, or to a warning-light contact assembly. The materials used for their manufacture are generally the same as those used for Bourdon tubes. The purpose of the corrugations is to permit larger deflections, for given thicknesses, than would be obtained with a flat disc. Furthermost number and depth control the response and sensitivity characteristics; the greater the number and depth the more nearly linear is its deflection and the greater is its sensitivity.

Capsules

Capsules are made up of two diaphragms placed together and joined at their edges to form a chamber which may be completely sealed or open to a source of pressure. Like single diaphragms they are also employed for the measurement of low pressure, but they are more sensitive to small pressure changes. The operation of capsules in their various applications has already been described in the chapters on height and airspeed measuring instruments.

Bellows

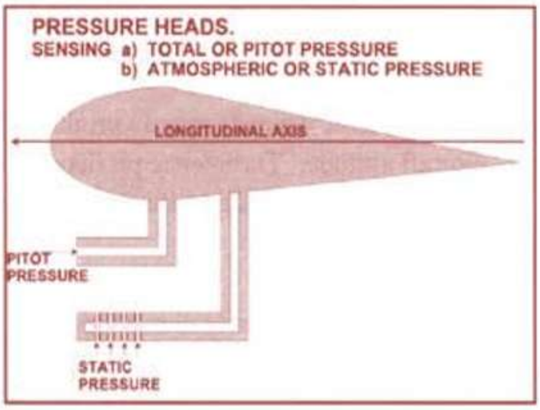
A bellows type of element can be considered as an extension of the corrugated diaphragm principle, and in operation it bears some resemblance to a helical compression spring. It may be used for high, low or differential pressure measurement, and in some applications a spring may be employed (internally or externally) to increase what is termed the 'spring-rate' and to assist a bellows to return to its natural length when pressure is removed. The element is made from a length of seamless metal tube with suitable end fittings for connection to pressure sources or for hermetic sealing.

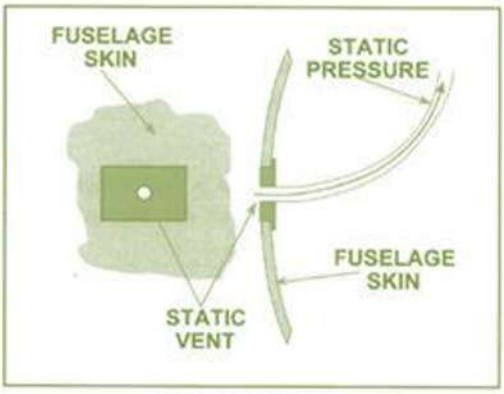
PITOT STATIC SYSTEM INTRODUCTION

An aircraft at rest on the ground in still air is subject to normal atmospheric pressure, which bears equally on all parts of the aircraft. This ambient pressure is known as Static pressure. An aircraft in flight, while still subject to the static pressure at its flight level, experiences an additional pressure on the leading edges due to the resistance Of the air to the aircraft's movement. This additional pressure is Dynamic pressure, and its value depends on the speed Of the aircraft through the air and on the density Of the air. The leading edges, therefore, encounter a total pressure consisting of static plus dynamic pressures. This total pressure is also known as Pitot pressure. Two of the pressure-dependent flight instruments, the altimeter and vertical speed indicator, operate solely on static pressure, whereas the airspeed indicator and mach meter utilize both static and pitot pressures. Inside an aircraft, pressure and temperature are seldom the same as outside the aircraft so pitot and static pressures must be sensed by devices mounted on the outside of the aircraft.

PITOT/STATIC HEADS.

An open-ended tube parallel to the longitudinal axis of the aircraft is used to sense the total pressure (static plus dynamic). This device is a 'pitot tube' mounted in a 'pitot head'.
The Open end of the tube faces into the moving
airstream, the other end leading to the airspeed capsules in the ASI and mach meter. The moving airstream is thus brought to rest in the tube, so generating the extra (dynamic) pressure which together with the static pressure already in the tube provides the required total (pilot) pressure. A 'static head' consists of a tube with its forward end sealed but with holes Or slots Cut in the sides. These slots do not face into the airflow and therefore they sense only the static pressure. This pressure supplies the static 'line' to the pressure instruments. A pressure sensing system consisting of separate pitot and static heads





STATIC VENT

POSITION ERROR

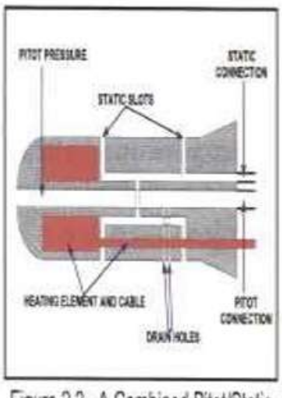


Figure 2.2. A Combined Pitot/Static Pressure Head.

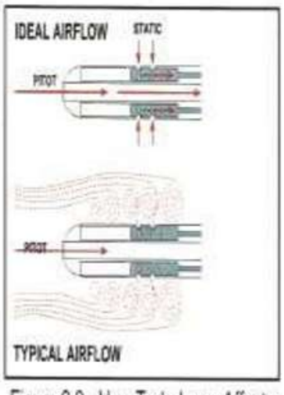


Figure 2.3. How Turbulence Affects the Value of Static Pressure

It will be appreciated that if, due to turbulent airflow in the region of the pitot/static heads, the pressures sensed are not truly representative of the pitot and static pressures, the pressure dependent instruments will not read correctly. The error involved is called Position Error (Or alternatively 'pressure' error). At large angles of attack the pressure head is inclined at an angle. Its the airstream so that position error is usually biggest at the lower airspeeds. Flight manuals may list different values of position error for different flap settings. Position error depends mainly on the positioning of the pressure head, the airspeed, and the aircraft attitude. Turbulence produced in the airstream by the pressure head itself affects the value of static pressure sensed rather than the pitot pressure. Approximately 95% of the position error associated with a combined pressure head is produced by turbulence about the static head. Because of this, the Static Vent, was introduced as a source of static pressure instead of the static head, pitot pressure then being sensed by a simple pitot head. About 90% of the combined pressure head position error is eliminated by use of a separate pitot head and static vent There is usually some place on the airframe, usually on the side of the fuselage, where true (or nearly true) static pressure obtains over the whole speed range of the aircraft.

A flat metal plate is fitted at this position, the static line from the pressure instruments terminating at a small circular hole - the static vent - in this plate. A similar vent may be positioned on the opposite side of the fuselage and the two interconnected for transmission of static pressure to the instruments so that errors produced by yawing are largely eliminated. The shock waves associated with flight at high mach numbers can produce significant errors in pressure sensed by a static vent. Modern high speed aircraft may accordingly be fitted with a more sophisticated combined pitot/static pressure head in Order to keep position error within acceptable limits. The choice of location for a probe, or vent, is dependant upon the aerodynamics of the aircraft. Typical locations are: ahead of a wing tip, under a wing, ahead of the vertical stabiliser tip, at the side of the fuselage nose section, and ahead of the fuselage nose section.

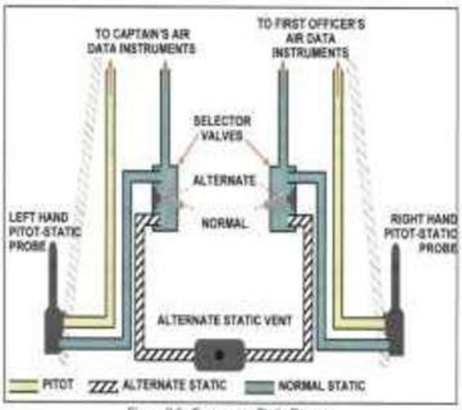


Figure 2.6. Emergency Static Source.

EMERGENCY STATIC
SOURCE.

An emergency static source is normally provided in the event of the static head/vents becoming blocked. The emergency static source may be to the outside of the aircraft or

from inside the cabin(in unpressurised aircraft only). On those aircraft which sense emergency static pressure from outside the hull of the aircraft, the source will be less accurate than the primary (blocked) static vent head, since that would have been in the optimum position. When an emergency static source is fed from within the cabin, the static pressure sensed is likely to be lower than ambient due to aerodynamic suction.

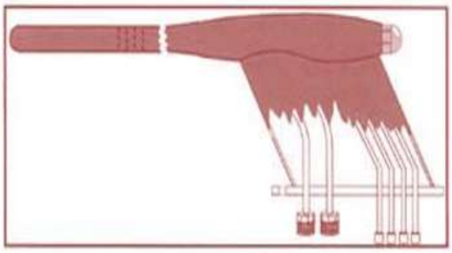
Note: When alternate (standby) pressure Systems are used, correction values for the instruments concerned may be found in the Operating Data Manual for the aircraft.

ADVANTAGES OF THE STATIC VENT.

A. The airflow in the region of the vents is less turbulent and the static pressure measured is more accurate.
B. Errors produced when side slipping or yawing are reduced.
C. Duplication Of vents either side Of fuselage reduces blockage errors.

PRE FLIGHT CHECKS OF THE PITOT/STATIC SYSTEM.

A. All covers and plugs removed and stowed.
B. All tubes, holes, slots free of obstructions.
C. Pilot head heater operating.



HIGH SPEED PITOT STATIC PROBE

ALTIMETER PRINCIPLE OF OPERATION

The pressure altimeter is a simple, reliable, pressure gauge calibrated to indicate height. The pressure at a point depends on the weight of the column of air which extends vertically upwards from the point to the outer limit of the atmosphere. The higher an aircraft is flying, the shorter is the column Of air above it and consequently the lower is the atmospheric pressure at the aircraft. In Other words, the greater the height, the lower the pressure, and by measuring the pressure the altimeter measures height. Unfortunately, the relationship between pressure and height is not a linear one, so that calibration of the altimeter scale is not a simple matter. The situation is further complicated by high and low pressure weather systems which produce pressure differences in the horizontal plane. Furthermore, the temperature of the air at the surface and the temperature lapse rate in the air above vary considerably; this affects pressure.

CALIBRATION

With all these variables it becomes necessary to assume certain average or 'standard' conditions, base the calibration formulae on these, and then apply corrections appropriate to the deviations from standard conditions which Occur with position and time. The conditions used for calibration are usually those assumed for the ISA (International Standard Atmosphere). The relevant assumptions are:

a. At mean Sea Level Pressure 1013.25 millibars Temperature
+15‘C
b. From MSL up to 11 km (36,090 feet) Temperature falling at 6.5_c per km (1.98 OC/IOOO feet)
c. From 11 km to 20 km (65,617 feet) A constant temperature Of
- 56.5_c
d. From 20 km to 32 km (104, 987 feet) Temperature rising at 1‘c per km (0.30/1000 feet). Density 1225gm m'

With these assumptions, the pressure corresponding to any given level in the ISA can be calculated from the calibration formulae. Graphs or tables can be produced showing height in terms of pressure under standard conditions. These tables can be used for the manufactures calibration of the altimeter scale. Basically, the laboratory calibration consists of applying a series of pressures to

the altimeter and checking that the instrument indicates the respective levels which correspond to these pressures in the ISA. Any discrepancies, if within certain agreed tolerances, would be listed over the operating height ranges as instrument errors. (The calibration is carried out with increasing and decreasing readings so that the amount of lag at calibration conditions can be determined).
Note: 1 The Pressure Altimeter is calibrated to give a Linear Presentation Of the Non-linear Atmospheric distribution. This is achieved by the use of a variable magnification lever system and the dynamic design of the capsules.
Note: 2 Temperature compensation is achieved by the use of a bi- metal compensator connected in the lever/linkage system.
Note: 3 Conversion between "Inches of Mercury (Mg)" and "Hectopascals (Hpa) / Millibars (Mb)" can be achieved by the formula Hg x 33.86 — "pa / Mb
and the reverse by the formula Hpa / Mb x 0.0295 — Hg.

CONSTRUCTION

Simple Altimeter. Static pressure is fed into the case of the instrument from the static source. As height increases, static pressure decreases and the capsule expands under the control of a leaf spring. A mechanical linkage magnifies the capsule expansion and converts it to a rotational movement of a single pointer over the height scale. The linkage incorporates a temperature- compensating device to minimize errors caused by expansion and contraction of the linkage and changes in spring tension due to fluctuations in the temperature of the mechanism.

SIMPLE ALTIMETER

The simple altimeter has a setting knob which is geared to the pointer. With this knob the pointer can be set to read zero with the aircraft on the ground so that when airborne the altimeter indicates approximate height above aerodrome level. Alternatively the pointer can be Set (before flight) to the aerodrome elevation so that when airborne the instrument shows
approximate height above mean sea level. The simple altimeter has a setting
knob which is geared to the pointer. With this knob the pointer can be set to read zero with the aircraft on the ground so that when airborne the altimeter indicates approximate height above aerodrome level. Alternatively the pointer can be Set (before flight) to the aerodrome elevation so that when airborne the instrument shows approximate height above mean sea level.

Sensitive Altimeter.
Most aircraft are now equipped with the three-pointer or 'sensitive' type altimeter.
The principle of operation is similar to that of the simple altimeter but there are the following refinements:-

SENSITIVE ALTIMETER

A. A bank of two or three capsules gives the increased movement necessary to drive three pointers. These are geared 100: 10: I, the smallest indicating 100,000 feet per revolution, the next 10,000 feet per revolution and the largest 1,000 feet per revolution.
B. Jeweled bearings are fitted, reducing friction and the associated lag in indications.
Note: Some altimeter systems employ "Knocking / Vibrating" devices to help overcome initial inertia of the internal gear train when transmitting movement from the capsules to the pointers.
C. A variable datum mechanism is built in. This, with the aid of a setting knob, enables the instrument to be set to indicate height above any desired pressure datum.

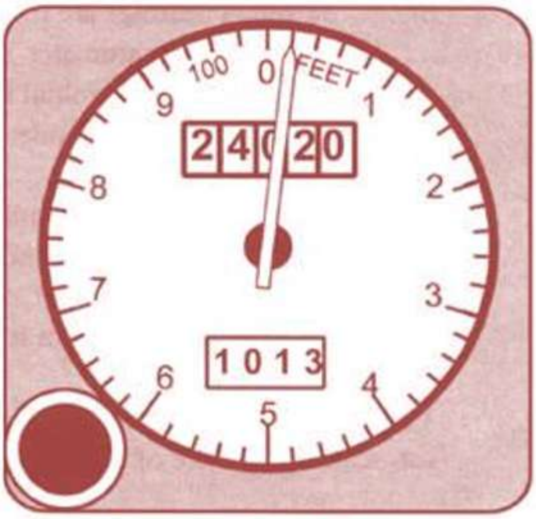
The variable datum mechanism is used as follows:-
The pilot turns the knob until the desired pressure level (say, 1005 mb) appears on a pressure sub-scale on the face of the instrument.
As he turns the knob, the height pointers rotate until, when the procedure is completed with the sub-scale showing the desired 1005, the altimeter indicates the aircraft's height above this pressure level. If for instance the aerodrome level pressure happened to be 1005 mb, the altimeter would be reading height above the aerodrome (and the pilot would have set a 'QFE' of 1005 on the sub-scale). The sub-scale setting only changes when the pilot turns the knob. A change in altitude or surface pressure has no direct effect on the reading of the sub- scale. As the pilot alters the sub-
scale setting, the altimeter pointers move, but the design of the mechanism is such that the reverse does not apply (for example, during a climb, the pointers rotate but the sub-scale setting remains unchanged). British altimeters have a sub-scale setting range between 800 to millibars.

READING ACCURACY
The simple altimeter is not sensitive, recording perhaps for each revolution of its single pointer. The three-pointer instrument gives a much more sensitive indication of height and change of height but suffers from the severe disadvantage that it can be easily misread. It is not difficult for the pilot to make a reading error of10,000ft, particularly during a rapid descent under difficult conditions with a high flight-deck work- load. Accidents have occurred as a result Of such misreading. Various modifications to the pointers and warning systems have been tried with the Object of preventing this error, including a striped warning sector which appears as the aircraft descends through the 16,000 feet level.

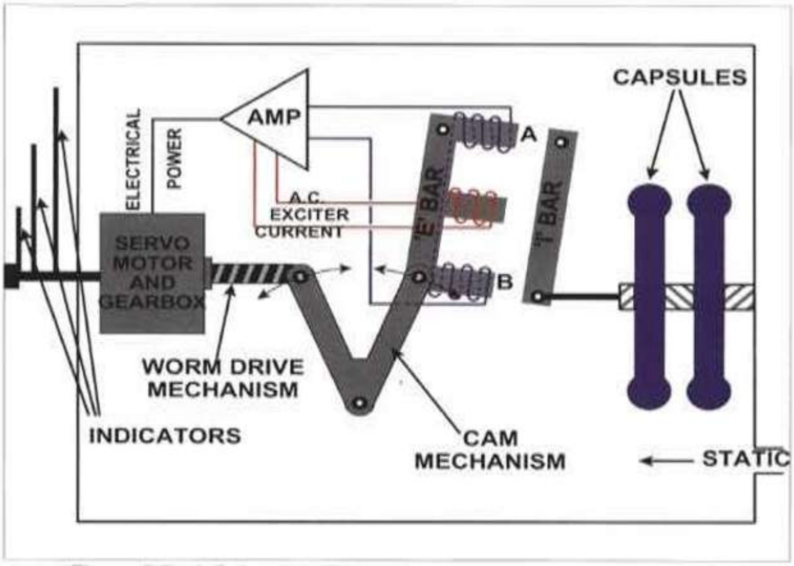
The greatest advance has been the introduction to the counter- pointer altimeter, illustrated in Figure, which gives a much more positive indication than the three-pointer, it will' be realized that though the digital counters give an unambiguous indication Of the aircraft's height, they give a relatively poor display of the rate of change of height. For this reason the instrument also has a single pointer which makes one revolution per 1000 feet, giving the clear indication of change of height which is extremely important to the pilot, particularly on the final approach in instrument conditions.

SERVO-ASSISTED ALTIMETERS.

Though at least one counter-pointer altimeter driven directly by pressure capsules has been produced, most instruments of the type are servo- assisted. This servo- assistance not only gives the altimeter an



A Counter Pointer Altimeter



increased operating range but also improves the instrument accuracy, particularly at high levels. At high altitude the change in pressure corresponding to a given change in height is much smaller than at low altitude. This means that for a given height change, capsule movement at high altitude is relatively small, and frictional resistance in the linkage of an unassisted altimeter causes correspondingly greater errors and more lag. With servo- assistance, the requisite power is available to overcome the frictional resistance with consequently enhanced instrument accuracy.

SERVO ALTIMETER

The principle of the servo-altimeter is that the small movements of the capsules are detected by a very sensitive electro-magnetic pick- off. This produces an electric current which is amplified and used to drive a motor which rotates the counters and pointer. AC is fed to the middle leg of the E bar, setting up alternating magnetic fields in the Outer legs 'A' and The coils on these two legs are wound 180* out of phase. The exciter therefore induces a current in each leg, but since these are 180* out of phase and of equal strength, they cancel each other out when the I bar is equidistant from the legs of the E bar (that is when no pressure change acts on the capsules. With a change of pressure the capsules expand or contract, moving the I bar on its pivot, closing the gap between the I Bar and E Bar at one end and opening it at the other. This causes an imbalance of magnetic fields and therefore of the currents induced in the 'A' and 'B' coils. The imbalance causes an error signal which is passed to the amplifier, where it is amplified and rectified, and thence to the servo motor. The servo motor drives the counter-pointer system of the altimeter and at the same time, via a cam drive, re-aligns the E Bar with the I Bar. Once re-aligned, the error signal ceases and the altimeter indicates the correct height. In this system the only work required of the capsules is to move the I Bar, eliminating the effects of friction and manufacturing imperfections in the gearing of a conventional altimeter. This type of altimeter is sensitive to very small pressure changes and therefore more accurate than the sensitive altimeter, particularly at high altitudes where pressure changes (per unit height increment) are very small. The lag experienced in other types of altimeter with rapid changes of height is greatly reduced. The normal instrument error is approximately equivalent to the effect of 1mb change of pressure (about 30 ft at msl, 50 ft at 20,000ft, or at 40,000 R). The tolerance at MSL (JAR - 25) is ± 30ft per 100 kts CAS a further development in the indication of vertical position comes with the Air Data Computer (ADC) installed in many transport aircraft. The ADC provides (among Other things) an electrical output proportional to static pressure which can be used to drive a counter-pointer altimeter. the previously required altimeter pressure capsules being dispensed with, unless a standby capability is required.

Typical Simple Altimeter (range zero to 35,000 feet)

Height (feet)	0	35,000
Tolerance (feet)	±100	± 1,000

Typical Sensitive Altimeter (range zero to 80,000 feet)

Height (feet)	0	40,000	80,000
Tolerance (feet)	±70	± 600	± 1,500

Typical Servo Altimeter (range zero to 100,000 feet)

Height (feet)	0	40,000	60,000	100,000
Tolerance (feet)	±30	± 100	± 300	± 4,000

ALTIMETER ERRORS

The errors which affect altimeters are many and the extent of some of them varies with altimeter type. Much effort is expended on improving instrument accuracy, and the permissible tolerances of modern altimeters are smaller than with earlier types. There are other errors caused by deviation of the actual atmosphere from standard conditions, and also the difficulty in sensing correctly the outside air pressure. A list of the main errors follows.

a) Time Lag. With many types of altimeter the response to change Of height is not instantaneous. This causes the altimeter to under- read in a climb and over-read in a descent. The lag is most noticeable when the change in altitude is rapid and prolonged. In the laboratory calibration Of the sensitive altimeter, the lag between increasing readings and decreasing readings should not exceed 150 feet. With servo-assisted altimeters there is said to be no appreciable lag unless the rate Of change of height exceeds 10,000 feet per minute. This is because the servo-altimeter does not suffer from the linkage friction which causes a much larger error in the sensitive altimeter.

b) Instrument Error. Manufacturing imperfections, including friction in the linkage, cause errors throughout the operating range. The errors are kept as small as possible by adjustments within the instrument, and the calibration procedure ensures that they are within permitted tolerances. Residual errors may be listed on a correction card.

c) Position (Or Pressure) Error. This is largely due to the inability to sense the true static pressure outside the aircraft, as described in the chapter on Pressure Heads. The error is usually small but increases at high mach numbers (and, consequently, at high altitudes
usually associated with high mach numbers). Altimeters driven by an Air Data Computer (ADC) may have their accuracy improved by the ADC automatically correcting its static output signal for position error.

d) Manoeuvre-Induced Error. This is caused by transient fluctuations of pressure at the static vent during change of, mainly, pitch attitude and delays in the transmission of pressure changes due to viscous and acoustic effects in the static pipeline.

e) Barometric Error. Providing the altimeter has a pressure sub- scale, and the local QNH is set on it, the altimeter will indicate height AMSL (though still subject to the other errors). If the local surface pressure has changed since the QNH value was set, a barometric' error Of roughly 30 feet per millibar will result. If pressure has fallen the altimeter over-reads.

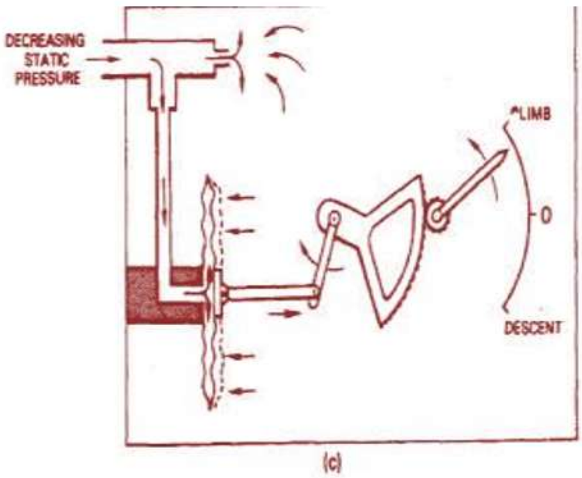
f) Temperature Error. Even with no other errors at all, the pressure altimeter will not indicate true altitude (height AMSL) unless the surface temperature and lapse rate Of the column of air are those assumed in the calibration. When flying in colder air (with an air density greater than ISA at that altitude), the altimeter will Over-read. Where the temperature at cruising level deviates from standard, an approximate correction can be made with most navigational computers. The correction can only be approximate since temperatures in the rest of the column Of air are not known. The correction is considered too inaccurate to be worth making at heights above 25,000 feet.

VERTICAL SPEED INDICATOR

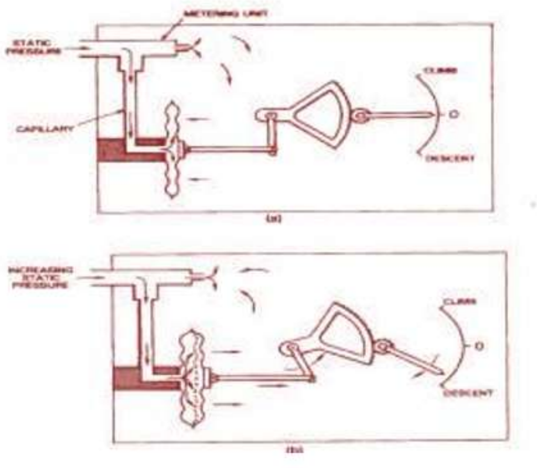
These indicators, also known as rate-of-climb indicators, are the third of the primary group of pitot-static flight instruments, and are very sensitive differential pressure gauges, designed to indicate the rate of altitude change from the change of static pressure alone but the operative clause is 'the rate at which the static pressure changes and as this involves a time factor we have to introduce this into the measuring system as a pressure function. It is accomplished by using a special air metering unit, and it is this which establishes the second pressure required. An indicator consists basically of three main components, a capsule, an indicating element and a metering unit, all of which are housed in a sealed case provided with a static pressure connection at the rear. The dial presentation is such that zero is at the 9 o'clock position; thus the pointer is horizontal during straight and level flight and can move from this position to indicate climb and descent in the correct sense. Certain types of indicator employ a linear scale, but in the majority of applications indicators having a mechanism and scale calibrated to indicate the logarithm of the rate of pressure change are preferred. The reason for this is that a logarithmic scale is more open near the zero mark and so provides for better readability and for more accurate observation of variations from level flight conditions.

The indicator mechanism, from which it will be noted that the metering unit forms part of the static pressure connection and is connected to the interior of the capsule by a length of capillary tube. This tube serves the same purpose as the one employed in an airspeed indicator, i.e. it prevents pressure surges reaching the capsule.

It is, however, of a greater length due to the fact that the capsule of a vertical speed indicator is much more flexible and sensitive to pressure. The other end of the metering unit is open to the interior of the case to apply static pressure to the exterior of the capsule. Let us now see how the instrument operates under the three flight



(c)



conditions: (a) level flight, (b) descent, and [c] climb

In level flight, air at the prevailing static pressure is admitted to the interior of the capsule, and also to the instrument case through the metering unit. Thus, there is zero differential across the capsule and the pointer indicates zero. We will now consider the operation during a descent. At the instant of commencing the descent the differential pressure will still be zero, but as the aircraft descends into the higher static pressure this will be applied at the static pressure connection of the instrument causing air to flow into the capsule and case. As the capsule is directly connected to the static pressure connection, the flow of air will create the same pressure inside the capsule as that prevailing at the levels through which the aircraft is descending. The pressure inside the case, however, is not going to be the same because the metering unit is a specially calibrated leak assembly designed to restrict the flow of air into or out of the instrument case. Therefore, as far as the case pressure is concerned, it is still at the same value which obtained at the original level flight altitude, and cannot build up at the same rate as the pressure in the capsule is increasing. The restriction of the metering unit thus provides the second pressure from one source and establishes a differential pressure across the capsule, causing it to distend and make the pointer indicate a descent. This, of course, is just what is required, but during the descent the case pressure must be maintained lower than the capsule pressure and made to change at the same rate in order to obtain a constant differential pressure. The metering unit, being a restrictor, increases the velocity of the air flowing into the static pressure connection, and as happens with devices of this nature, increased velocity brings about a reduction in pressure. In addition, the instrument case is of much greater volume than the capsule; consequently the flow of air into the case is going to take some time to build up a pressure equal to that coming in at the static pressure connection. By the time this is reached, however, the aircraft will have descended to a new altitude and the static pressure will have again changed. Thus, the metering unit introduces the required rate and time-lag factors, and differential pressure across the capsule which positions the pointer to indicate the altitude change in feet per minute. The design of a system is such that it takes approximately four seconds for the case pressure to build up to that in the capsule; but as the capsule always has an unrestricted air flow to it, it will lead the case by four seconds and so there will be a constant difference in pressure between them corresponding to four seconds in time. The differential pressures produced are not very large, a typical value being approximately 20 mm H20 at full-scale deflection of the pointer.
During a climb, the metering unit will establish the required factors and differential pressure, but as the static pressure under this condition is a decreasing one, and because the metering unit restricts the flow out

of the case, the case pressure leads the capsule pressure. Apart from the changes of static pressure with changes of altitude, which as we know are not constant, air temperature, density and viscosity changes are other very important variables which must be taken into account, particularly as the instrument depends on rates of air flow. From the theoretical and design standpoints, a vertical speed indicator is therefore quite complicated, but the metering units are designed to compensate for the effects of variables over the ranges normally encountered.

Metering Units is known as a 'capillary-and-orifice' type, the two devices in combination providing compensation for the effects of the atmospheric pressure and temperature variables
The pressure difference across a capillary, for a constant rate of climb, increases with increasing altitude and at a constant temperature. Thus, the use of a capillary alone would introduce a positive error in instrument indications at altitudes above sea-level. With an orifice, the effect is exactly the opposite. The primary reasons for the difference are that the air flow through a capillary is a laminar one while that through an orifice is turbulent; furthermore, the rate of flow through a capillary varies directly as the differential pressure, while that through an orifice varies as the square root of the differential pressure. In combining the two devices we can therefore obtain satisfactory pressure compensation at a given temperature. The differential pressure across a capillary also depends on the viscosity of the air, and as this is proportional to the absolute temperature, it therefore decreases with decreasing temperature.
The differential pressure across an orifice varies inversely as the temperature, and therefore increases with decreasing temperature. Thus, satisfactory temperature compensation can be obtained by combining the two devices. The sizes of the orifice and capillary are chosen so that the readings of the indicator will be correct over as wide a range of temperature and altitude conditions as possible.

Instantaneous Vertical Speed Indicators

These indicators consist of the same basic elements as conventional VSIs, but in addition they employ an accelerometer unit which is designed to create a more rapid differential pressure effect, specifically at the initiation of a climb or descent. The accelerometer comprises two small cylinders or dashpots, containing pistons held in balance by springs and their own mass. The cylinders are connected in the capillary tube leading to the

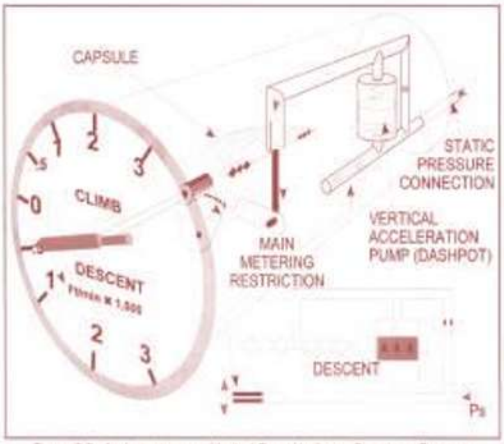


Figure 6.2: An Instantaneous Vertical Speed Indicator Showing a Descent.

capsule, and are thus open directly to the static pressure source. When a change in vertical speed occurs initially, the pistons are displaced under the influence of a vertical acceleration force, and this creates an immediate pressure change inside the capsule, and an
instantaneous indication by the indicator pointer. The accelerometer response decays after a few seconds, but by this time the change in actual static pressure becomes effective, so that a pressure differential is produced by the metering unit in the conventional manner.

THE ERRORS OF THE VSI.

a) Instrument Error. Due to manufacturing imperfections.

b) Position (or Pressure) Error. If the static pressure is subject to position error the V.S.I. will wrongly indicate a climb or descent when speed is suddenly changed, this is most noticeable during take-off acceleration.

c) Manoeuvre-Induced Error. Any short term fluctuations in pressure at the static vent during attitude changes will cause the instrument to indicate a false rate of climb or descent. Additionally with most V.S.I.s, the linkage includes a small counterbalance weight, the inertia of which causes delays in the indications of changes in vertical speed during manoeuvres.

d) Time Lag. The pointer takes a few seconds to steady because ofthe time taken to build up a steady pressure difference on climb or descent. There will also be a time lag on levelling out because of the time taken for the pressures to equalise. This error is most noticeable after a prolonged climb or descent, especially at a high rate.

Any blockages of the static line or vent will cause the needle to return to zero. If the supply of air to this instrument is blocked it is probable that the other pressure instruments (A.S.I., altimeter and machmeter) will also be affected.

Note:

1) The Vertical Speed Indicator as fitted in Glider Aircraft is sometimes known as a "Variometer" (a simple tube device - Green for upward movement / Red for downward movement).

2) The device fitted to the IVSI to give an instant indication of vertical movement may be of the Dashpot or Dynamic-Vane type - the indications and errors being the same for both instrument types.

SERVICEABILITY CHECKS

On the Ground.

a) The instrument should read zero, or the error should be within the permissible limits

i) ± 200 feet per minute at temperatures - 200C + 500C

ii) ± 300 feet per minute outside these temperatures

b) There should be no apparent damage to the instrument.

In The Air.

The accuracy of the instrument may be checked against the altimeter and a stop watch during a steady climb/descent and the instrument

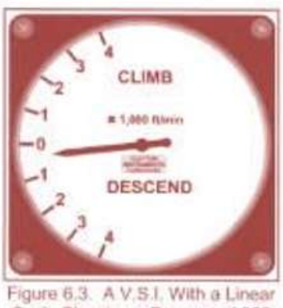


Figure 6.3. A V.S.I. With a Linear Scale Showing a Descent of 250 ft/min.

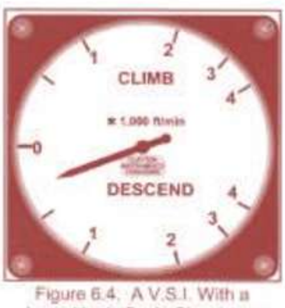


Figure 6.4. A V.S.I. With a Logarithmic Scale Showing a Descent of 250 ft/min.

should indicate zero climb or descent when in level flight.

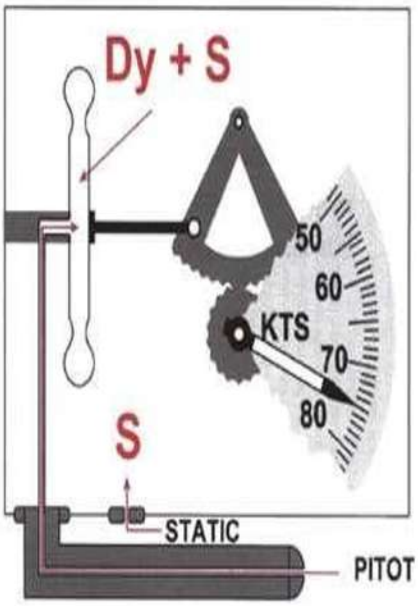
AIRSPEED INDICATOR PRINCIPLE OF OPERATION

The notes on pressure heads explain that whereas an aircraft on the ground in still air is subject only to atmospheric (static) pressure (S), the leading edges of an aircraft in forward flight are subject to an additional (dynamic) pressure. This results in a total (pitot) pressure (P) on the leading edges of dynamic

pressure plus static pressure.
Pitot = Dynamic + Static or P = Dy + S
The dynamic pressure is often called 'pitot excess' pressure (PE) so we have:-
P = PE + s
The pitot head senses pitot pressure and the static/vent senses static pressure. These two pressures are fed to the airspeed indicator, a differential pressure gauge, which measures their difference PE (the dynamic pressure). Now dynamic pressure isa measure of airspeed, because:- Dynamic Pressure — 1/4pV2 where V is true airspeed and p is density of the surrounding air.

CONSTRUCTION.

In principle, the simple ASI can be considered as an airtight box divided by a flexible diaphragm, with pitot pressure fed to one side and static pressure to the other side. The pressure difference across the diaphragm is (Dy + S) - S, which is Dy, the dynamic pressure. Accordingly, the diaphragm deflects by an amount proportional solely to this dynamic pressure, its movement being transmitted by a system of levers to the indicating needle on the face of the ASI. Note that static pressure is common to both sides of the diaphragm, and so does not influence diaphragm movement.



In practice, the static pressure is fed into the hermetically- sealed instrument case, the pitot pressure being piped to a thin metal capsule capable Of expansion and contraction. Note that the pressure differential between the inside and outside of the capsule is (Dy+S)-S which is Dy, as with the diaphragm.

Expansion or contraction on the capsule will
therefore be proportional to the changes in dynamic pressure produced by changes of airspeed. The

capsule movements are transmitted by a temperature-compensated magnifying linkage to the pointer indicating airspeed on the face of the ASI.

CALIBRATION

The dynamic pressure depends not only on the speed of the aircraft but also on the air density. This density varies with temperature and pressure and so with altitude. The ASI is calibrated to read true airspeed for the air density of 1225 grams per cubic metre which would be produced by the ISA MSL pressure Of 1013.25 mb and temperature + 15*C (dry air conditions). No allowance is made in the calibration for the change in density which occurs with change of altitude. It follows that even if there were no other errors, the ASI could only indicate TAS when ISA MSL air density existed at the flight level, that is to say when the aircraft was flying in air having a density of 1225 grams per cubic metre. This density value can only be found at or close to sea level (depending on how much the actual conditions deviate from standard).
A functional diagram of an ASI

A.S.I. ERRORS

Density Error. Unless the air round the aircraft is at the calibration density Of 1225 grams per cubic metre, which can Only occur near sea level, the ASI cannot correctly indicate TAS. The dynamic pressure is proportional to density, so at altitude, where density is less, the dynamic pressure generated by a given TAS will be less than for the same TAS in flight at sea level. ASI capsule expansion will be proportionately less and the speed indicated will be less than the true speed.
Instrument Error. Manufacturing imperfections and usage result in small errors which are determined on the ground under laboratory conditions by reference to a datum instrument. A correction card can be produced for the speed range of the instrument.
Position Error. Alternatively known as 'pressure' error, this arises mainly from the sensing of incorrect static pressure, and is described more fully in the section entitled Pressure Heads. Position errors throughout the speed range are determined by the aircraft manufacturer during the test flying program for a particular aircraft type. It is not unusual to compile a joint correction card for position and instrument errors and place it in the aircraft near the ASI concerned.

Manoeuvre-Induced Errors. These are associated chiefly with maneuvers involving change in angle of attack, giving transient errors and a lag in the indication of changes in airspeed.
Compressibility Error. Air is compressible and except at true airspeeds below about 150 knots where the effect is negligible, the pressure produced in the pitot tube is higher than it would be for an ideal incompressible fluid, for which the dynamic pressure is h1/2pV2.

Order Of Correction.
a) Apply P and I correction to IAS, giving CAS.
b) At high speeds, apply the subtractive compressibility correction to CAS to give EAS.
c) From CAS Or EAS, obtain TAS by use of the computer (or a special chart) to correct for density error.

Air Speed Definitions.
IAS(Indicated Air Speed) = indicated reading on instrument CAS(Calibrated Air Speed) = IAS ± correction for instrument & position error
EAS(Equivalent Air Speed) = CAS ± compressibility corrections TAS(True Air Speed) = CAS ± density error compressibility error or EAS ± density error

More ASI Definitions:
Vso=The stall speed or the minimum steady flight speed in the landing configuration.

Vsl = The stall speed or the minimum steady flight speed in a specified configuration.
Vfe = The maximum Flap Extension speed
Vno = The maximum normal operating limit speed
Vne The Never Exceed speed
Vlo = The maximum Landing Gear Operation speed (up or down).
Vle = The maximum speed Landing Gear Extended speed
Vyse =Best rate of climb when Single Engine (2 engine a/c)

Some ASI's incorporate coloured markings on the dial - these 'range markings' consist of coloured arcs and radial lines.
The White Arc denotes the flap operating range, from stall at maximum AUW in the landing configuration (full flap, landing gear down, wings level, power-on up to VFF (maximum flaps extended Speed).
The Green Arc denotes the normal Operating speed range, from stall speed at maximum all-up weight (flaps up, wings level) up to V NO (―normal operating limit speed‖ or ―maximum structural cruising speed‖) which should not be exceeded except in smooth air. Operations at IASs in the green arc should be safe in all conditions, including turbulence.
The Yellow Arc denotes the caution range, which extends from VNO (normal operating limit speed) up to VNE (the never exceed speed). The aircraft should be operated at LASS in the caution range only in smooth air.
A Red Radial Line denotes VNE, the never exceed speed. Some ASIs have blue radial lines to denote certain important speeds, (e.g. best single-engines speed for a light twin-engined aeroplane). Optionally for piston engined light twins:
A blue radial line denotes the best rate of climb speed for one engine out, maximum weight, at mean sea level (Vyse).
A red radial line denotes the minimum control speed at maximum weight (Vmca).
A useful formula for estimating TAS is: TAS CAS + (1.75% of CAS per I ft of Altitude).
e.g. for a CAS of100ktat 10,000 n: TAS + (1.75 x 100x 10) =
117.5 kt
TOLERANCE - (J AR 25) - ± 3% or 5 Kts whichever is the greater.

BLOCKAGES AND LEAKS
Pitot Head: If the pitot head becomes blocked, the ASI reading will, in general, remain unchanged. In level cruise, a blockage (probably ice) will lock in the previous pitot pressure and any change in actual airspeed will not be registered. However, a slow leakage in the pitot pipeline is likely, so that the IAS gradually decreases. If altitude is changed with a blocked pitot head and clear static source, the IAS will increase during a climb because the pressure locked inside the capsule remains constant while the static pressure of the air surrounding the capsule decreases. Conversely the IAS decreases during a descent with blocked head.
Static Head: A Static head is more exposed to icing conditions and is therefore more likely to become obstructed than is a static vent. A blocked static source during descent will mean that the 'old' (higher altitude) static pressure Surrounding the capsule will be lower than it should be, so that if the pitot supply is normal the ASI will over-read. This could be dangerous in that the aircraft is nearer the stall than the ASI is indicating.
Note: that a climb with blocked static source and normal pitot air will result in the ASI under- reading. If the alternative static source is selected an error may occur. This error will be due to position error. Any dynamic, or turbulence, effects would usually result in a higher static pressure and thus produce an under-reading.
Leaks. Leaks can occur either inside or outside the pressure cabin. A leak in the pitot tube causes the ASI to under read. A leak in the static tube where the outside pressure is lower than static, some unpressurised

aircraft, will cause the ASI to over read. Where the outside air is higher than static, in a pressurized cabin the ASI will under read.

SERVICEABILITY CHECKS

The following checks of the ASI and pressure supply system should be made before flight:-

a) Pressure head cover(s) and static vent plug(s) removed and stowed aboard the aircraft.
b) Pitot tube(s), holes/slots in static head(s) and/or static vent(s) should be checked free
from obvious obstructions such as insects. c)Pitot head heater operative (if fitted).
d) Dial glass clean and undamaged.
e) The instrument should indicate airspeed in the correct sense shortly after starting the take-off run.

MACHMETER

HIGH SPEED FLIGHT.

In high speed aircraft the mach meter is an essential instrument. As an aircraft approaches the local speed of sound the airflow over some parts of the fuselage or wings may be accelerated up to the speed of sound and a shock wave will form. These shock waves cause more drag, less lift, buffeting, and reduction in effectiveness or even a reversal of control reactions. The speed at which airflow over some part of the aeroplane first reaches the speed of sound, and shock waves form, is called the critical Mach number, known as - Mcrit. Mcrit varies with aircraft design, typical values ranging from 0.7 to 0.9 Mach. Only aircraft which are designed for transonic or supersonic flight should closely approach or exceed Mcrit. The Mach meter therefore displays the present Mach Number so that the pilot can keep his Mach Number well below the particular Mach Crit for his aircraft and avoid the problems associated with high speed flight.

OPERATING LIMITS.

To limit an aircraft's Mach number to a safe speed below Mcrit operating limits are specified:
Mno = Maximum permitted Mach number for normal operation Mmo = Maximum permitted operating Mach number under any conditions

SPEED OF SOUND

The speed Of sound is not constant but varies with air temperature. A formula for calculating the local speed of sound (LSS) is:
LSS= 38.95T
LSS is given in Knots, 38.95 is a constant, and T is the absolute temperature, (O*C = 2730A = 2730 K)
Therefore the higher the air temperature, the higher the speed Of sound, and vice versa. Since temperature normally reduces as altitude increases, the speed of sound normally reduces as altitude increases. In ISA conditions at mean sea level (+ 15*C) the speed of sound is 661.32 knots, while at 30 000 feet ISA (- 45*C) the speed of sound will have reduced to 589.18 knots.

MACHMETER

MACHMETER PRINCIPLE OF OPERATION.

The mach meter uses two capsules and linkages to indicate the aircrafts True Air Speed (TAS) as a proportion of the local speed of sound (LSS) The first capsule is an Airspeed Capsule which will expand and contract as a result of changes in the Dynamic pressure.
Mach Number = TAS/LSS

The second capsule is a sealed Altimeter Capsule which will expand and contract as the static pressure inside the instrument case changes.
However, MN is proportional to. D/p*S/p
As Density (p) cancels out, we can see that:- MN is proportional to
= D/S = P-S/S

MACHMETER CONSTRUCTION.
As shown in figure of mach meter, it consists of a simple aneroid altitude capsule and an airspeed capsule which are connected at the ratio arm. Static pressure enters the case Of the instrument, while Pitot pressure is fed directly into the airspeed capsule. Expansion Or contraction Of the airspeed capsule is transmitted via the airspeed link and main shaft to the ratio arm, but the position of the ratio arm is also governed by expansion or contraction Of the altitude capsule. A spring-loaded ranging arm transmits the movement of the ratio arm to the pointer mechanism. Basically, if either or both capsules expand (due to an increase in IAS and / Or altitude) then the ranging arm will rotate out of the diagram and the indicated Mach Number will increase. If airspeed or altitude reduce then the ratio and ranging arms move back into the paper, and a lower Mach Number is displayed. An adjustable index on the instrument scale can be positioned by means Of a small knob. This index can be set to the limiting Mach Number for the aircraft type (in straight and level flight), to provide a visual warning to the pilot. It may be a wide index, so emphasizing the critical Mach number.
MACHMETER ERRORS.
The mach meter suffers from instrument, position and maneuver induced errors only. It does not suffer from temperature or density errors, as these errors cancel out. In addition since compressibility error depends on dynamic / static pressure, and the instrument is calibrated to this ratio, compressibility error is calibrated out.

Position Error. The instrument uses the same sources of Pitot and Static pressure as the ASI and therefore suffers from position error caused by disturbed airflow at the pitot head and / or static vent. At Mach Numbers below careful design and positioning of the pressure sources ensure that position error on modern jet aircraft is small. However, above changes in airflow may cause position error to become bigger and possibly change its sign. If the sign of the position error is such that the mach meter under-reads, the error could become dangerous at high Mach numbers. The normal arrangement in modern jet transport aircraft is to allow for instrument and position error such that the mach meter always over-reads.
Maneuver Induced Error. The mach meter will suffer an additional, unpredictable error whenever the aeroplane maneuvers. This is due to the unpredictable changes in the airflow over the static source.
BLOCKAGES.

Static Source Blocked. If the static source is blocked the pressure in the Machmeter case will not alter when the aircraft climbs or descends. If a blockage occurs in a climb the altitude capsule will not move. Assuming a constant IAS (and therefore a constant dynamic pressure) the airspeed capsule will contract as the static component of pitot pressure reduces. The Machmeter will therefore under read. If a blockage occurs in a descent at a constant IAS, the airspeed capsule will expand due to the increasing static component of pitot pressure. The Machmeter will therefore over read.
Pitot Source Blocked. Assuming a climb or descent at a constant IAS (and therefore a constant dynamic pressure) the Machmeter will over read in the climb and under read in the descent. In the climb the airspeed capsule will expand in error because the static component of pitot in the capsule will be greater than the static in the case. In the descent the static component of pitot will be too small and therefore the airspeed capsule will contract.
NOTE. It may be noticed that the Machmeter blockage errors are the same as the ASI
blockage errors.
MACH / AIRSPEED INDICATOR.

Since many commercial aircraft require indications of both IAS and Mach number, it is sensible to combine both instruments. The basic principals of both instruments still apply.

Errors

The combined instrument will have the errors of both the Mach meter and the airspeed indicator. namely; instrument, position, manoeuvre induced, density and compressibility errors.

CONSTRUCTION

There are two types of Mach/Airspeed Indicator:

a) A self contained instrument fed from Pitot and Static sources.

b) A combined instrument fed from the Air Data Computer.

c) A second striped needle may be present to mark V

Note that:

i) The airspeed pointer moves clockwise over a fixed scale.

ii) From 0.5 M the Mach number is read off the same pointer as it moves over a moving Mach number scale. This scale rotates anti- clockwise beneath the pointer as Mach number increases.

ALTITUTE REPORTING AND ALERTING SYSTEM

Altitude Alerting Systems

In certain aircraft systems the control and operating conditions are related to one specific altitude; for example, in a cabin- pressurization system, the necessity arises for an indication of a possible increase of cabin altitude above the desired level while the aircraft is at its normal operating altitude. Furthermore, and particularly as a result of the introduction of altitude reporting systems, it is necessary for a pilot to be warned of an approach to, and/or deviation from, a selected operational altitude. To cater for the appropriate requirements therefore, it is usual to employ altitude switching units or servo type altimeters capable of transmitting altitude signals to a separate alerting unit via a synchronous transmission link.

Altitude switching units normally consist of an aneroid-capsule measuring element similar to that used in altimeters, but in lieu of a pointer actuating mechanism, the capsule is so designed that at a preset altitude, its expansion actuates an electrical contact assembly
such as to complete a circuit to a warning light or aural warning device.

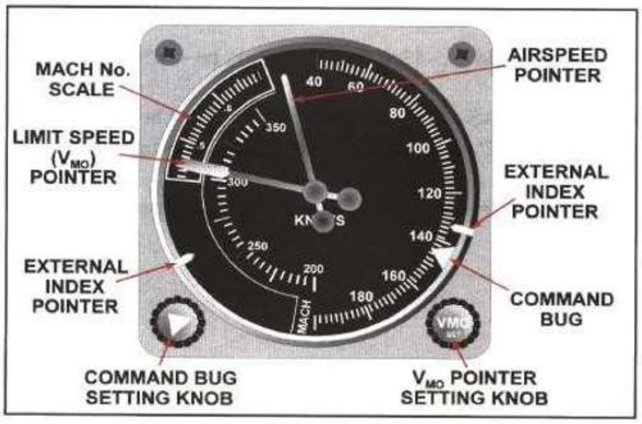
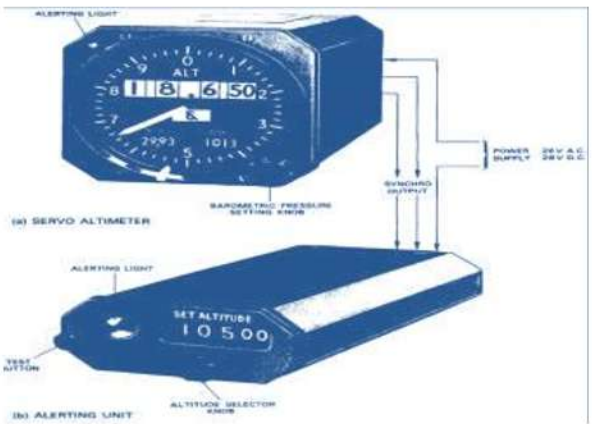
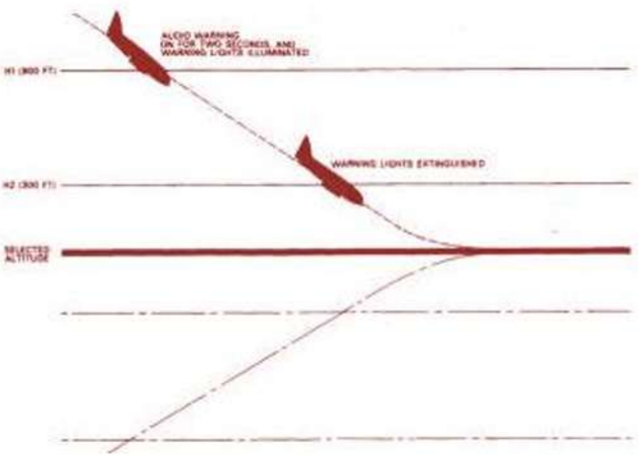


Figure 7.7. A Mach / Airspeed Indicator Fed from Pitot and Static Sources.



The components of an altitude alerting system designed to give audio and visual warnings when an aircraft approaches or deviates from a pre-selected altitude by more than a pre-determined amount. The altitude is selected on the alerting unit, and is indicated by a digital counter which is geared to the rotors of a control transformer (CT) synchro, and a control transformer/ resolver synchro or transolver as it is called. The synchros are referred to as coarse and fine synchros respectively, and are electrically connected to corresponding transmitter synchros within the servo altimeter.
The rotors of the transmitter synchros are mechanically positioned by a linkage system coupled to the altimeter capsule assembly so that the output from the synchros is proportional to the aircraft's altitude. When an altitude is selected on the alerting unit, the selector knob, in addition to rotating the digital counters, also rotates the rotors of the unit synchros thereby developing a signal corresponding to the difference between the indicated and the selected altitudes. This signal difference is supplied to an input section of the overall circuit of the alerting unit, and at pre- determined values of both synchro rotor voltages, two signals are produced and are supplied as inputs to a logic circuit. The logic circuit comprises a timing network which controls a remote audio warning device and the operation of the warning lights in the semo altimeter and altitude alerting unit.
As the aircraft descends or climbs to the pre-selected altitude, the difference signal referred to earlier is reduced, and the logic circuit so processes the two signals supplied to it, that at a preset outer limit H1 (typically 900 ft) above or below the pre-selected altitude, one signal actuates the audio warning device which remains on for two seconds, and also illuminates the warning lights.
The lights remain on until at a further pre-set inner limit H2 (typically 300 ft) above or below the pre-selected altitude, the second signal causes the circuit to the warning lights to be interrupted thereby extinguishin g them. As the aircraft approaches the pre- selected altitude the rotor voltages of the synchros approach their null and no further warnings are given. If the aircraft should subsequently depart from the pre-selected altitude by more than the inner limit H2, the logic circuit changes the alerting sequence such that the warnings correspond to those given during the approach through outer limit HI, i.e. audio on for two seconds and warning light illumination.
A 'reset' circuit is incorporated in the alerting unit, its function being to reset the logic circuit whenever the selector knob is operated to change the selected altitude by more than 100 ft and at



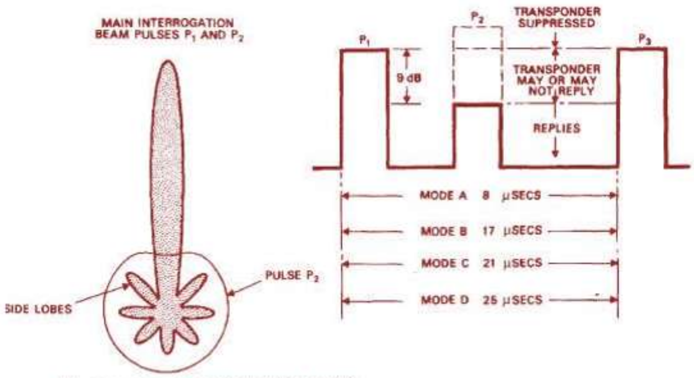
a rate greater than 8,000 ft per minute. The circuit utilizes a photoelectric cell which produces a signal of sufficient magnitude to override any signal present at the output of the logic circuit.
Functional testing of the audio and visual warning system is accomplished by operating a test switch on the alerting unit while rotating the altitude selector knob. In the event of failure of the power supply to the system (26 V a.c. 400 Hz and 28 V d.c.) and also of altitude signals, a solenoid is de-energized to actuate a warning flag which obscures the altitude counters of the alerting unit.

Altitude Reporting System

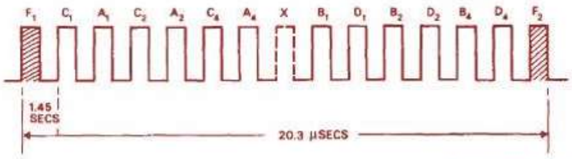
The control of air traffic along the many air corridors in the vicinity of major airports is dependent on rigid procedures for communication between individual aircraft and ground control stations in order that traffic may be identified and assigned to requisite separation levels. In addition to normal voice transmissions, the communication procedure involves the use of an airborne transponder which, in response to interrogation signals from a radar transmitter/receiver at the air traffic control centre, automatically transmits coded reply signals to the centre. The signals are then computer-processed, decoded, and then alpha- numerically displayed to the air traffic controller on his primary radar screen.

Aircraft altitude is one of the important parameters required to be known, and to further reduce time-consuming voice transmissions, a method of automatically transmitting such data from an altimeter was devised and also became a mandatory feature of the air to ground communication procedure. The interrogation system as a whole forms what is termed Air Traffic Control Secondary Surveillance Radar, and can operate in four modes of interrogation: A, B, C and D. Modes A and B are used for identification, Mode C for altitude reporting, while Mode D is at present unassigned. In each case the interrogating signal, which is transmitted on a frequency of 1,030 MHz from a rotating directional antenna, comprises a pair of pulses P1 and P3, and in order that the airborne transponder can 'recognize' in which mode it is being interrogated, the pulses are spaced at different time intervals. The spacing is taken from the leading edge of the first pulse to the leading edge of the second. It will be noted from diagram (a) that a third pulse P2 can also be transmitted from a control antenna; its purpose is to suppress side lobe radiation from the interrogator antenna, and to ensure that the transponder replies only to the main beam directional signal pulses. This is effected by a 'gating' circuit which compares the relative amplitudes of the pulses and enables the transponder to determine whether the interrogation is a correct one, or due to a side lobe.When the transponder decodes the interrogating signal it will reply by transmitting a train of information pulses on a frequency of 1,090 MHz, and in a coded sequence which depends not only on the interrogation mode, but also on pre-allocated code nulribers which, for operation in Modes A and B, are selected by the pilot on the transponder control unit. In Mode C operation, code numbers are automatically transmitted by the transponder which also receives signals corresponding to specific altitudes from an altimeter The train of information pulses from the transponder may consist of up to twelve pulses spaced 1 -45 p S apart, depending on the reply code selected on the control unit (diagram (b) Fig 4.24). The pulses lie between two additional framing pulses F1 and F2, which are fixed at a spacing of 20.3 PS, and which are always transmitted in the reply.

In a twelve-pulse train the number of codes available is 212 = 4,096 the codes being numbered 0000 through 7777, the latter giving all twelve pulses when four selector knobs of the control unit are correspondingly set. Each control knob controls a group of three pulses as shown; the letters in this case designate pulse groups and not interrogation modes. The first control knob controls the A group, the second knob the B group, and so on. The subscripts to each letter of a group are significant since their sum equals the digit selected on the control unit, and this may be seen pulse train in this example would consist of only three pulses spaced between the framing pulses at the intervals indicated in thediagram.



(a) INTERROGATION BEAM AND PULSE SPACING



(b) REPLY PULSE TRAIN

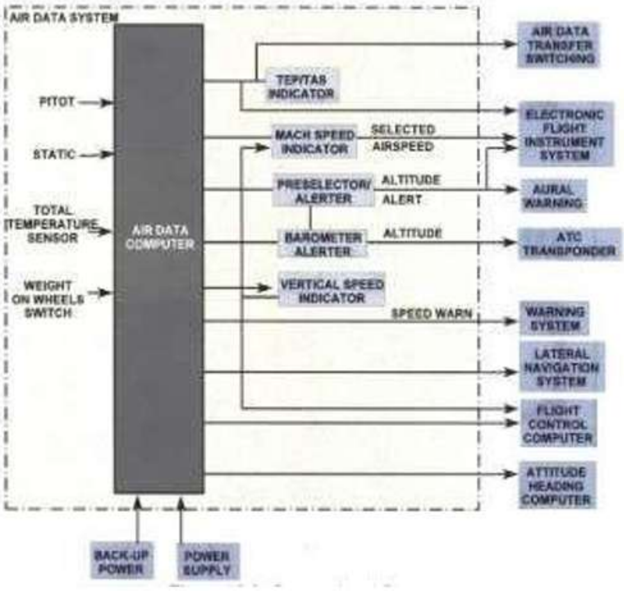
X PULSE NOT USED IN PRESENT CODING

AIR DATA COMPUTER

INTRODUCTION

from the basic code table. Since the whole system is based on a digital computing process, then the encoding and decoding of interrogation and reply signals is dependent on logic variables and corresponding binary digits or 'bits' as they are termed. In the example shown at (c) the code 2300 in Mode A has been selected on the control unit and this produces the equivalent binaries 0 1 0, 1 1 0,0 0 0 and 0 0 0 respectively. Since in logic networks 0 signifies the absence of a signal, and 1 the presence of a signal, then the selection of the digit 2 causes only pulse 2 of the A group to be transmitted, while the selection of digit 3 causes the transmission of pulse 1 plus pulse 2 of the B group. Thus, the reply

The pressures on which the operation of the primary flight instruments is dependent are transmitted through a system of pipelines. It should be apparent therefore, that the length and quantity of the pipelines will vary according to the size of the aircraft, and also on the number of stations at which indications of the relevant air data are required. In order to minimize the 'plumbing' arrangements, the concept of supplying the pressures to a special unit at some centralized location, and then transmitting the air data electrically to wherever required, was developed and resulted in the design of units designated as central aid data computers (CADC). The modular arrangement of a computer, and adopted by any one manufacturer.

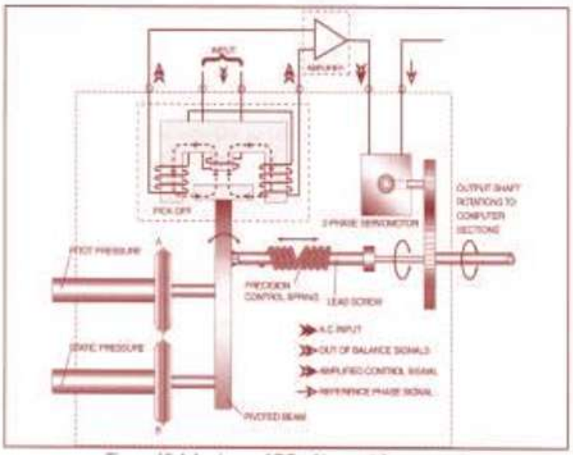


the methods by which signal processing is carried out, can vary dependent on the number of parameters to be monitored, and also on the techniques

An example of a pitot-static pressure transducer utilizing an 'E' and 'I' bar type of inductive pick-off unit. The transducer operates on what is termed the force-balance principle, and comprises two capsules, the interior of one being connected to the pitot pressure source, while the other is connected to the static pressures source. Both capsules are connected to a pivoted beam which, in turn, is connected to the 'I' bar of the pickoff unit.
When a change in airspeed takes place, the capsules respond to the corresponding change in differential pressure and the force they produce deflects the pivoted beam thereby displacing the 'I' bar relative to the limbs of the 'E' bar. Thus, the air gaps are varied to cause out-of- balance signals to be induced in the outer limb coils in

Basically however,(a computer is an analogue device that produces electrical signal equivalents of pitot and static pressures by the combined operation of mechanical and synchronous transmission devices. The final computed output signals are then supplied to the appropriate indicators which, unlike their conventional counterparts, contain no pressure-sensing elements Pressure sensing is accomplished by two pressure transducers, one sensing static pressure within the altitude module, while the other senses both pitot and static pressures within the computed airspeed (CAS) module. The Mach speed module and true airspeed (TAS) module are pure signal-generating devices, which are supplied with altitude and airspeed signal data from the respective modules. Static air temperature data required for TAS computation is sensed by a probe located outside the aircraft at some predetermined position, and is routed through the Mach speed module.

the same manner as those induced in the servo altimeter pick-off described earlier.
The signals are amplified and applied to the control phase of the servomotor which drives an output shaft and a lead screw. The lead screw is coupled to the pivoted beam via a precision control spring so that as the screw rotates the spring tension is varied, to balance the force exerted on the beam, and therefore, to start 'backing-off' the signal induced in the appropriate outer limb coil. When a constant speed condition is attained, equilibrium between capsule force and spring tension is established, no further signals are fed to the amplifier and the servomotor ceases to rotate. Since the output shaft is also rotated by the servomotor then, by coupling the shaft to a CX synchro, its angular position can be measured in terms of the pressure differential p - s applied to the transducer, and hence in terms of airspeed.



In the case of an altitude module which employs a force-balance transducer, the construction is the same as that of an airspeed module, except that in place of the pitot pressure capsule, an evacuated and sealed capsule is employed.

In some types of CADC altitude and airspeed transducers, pressure sensing and transmission of corresponding signals is accomplished by means of a silicon diaphragm containing piezoelectric elements. The elements function in a similar manner to a strain gauge, in that their resistance changes as a function of the strain imposed on the diaphragm under the influence of pressure. The resulting voltage signal outputs are fed via integrated circuits to the associated indicators.

BUILT IN TEST EQUIPMENT (BIT OR BITE).

There is no provision made for the manual input of data into the ADC in the event of any failure, but the Built In Test Equipment will give prompt indication of any malfunction that might occur.

In any ADC there will be three types of BITE process:

a) Power Up BITE.

This functions when power is applied to the ADC on start up or after a break. A check is made on the Microprocessor, the Memory Store and the Air Data functions.

b) Continuous BITE.

This is an automatic check Of all stages of input and output carried out throughout the operation of the ADC about once every second. c)Maintenance BITE.

This enables maintenance crew to carry out checks on the ground using a Test or Test/History switch (current or post failures).

ADVANTAGES OF AN AIR DATA SYSTEM.

An ADS has certain advantages when compared with conventional mechanical instruments:

a) Improved Displays. Electrically-servoed instrumentation allows the manufacturer complete freedom to design new displays that are easier to read and unambiguous. These include digital, moving tape and combined displays.

b) Reduced Instrument and Lag Errors. The major cause of instrument error in conventional mechanical instruments is friction loss within the linkage. The limited response rate of such linkages gives rise to lag error. Both problems are largely overcome with ADS's by the use of servomotors.

c) Error Correction. Computation of height, airspeed and other variables within one Computer permits error corrections to be applied through especially shaped cams appropriate to the particular aircraft. For example, position error correction (PEC) can be calculated within the Mach No computer channel for additional use within the height and airspeed channels.

d) Central Source for Other Systems. The ADC provides not only the conventional information displayed on the instrument panel but also air data in many forms as required for Other Systems.

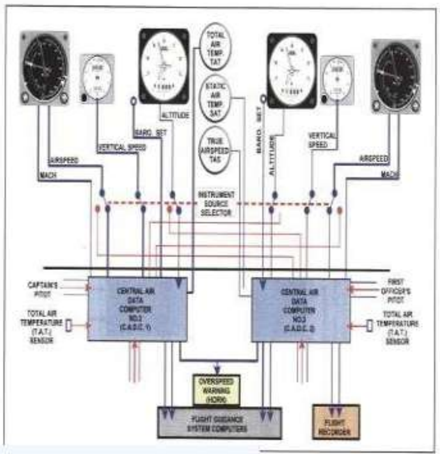
e) Clean Design. The use of electrically-driven instruments reduces the amount of pneumatic plumbing required behind the instrument panel to only those lines connected to the standby airspeed indicator and altimeter. In addition to space saving and easier maintenance, the use of shorter pitot/static line reduces error-producing acoustic effects.

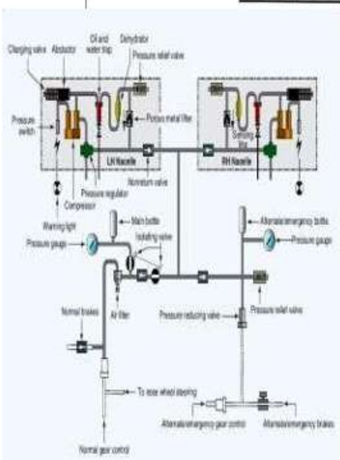
Failure Warning. A comparison monitor can be incorporated to compare the outputs of the ADC's and to give automatic warning to the pilot of malfunction. With a purely mechanical system, comparison between left-hand and right-hand instruments must be carried out visually. A warning flag will appear on the appropriate ADS instrument if there is loss of valid data or if an internal failure occurs. In addition, a light will illuminate either on the instrument warning panel Or on the central warning System indicator
Instrument pneumatic system

Some aircraft manufacturers have equipped their aircraft with a high pressure pneumatic system (3,000 psi) in the past. The last aircraft to utilize this type of system was the Fokker F27. Such systems operate a great deal like hydraulic systems, except they employ air instead of a liquid for transmitting power. Pneumatic systems are sometimes used for:

- Brakes
- Opening and closing doors
- Driving hydraulic pumps, alternators, starters, water injection pumps, etc.
- Operating emergency devices

Both pneumatic and hydraulic systems are similar units and use confined fluids. The word confined means trapped or completely enclosed. The word fluid implies such liquids
as water, oil, or anything that flows. Since both liquids and gases flow, they are considered as fluids; however, there is a great deal of difference in the characteristics of the two. Liquids are practically incompressible; a quart of water still occupies about a quart of space regardless of how hard it is compressed. But gases are highly compressible; a quart of air can be compressed into a thimbleful of space. In spite of this difference, gases and liquids are both fluids and can be confined and made to transmit power.





The type of unit used to provide pressurized air for pneumatic systems is deter mined by the system's air pressure requirements.

The other valve is a control valve. It acts as a shutoff valve, keeping air trapped inside the bottle until the system is operated. Although the high pressure storage cylinder is light in weight, it has a definite disadvantage. Since the system cannot be recharged during flight, operation is limited by the small supply of bottled air. Such an arrangement cannot be used for the continuous operation of a system. Instead, the supply of bottled air is reserved for emergency operation of such systems as the landing gear or brakes. The usefulness of this type of system is increased, however, if other air-pressurizing units are added to the aircraft.

Pneumatic System Components

Pneumatic systems are often compared to hydraulic systems, but such comparisons can only hold true in general terms. Pneumatic systems do not utilize reservoirs, hand pumps, accumulators, regulators, or engine-driven or electrically driven power pumps for building normal pressure. But similarities do exist in some components.

Air Compressors

On some aircraft, permanently installed air compressors have been added to recharge air bottles whenever pressure is used for operating a unit. Several types of compressors are used for this purpose. Some have two stages of compression, while others have three, depending on the maximum desired operating pressure.

Pneumatic brake system

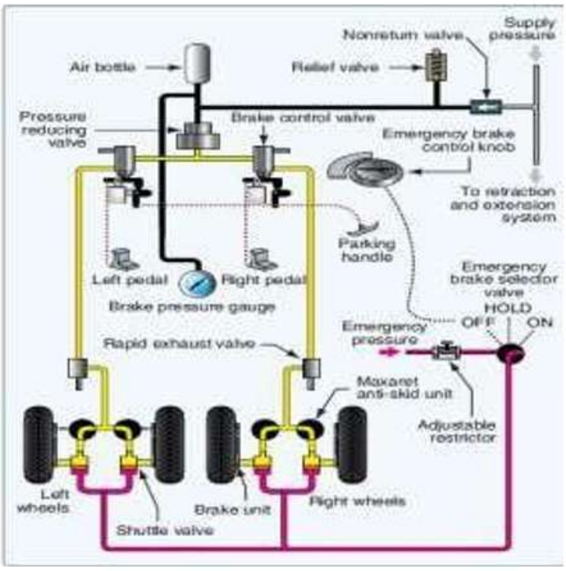
Relief Valves

Relief valves are used in pneumatic systems to prevent damage. They act as pressure limiting units and prevent excessive pressures from bursting lines and blowing out seals.

Control Valves

Control valves are also a necessary part of a typical pneumatic system. Above figure illustrates how a valve is used to control emergency air brakes. The control valve consists of a three-port housing, two poppet valves, and a control lever with two lobes.

In Figure, the control valve is shown in the off position. A spring holds the left poppet closed so that compressed air entering the pressure port cannot flow to the brakes. In Figure the control valve has been placed in the on position. One lobe of the lever holds the left poppet open, and a spring closes the right poppet. Compressed air now flows around the opened left poppet, through a drilled passage, and into a chamber below the right poppet. Since the right poppet is closed, the high-pressure air flows out of the brake port and into the brake line to apply the brakes. To release the brakes, the control valve is returned to the off position. The left poppet now closes, stopping the flow of high-pressure air to the brakes. At the same time, the right poppet is opened, allowing compressed air in the brake line to exhaust through the vent port and into the atmosphere.

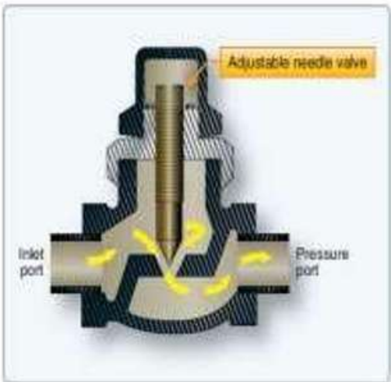


Check Valves
Check valves are used in both hydraulic and pneumatic systems. A flap-type pneumatic check valve. Air enters the left port of the check valve, compresses a light spring, forcing the check valve open and allowing air to flow out the right port. But if air enters from the right, air pressure closes the valve, preventing a flow of air out the left port. Thus, a pneumatic check valve is a one- direction flow control valve. Flap-type pneumatic check valve.

Restrictors
Restrictors are a type of control valve used in pneumatic systems, an orifice-type restrictor with a large inlet port and a small outlet port. The small outlet port reduces the rate of airflow and the speed of operation of an actuating unit.

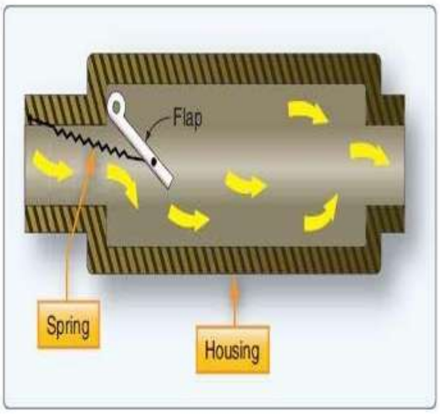
Pneumatic orifice valve.

Variable Restrictor
Another type of speed-regulating unit is the variable restrictor. It contains an adjustable needle valve, which has threads around the top and a point on the lower end. Depending on the direction turned, the needle valve moves the sharp point either into or out of a small opening to decrease or increase the size of the opening. Since air entering the inlet port must pass through this opening before reaching the outlet port, this adjustment also determines the rate of airflow through the restrictor.



Variable pneumatic restrictor.

Filters
Pneumatic systems are protected against dirt by means of various types of filters. A micronic filter consists of a housing with two ports, a replaceable cartridge, and a relief valve. Normally, air enters the inlet, circulates around the cellulose cartridge, and flows to the center of the cartridge and out the outlet port. If the cartridge



becomes clogged with dirt, pressure forces the relief valve open and allows unfiltered air to flow out the outlet port. A screen-type filter is similar to the micron filter but contains a permanent wire screen instead of a replaceable cartridge. In the
screen filter, a handle extends through the top of the housing and can be used to clean the screen by rotating it against metal scrapers.

Desiccant/Moisture Separator
The moisture separator in a pneumatic system is always located downstream of the compressor. Its purpose is to remove any moisture caused by the compressor. A complete moisture separator
consists of a reservoir, a pressure switch, a dump valve, and a check valve. It may also include a regulator and a relief valve. The dump valve is energized and de-energized by the pressure switch. When de-energized, it completely purges the separator reservoir and lines up to the compressor. The check valve protects the system against pressure loss during the dumping cycle and prevents reverse flow through the separator.

Chemical Drier
Chemical driers are incorporated at various locations in a pneumatic system. Their purpose is to absorb any moisture that may collect in the lines and other parts of the system. Each drier contains a cartridge that should be blue in color. If otherwise noted, the cartridge is to be considered contaminated with moisture and should be replaced.

Emergency Backup Systems
Many aircraft use a high-pressure pneumatic back-up source of power to extend the landing gear or actuate the brakes, if the main hydraulic braking system fails. The nitrogen is not directly used to actuate the landing gear actuators or brake units but, instead, it applies the pressurized nitrogen to move hydraulic fluid to the actuator. This process is called pneudraulics. The following paragraph discusses the components and operation of an emergency pneumatic landing gear extension system used on a business jet.

Nitrogen Bottles
Nitrogen used for emergency landing gear extension is stored in two bottles, one bottle located on each side of the nose wheel well. Nitrogen from the bottles is released by actuation of an outlet valve. Once depleted, the bottles must be recharged by maintenance personnel. Fully serviced pressure is approximately

3,100 psi at 70 °F/21 °C, enough for only one extension of the landing gear.

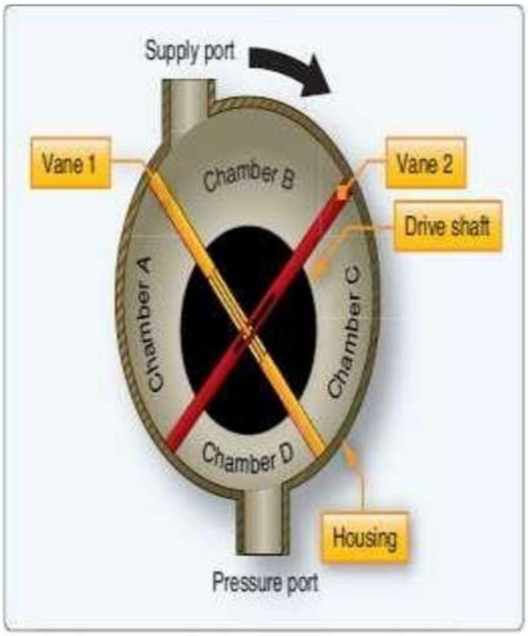
Gear Emergency Extension Cable and Handle
The outlet valve is connected to a cable and handle assembly. The handle is located on the side of the copilot‘s console and is labeled EMER LDG GEAR. Pulling the handle fully upward opens the outlet valve, releasing compressed nitrogen into the landing gear extension system. Pushing the handle fully downward closes the outlet valve and allows any nitrogen present in the emergency landing gear extension system to be vented overboard. The venting process takes approximately 30 seconds.

Dump Valve
As compressed nitrogen is released to the landing gear selector/dump valve during emergency extension, the

pneudraulic pressure actuates the dump valve portion of the landing gear selector/dump valve to isolate the landing gear system from the remainder of hydraulic system. When activated, a blue DUMP legend is illuminated on the LDG GR DUMP V switch, located on the cockpit overhead panel. A dump valve reset switch is used to reset the dump valve after the system has been used and serviced.

Emergency Extension Sequence:
1. Landing gear handle is placed in the DOWN position.
2. Red light in the landing gear control handle is illuminated.
3. EMER LDG GEAR handle is pulled fully outward.
4. Compressed nitrogen is released to the landing gear selector/dump valve.
5. Pneudraulic pressure actuates the dump valve portion of the landing gear selector/dump valve.
6. Blue DUMP legend is illuminated on the LDG GR DUMP switch.
7. Landing gear system is isolated from the remainder of hydraulic system.
8. Pneudraulic pressure is routed to the OPEN side of the landing gear door actuators, the UNLOCK side of the landing gear uplock actuators, and the EXTEND side of the main landing gear sidebrace actuators and nose landing gear extend/retract actuator.
9. Landing gear doors open.
10. Uplock actuators unlock.
11. Landing gear extends down and locks.
12. Three green DOWN AND LOCKED lights on the landing gear control panel are illuminated.
13. Landing gear doors remain open.

Medium-Pressure Systems
A medium-pressure pneumatic system (50–150 psi) usually does not include an air bottle. Instead, it generally draws air from the compressor section of a
turbine engine. This process is often called bleed air and is used to provide pneumatic power for engine starts, engine deicing, wing deicing, and in some cases, it provides hydraulic power to the aircraft systems (if the hydraulic system is equipped with an air- driven hydraulic pump). Engine bleed air is also used to pressurize the reservoirs ofv the hydraulic system.



Low-Pressure Systems
Many aircraft equipped with reciprocating engines obtain a supply of low-pressure air from vane-type pumps. These pumps are driven by electric motors or by the aircraft engine. which consists of a housing with two ports, a drive shaft, and two vanes. The drive shaft and the vanes contain slots so the vanes can

slide back and forth through the drive shaft. The shaft is eccentrically mounted in the housing, causing the vanes to form four different sizes of chambers (A, B, C, and D). In the position shown, B is the largest chamber and is connected to the supply port. As depicted in Figure outside air can enter chamber B of the pump. When the pump begins to operate, the drive shaft rotates and changes positions of the vanes and sizes of the chambers. Vane No. 1 then moves to the right, separating chamber B from the supply port. Chamber B now contains trapped air.

As the shaft continues to turn, chamber B moves downward and becomes increasingly smaller, gradually compressing its air. Near the bottom of the pump, chamber B connects to the pressure port and sends compressed air into the pressure line. Then chamber B moves upward again becoming increasingly larger in area. At the supply port, it receives another supply of air. There are four such chambers in this pump and each goes through this same cycle of operation. Thus, the pump delivers to the pneumatic system a continuous supply of compressed air from 1 to 10 psi. Low- pressure systems are used for wing deicing boot systems.

Pneumatic Power System Maintenance

Maintenance of the pneumatic power system consists of servicing, troubleshooting, removal, and installation of components, and operational testing. The air compressor‘s lubricating oil level should be checked daily in accordance with the applicable manufacturer‘s instructions. The oil level is indicated by means of a sight gauge or dipstick. When refilling the compressor oil tank, the oil (type specified in the applicable instructions manual) is added until the specified level. After the oil is added, ensure that the filler plug is torqued and safety wire is properly installed. The pneumatic system should be purged periodically to remove the contamination, moisture, or oil from the components and lines. Purging the system is accomplished by pressurizing it and removing the plumbing from various components throughout the system. Removal of the pressurized lines causes a high rate of airflow through the system, causing foreign matter to be exhausted from the system. If an excessive amount of foreign matter, particularly oil, is exhausted from any one system, the lines and components should be removed and cleaned or replaced. Upon completion of pneumatic system purging and after reconnecting all the system components, the system air bottles should be drained to exhaust any moisture or impurities that may have accumulated there. After draining the air bottles, service the system with nitrogen or clean, dry compressed air. The system should then be given a thorough operational check and an inspection for leaks and security.

DIRECT READING PRESSURE AND TEMPERATURE GAUGE

TYPES OF DISPLAY

In the former, there are a multitude of gauges displaying information, in the latter the display is shown on cathode ray tubes (C.R.T.s) or liquid crystal display panels (L.C.D.s) with light emitting diodes (L.E.D.s) for digital displays. A small number of conventional gauges are retained in case of failure Of the electronic displays. Both types of display convey essentially the same information to the pilot, but the flexibility of the Glass Cockpit system means that it is now taking over as the preferred means of showing both flight and engine instrumentation.

THRUST AND POWER MEASURI NG INSTRUM ENTS.

Thrust measuring instruments are of two basic types:- a)the type that measures the jet pipe pressure

b) the type that measures the ratio of two parameters, the jet pipe pressure and the engine air intake pressure

c) propeller driven aircraft measurer and indicate Torque. This is an indication of engine power. The propeller converts power into thrust.

On some large turbo-fan engines the integrated turbine discharge pressure and fan outlet pressure is compared to the compressor inlet pressure to produce what is called 'integrated' E.P.R.. Pitot tubes, suitably positioned, sense the pressures which are required to work the system, the tubes can either be connected directly to the indicator in the cockpit or to a pressure transmitter which is electrically connected to the indicator. The gauge can be marked in inches of mercury (in Hg), pounds per square inch (p.s.i.), or a percentage of the engine's maximum thrust. Although E.P.R. can be indicated by either mechanical Or electronic means, it is more normal to find the electronic System in use. This system uses two transducers which sense the relevant air pressures and vibrate at

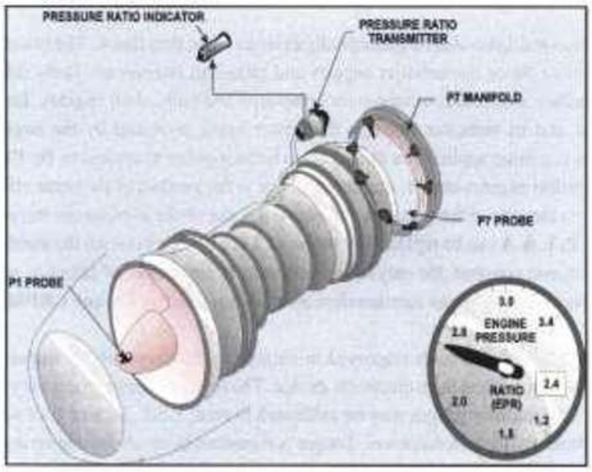


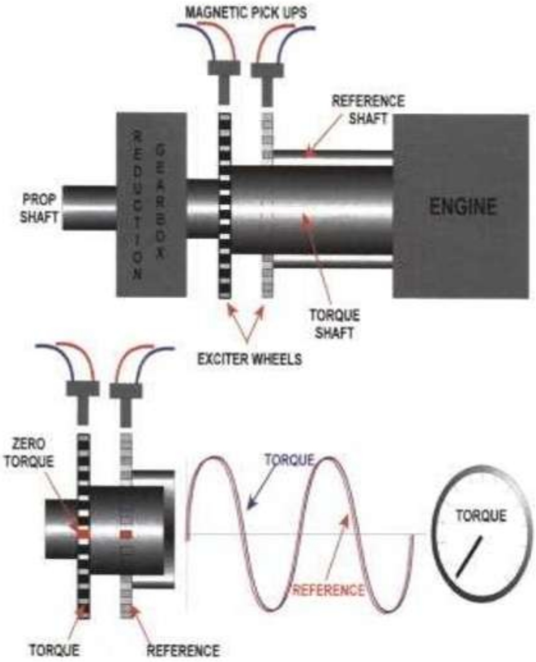
Figure 1.2 The Engine Pressure Ratio (E.P.R.) Indicating System.

frequencies proportional to these pressures. A computer works out the electrical Signal appropriate to the pressures and that signal is sent to the E.P.R. gauge in the cockpit and to the engine management system. Although E.P.R. can be indicated by either mechanical or electronic means, it is more normal to find the electronic System in use. This system uses two transducers which sense the relevant air pressures and vibrate at frequencies proportional to these pressures. A computer works out the electrical signal appropriate to the pressures and that signal is sent to the E.P.R. gauge in the cockpit and to the engine management system.

The engine intake pressure will vary with changing ambient pressure and also with changing airspeed. An increase in forward airspeed during take-off will cause a drop in the reading on the E.P.R. gauge. This is only an 'apparent drop' because it is only the ratio of two pressures, engine intake pressure and the jet pipe pressure, which is changing. This apparent change is caused by a relative increase of the engine intake pressure with forward airspeed during take off. The jet pipe pressure being unaffected at low airspeeds, which causes the ratio between the two pressures to fall. This will be seen as a reduction in E.P.R. on the gauge, which might cause the inexperienced engine operator to Open the throttles further in an attempt to restore the loss, having eyes for only the one parameter, while the other parameters, N I , N2, N3 & E.G.T are in danger of exceeding their limits without his knowledge. In an attempt to prevent this happening, most operators require that E.P.R. is set before the aircraft has reached approximately 60 knots, no increase in engine power being allowed unless in emergency after this speed. After take off, as the airspeed increases beyond V2, the increase in engine intake pressure is passed through the engine to the jet pipe, so changing the ratio back to that set on take-off.

ENGINE TORQUE.

Turbo-props and turbo-shaft engines produce torque rather than thrust. The systems that produce indications of thrust for turbo-jet engines and turbo-fan engines are vastly different to those which produce indications of torque for turbo-prop and turbo-shaft engines. The Torque meter measures, and its indicator displays the power being produced by the engine. Torque by definition is a force applied at a distance to a turning point. If applied to the PLANE formula, the turning force is the product Of the mean effective pressure P acting on the area of the piston A at distance L (the stroke is twice the throw of the crank) Therefore P, L & A can be replaced by the word Torque. N represents the number of cylinders and will remain constant, the only other variable is E the number of effective power strokes or RPM of the engine. Power can therefore also be expressed as Torque x RPM. There are two main methods employed in measuring the torque of the engine. One uses oil pressure and the second is an electronic device. The units Of measurem ent vary from system to system. The indicator gauges may be calibrated to read, P.S.I., Inch or Foot pounds,



Newton metres, Brake or Shaft Horsepower. Torque is measured between the engine and the reduction gearbox. As the gears in the propeller reduction gearbox rotate to drive the propeller, the amount of torque that they are transmitting attempts to move them axially, this axial force is proportional to the torque that is producing it. The gears cannot be allowed to move axially because this would cause the teeth to no longer mesh with each other and the drive would fail, the axial force has to be counteracted to maintain the gears in alignment. The force comes from passing engine oil through a filter and then to a torque meter pump which enables its pressure to be boosted to (in some cases) as much as 800 p.s.i.. This high pressure is allowed into cylinders which form the bearings within which the helical gear shafts rotate. A small bleed hole in the wall of the cylinder will be covered by the gear shaft if it moves into the cylinder under increasing axial load, this will cause the oil pressure within the cylinder to build up until it can move the gear shaft back to its original position. Conversely, if the load on the helical gear shaft decreases, the existing oil pressure will force its shaft slightly out of the cylinder. This uncovers the bleed hole allowing the balancing Oil pressure to be reduced and so the gear shaft moves back into correct position within the cylinder. If the oil pressure balancing the axial force is measured, which take into account the ambient pressure and temperature and the performance of the engine, its power output, can be judged.

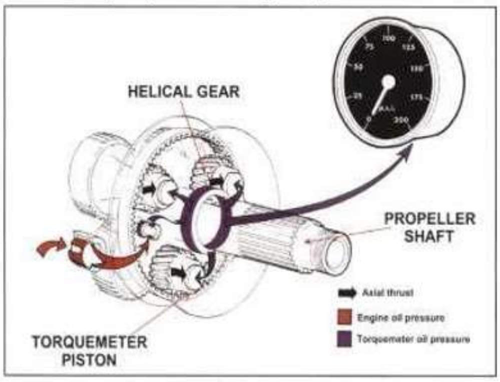


Figure 1.3. The Torquemeter System

The Electronic system comprises of two concentric shafts. One, the Torque Shaft, is connected to both the engine and the propeller's reduction gear box. The second shaft, the Reference Shaft, is connected only to the engine. An exciter wheel toothed gear formed at the forward end of each shaft. The exciter wheels rotate past an electro magnetic pick-up and produce an AC voltage. The exciter wheels are aligned at assembly, but as power is increased the torque shaft twists, this displaces the phase relationship of the voltages produced. The displacement is proportional to the change in power, and is used to drive an indicator. This system is simple, and lighter than other systems and has proven to be very reliable in service. The torque indicator may indicate negative (wind milling propeller) as well as positive torque. The torque limits are colour coded and shown on the gauge. A red coloured band or marker indicating maximum limits. On a FADEC system these limits may be adjusted and set by the crew, the indication can be presented in a digital readout.

ENGINE R.P.M.

The measurement of engine speed is of vital importance, since together with other parameters accurate control, and monitoring of the engine can be achieved. On piston engines it is crankshaft speed that is measured, whilst On gas turbine engines it is the speed Of the compressor. The RPM indicator is called a Tachometer (Tacho) There are three basic methods of measuring engine rotational speeds:-

1. Mechanical (Magnetic) Tachometer.
2. Electrical Generator System. (Tacho Gen)
3. Inductive Probe System.

There are no firm guidelines as to the application of each of the tachometer system, although engine and aircraft design will dictate which System Can be best utilized. The Mechanical Tachometer is now only found on older piston aircraft. It consists of a Flexible Drive Shaft that is connected to the flight deck Tacho Indicator. The input drive causes a magnet in the indicator to rotate. The magnet rotates inside a copper or aluminium drag- cup, this induces Eddy Currents in the drag-cup which opposes the magnetic field of the magnet. A torque is established which turns the drag -cup in the same direction as the permanent magnet. A shaft extends from the drag-cup and is connected to a pointer. The turning motion of the pointer is against the tension Of a Hairspring which controls the drag cup position and hence the position of the pointer. The flexible drive is driven at reduced speed, but true speed will be shown on the indicator. The indicator incorporates compensation devices for change in temperature.

The Electrical Generator System is possibly the oldest form of engine speed measurement still in use on large aircraft. It utilises a small three phase (tacho) generator, driven by the engine. The output of this generator is then taken to an indicator which consists of a synchronous motor turning a drag Cup assembly which moves a pointer over a scale as in the

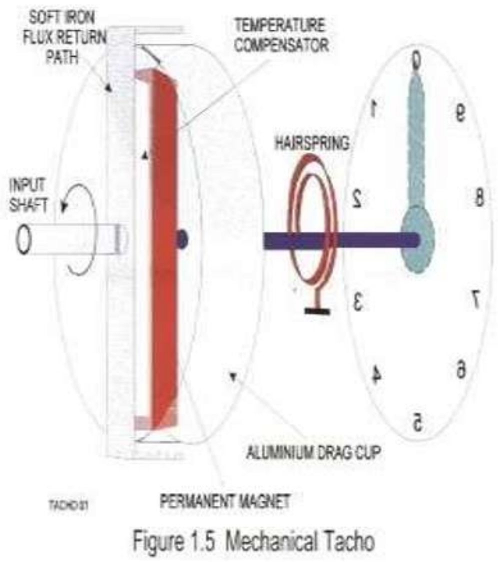


Figure 1.5 Mechanical Tacho

mechanical system.
Tacho Gen
The indicator can either show the actual revolutions per minute (not too common), or the

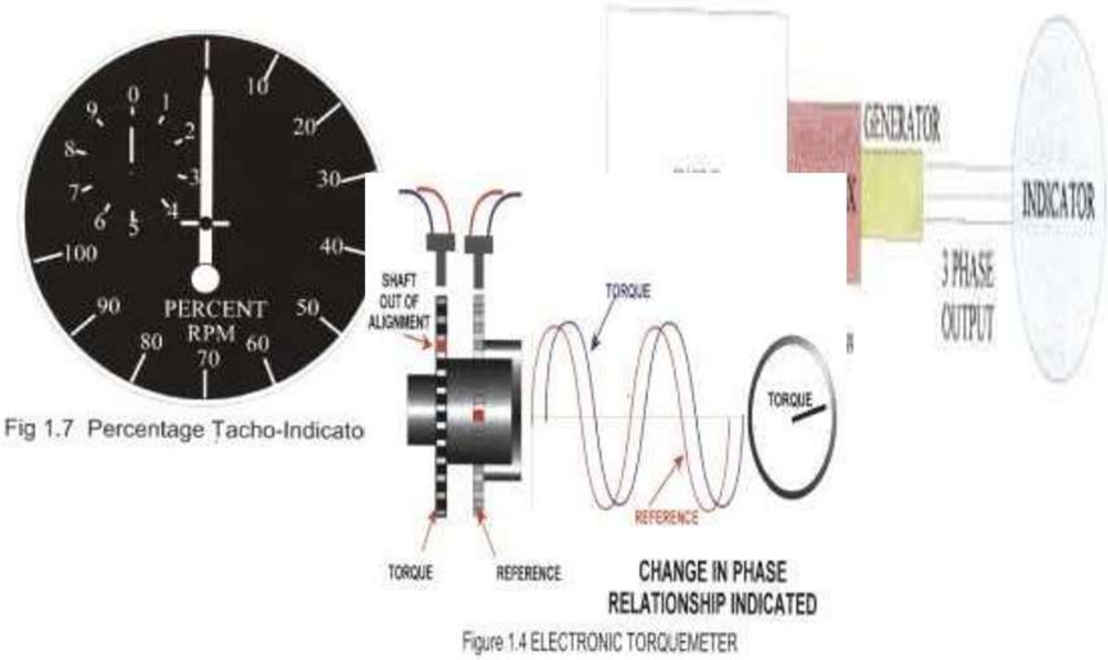


Fig 1.7 Percentage Tacho-Indicato
Figure 1.4 ELECTRONIC TORQUEMETER

speed as a percentage of maximum engine speed.

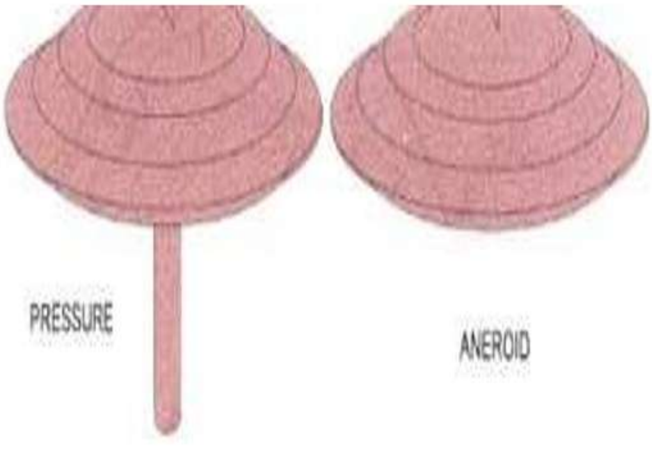
TEMPERATURE SENSING EQUIPMENT.
Piston and gas turbine aero engines are heat engines. The power they produce is directly
proportional to the heat released during combustion of fuel. Engine components and systems are designed to withstand certain temperatures. If their limits are exceeded they may fail. To allow safe operation the engine temperatures must be monitored. The effect of ambient temperatures as well as combustion must be considered. The following temperatures are monitored on piston and gas turbine engines: Air Inlet, Piston Cylinder Heads, Piston Exhaust Gas, Gas Turbine Compressor outlets, Turbines Oil and Fuel systems and Internal Air system.
The temperatures monitored may range from -560c to +12000c. Different sensors are used depending on the temperature range to be monitored. They fall broadly into two categories, High temperatures and low temperatures. There are four major types of measuring devices. They are:
Expansion Type. This relies on the principle that most solids, liquids and gases expand and contract with temperature changes.
e.g. The mercury Thermometer and Bi-metallic strip
Vapour pressure Type. Liquids when subjected to a rise in temperature change their state from liquid to vapour. Therefore by measuring the pressure of the vapour an indication of temperature can be gained.
Electrical Type. A Change in temperature Of an electrical conductor can cause a change in resistance of the conductor. Thus measuring the change in resistance can indicate the temperature of the conductor. This sensor is called the Resistance Type (Temperature Bulb). In addition, two dissimilar metals when joined together at their ends (a junction) can produce an electrical potential called a thermo EMF (Seebeck Effect). This is dependant on the temperature difference between the junctions. This is known as a 'Thermo-Electric Type' or 'Thermo-couple'.

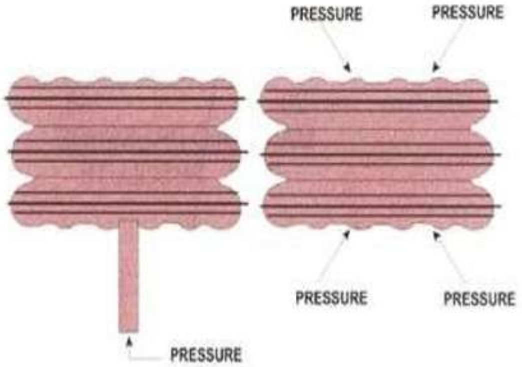
Radiation type. The radiation emitted by any body at any wavelength is dependant upon the temperature of that body. This is termed its "emissivity". If the radiation is measured and the emissivity is know the temperature Of the body can be determined. Such a measuring technique is known as Pyrometry.

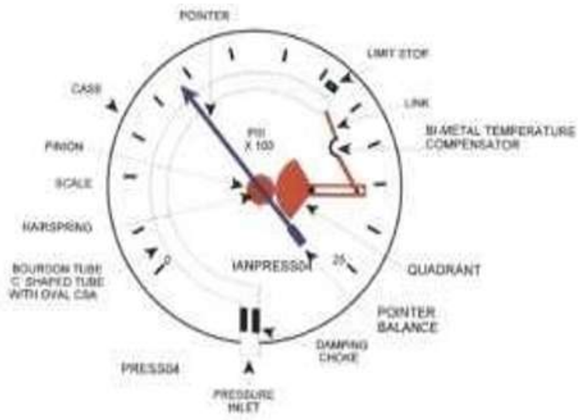
Generally the Expansion and Vapour Pressure sensors are used to indicate lower temperatures. They are direct reading. e.g Thermometer. The Electrical and Radiation sensors are used to measure higher temperatures and can be direct reading to a moving coil Indicator. e.g. Piston Engine Exhaust Gas Temperature. However most systems today use remote Sensors that feed to servo-operated indicators after the signal from the sensor has been amplified. A Ratiometer Type indicating system can be used to obtain the greater accuracy required when indicating the temperature of critical component (Turbines). The ratiometer uses a basic Wheatstone Bridge Balancing circuit that alters the magnetic field in a coil, this change produces a torque to drive an indicator. The temperature Of the gas passing through the turbine in a gas turbine engine is the most important parameter of those displayed on the engine instruments. Operation of the engine beyond the limits of turbine temperature, even for only a moment, is liable to cause excessive turbine blade creep which can be catastrophic if the rotating blades touch the casing of the engine. The gas temperature must be monitored closely and automatic temperature limiting equipment is fitted to most gas turbine engines operating today. to enable this monitoring to be achieved temperature probes are inserted in the gas stream. Temperature probes are formed from the junction of two dissimilar metals, when heated the junction generates a small voltage which is proportional to the actual temperature which produced it. The voltage can be measured on a milli-voltmeter and displayed in the cockpit as the temperature at the rear of the engine. A Galvanometer is a very sensitive instrument used to indicate these low voltages

PRESSURE GAUGES

In many of the systems associated with the operation of the aircraft and its engines, liquids and gases are used the pressures of which must be measured and
indicated. The gauges and indicating systems fall into two categories:
Direct Reading and Remote Indicating: Remote indicating is where a separate sensing element is connected to a pressure source at some remote point. Pressure, is defined as force per unit area. It is normally indicated either as Pounds Per Square Inch (PSI) or Inches of Mercury. In connection with pressure measurement we are concerned with the following terms:







Absolute Pressure and Gauge Pressure. Most pressure gauges measure the difference between absolute pressure and the atmospheric pressure. This is gauge pressure.

To actually measure pressure in a system Elastic Pressure Sensing Elements are used in which forces can be produced by applied pressures and converted to mechanical movement. The movement can then operate a direct reading gauge or electrical transmitter. The sensing elements commonly used are Diaphragms, Capsules, Bellows and Bourdon tubes. Diaphragms consist of corrugated

circular metal discs which are secured at their edge, and when pressure is applied they are deflected. Diaphragms are used to measure low pressures.

Capsules are made up of two diaphragms placed together and joined at their edges to form a chamber which may be sealed, called an Aneroid, or open to a pressure source and called a Pressure capsule. Like diaphragms they are used to measure low pressure, but they are more sensitive to small pressure changes.

Aneroid Capsule

The bellows type element can be considered as an extension of the corrugated diaphragm principle. It may be used for High, Low or

differential pressure measurement. It is typically used to measure pressures like the aircraft's LP Booster pump output. The Manifold Absolute Pressure Gauge or MAP of a piston engine measures both pressure and differential pressure. Note this gauge measures Absolute Pressure and indicates inches of mercury (in Hg.). When the engine is running this gauge can indicate less than atmospheric pressure.

Earlier versions of this gauge were calibrated to called Static Boost.

Bellow Sensor

read Boost in PSI and called Boost Gauges. Under standard conditions the Boost Gauge will read 'Zero' and the MAP gauge will read 30 in Hg. This indication is fluids can be measured at their source and not brought into the cockpit. Also weight can be saved by reducing the length of pipelines. The transmitters feed varying current to an indicator and can be AC or DC in operation. As well as indicating actual pressures, warnings can be displayed to the pilot by pressure operated switches. These switches can operate for Low, High Or incorrect Differential pressures. A differential Switch or gauge is subjected to pressure on both sides of its sensor Coloured arcs of Green, Amber or Red are used to indicate the range and limits of the system.

TEMPRATURE INDICATING SYSTEM

Methods and Applications

The Bourdon tube is about the oldest of the pressure-sensing element. The element is essentially a length Of metal tube with an elliptical cross section, and shaped into a letter C. One end of the tube is sealed, and called the free end. The other end is connected to the pressure source and fixed. When pressure is applied the tube tries to straighten, this movement is magnified to drive and indicator pointer. The Bourdon tube can be manufactured to indicate high or low pressures, but is normally associated with higher pressures such as engine oil pressure.

Manifold Pressure gauge

It would be impractical to run an oil feed pipe from the outer engine of a Boeing 747 to a flight deck pressure indicator of the Bourdon tube type. To overcome this problem remote- indicating systems are used. They consist of two main components, a transmitter unit located at the pressure source, and an indicator mounted on the appropriate panel. They have distinct advantages over direct-reading gauges; for example, the pressure of hazardous

In most forms of temperature measurement, the variation of some property of a substance with temperature is utilized. These variations may be summarized as follows

1, Most substances expand as their temperature rises; thus, a measure of temperature is obtainable by taking equal amounts of expansion to indicate equal increments of temperature.

2. Many liquids, when subjected to a temperature rise, experience such motion of their molecules that there is a change of state from liquid to vapour. Equal increments of temperature may therefore be indicated by measuring equal increments of the pressure of the vapour.

3. Substances change their electrical resistance when subjected to varying temperatures, so that a measure of temperatures is obtainable by taking equal increments of resistance to indicate equal increments of temperature.

4. Dissimilar metals when joined at their ends produce an electromotive force (thermo e.m.f) dependent on the difference in temperature between the junctions. Since equal increments of temperature are only required at one junction, a measure of the electromotive force produced will be a measure of the junction temperature.

5. The radiation emitted by any body at any wavelength is a function of the temperature of the body, and what is termed its emissivity. If, therefore, the radiation is measured and the emissivity is known, the temperature of the body can be determined; such a measuring technique is known as radiation pyrometry. The utilization of these various methods provides us with a very convenient means of classifying temperature- measuring instruments:

(a) expansion type (liquid or solid)
(b) vapour-pressure type,
(c) electrical type (resistance or thermo-electric)
(d) radiation type.

The majority of instruments currently in use are, however, of the resistance and thermoelectric type and are applied to the measurement of the temperature of such liquids and gases as fuel, engine lubricating oil, outside air, carburettor air, and turbine exhaust gas. In certain types of turbojet engine, the radiation pyrometry technique is also applied to the measurement of actual turbine blade temperature.

Heat and Temperature Heat
Heat is a form of energy possessed by a substance and is associated with the motion of the molecule of that substance. The hotter the substance the more vigorous is the vibration and motion

of its molecules. We may regard heat, therefore, as molecular energy. The quantity of heat a substance contains is dependent upon its temperature, mass and nature of the material from which the body is made. A bucketful of warm water will melt more ice than a cupful of boiling water; the former must therefore contain a greater quantity of heat even though it is at a lower temperature.
The transfer of heat from one substance to another may take place by conduction, convection and radiation.
Conduction requires a material medium, which may be solid, liquid or gaseous. Hot and cold substances in contact interchange heat by conduction.
Convection is the transmission of heat from one place to another by circulating currents and can only occur in liquids and gases.
Radiation is the energy emitted by all substances, whether solid, liquid or gaseous.
Temperature
Temperature is a measure of the 'hotness' or 'coldness' of a substance or the quality of heat. Therefore, in the strictest sense of the term, temperature cannot be measured. The temperatures of substances can only be compared with each other and the differences observed, and so the practical measurement of temperature is really the comparison of temperature differences. In order to make such a comparison, the selection of a standard temperature difference, a fundamental interval and an instrument to compare other temperature differences with this, are necessary

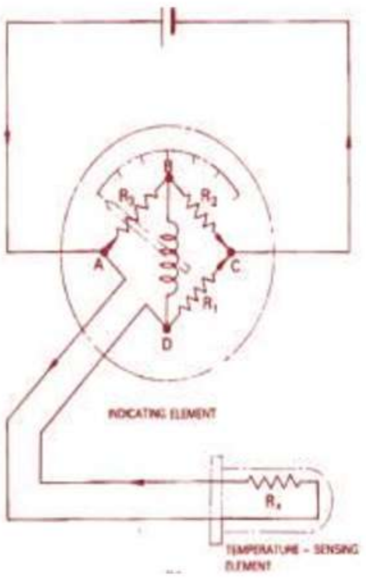
Melting Point and Boiling Point
For pure substances, the change of state from solid to liquid and from liquid to vapour takes place at temperatures which, under the same pressure conditions, can always be reproduced. Thus, there are two equilibrium temperatures known as (i) melting point, the temperature at which solid and liquid can exist together in equilibrium, and (ii) boiling point, the temperature at which liquid and vapour can exist together in equilibrium.

Scales of Temperature
The fundamental interval is divided into a number of equal parts or degrees, the division being in accordance with two scale notations Celsius (centigrade) and Fahrenheit.
Electrical Temperature Indicating Systems
As we have already noted, these systems fall into two main categories: variable-resistance and thermoelectric: the methods are termed resistance thermometry and pyrometery respectively.
Ohm's Law
This law may be stated as follows: When a current flows in a conductor, the difference in potential between the ends of such conductor, divided by the current flowing through it, is a constant provided there is no change in the physical condition of the conductor.

Wheatstone Bridge

The circuit is made up of four resistances arms, R , R2 , R3 and R,. A moving-coil or moving-spot galvanometer is connected across points B and D, and a source of low voltage is
connected across points A and C. Current flows in the directions indicated by the arrows, dividing at point A and flowing through R3 and R,, at strengths which we may designate respectively as I1 and I, . At point C the currents reunite and flow back to the voltage source, Let us assume that the resistance of the four arms of the bridge are so adjusted that B and D are at the same potential; then no current will flow through the

galvanometer and so it will read zero. Under these conditions the bridge is said to be 'balanced'. This can be shown by applying Ohm's law; if the potential difrerence between A and B or A and D is, say, VI , and the potential difference between B and C or D and C is V2, then and If now
eqn (3) is divided by eqn (4), we have from which Hence, an unknown resistance can be calculated by adjusting the values of the three others until no current flows through the galvanometer, as indicated by no movement of its pointer or spot. It will be apparent that, if the resistor R, is subjected to varying temperatures and its corresponding resistances are determined, then it is feasible for the network to serve as a simple electrical-resistance thermometer system.



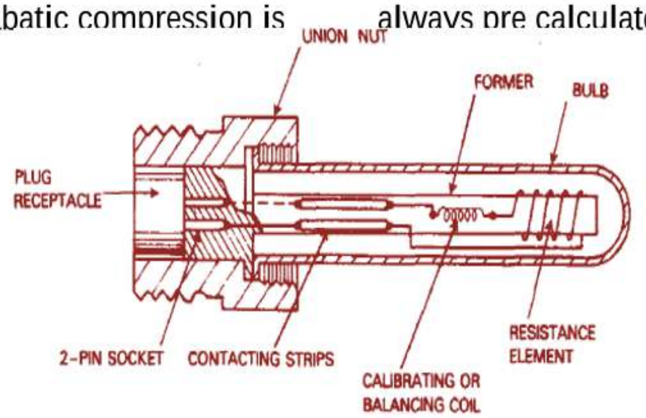
Obviously some rearrangement of the circuit is necessary in order to obtain automatic response to the variations in the temperature] resistance relation. The unknown resistance Rx forms the temperature-sensing element and is contained within a metal protective sheath, the assembly being called a bulb. The three other resistances instead of being adjustable, are fixed and are contained within the case of a moving-coil indicating element calibrated in units of temperature. Both components are suitably interconnected and supplied with direct current. When the bulb is subjected to temperature variations, there will be a corresponding variation in R,. This upsets the balance of the indicator circuit, and the value of R, at any particular temperature will govern the amount of current flowing through the moving coil. Thus, for a given value of R,, the out- of-balance current is a measure of the prevailing temperature It will be noted that there is an important difference between the two applications of this bridge network. In one the measurement of a resistance is dependent upon the circuit current being in balance, while in the other it is measured in terms of out-of- balance current. There is in fact only one point in a bridge type of temperature indicator at which the circuit is balanced and at which no current will flow through the moving coil; this is known as the null point. It is usually indicated on the scale of the instrument by means of a small datum mark and the pointer takes up this position when the power supply is disconnected. For accurate temperature measurement, however, this form of indicating circuit has the disadvantage that the out-of-balance current also depends on the voltage of the power supply. Hence, errors in indicated readings will occur if the voltage differs from that for which the instrument was initially calibrated.

Temperature-Sensing Elements

The resistance coil is wound on an insulated former and the ends of the coil are connected to a two-pin socket via contact strips. The bulb, which serves to protect and seal the element, may either be a brass or stainless-steel tube closed at one end and soldered to a union nut at the other. The union nut is used for securing the complete element in the pipeline or component of the system whose liquid temperature is required. The two-pin socket is made a tight fit inside the male portion of the union nut, the receptacle of which ensures correct location of the socket's mating plug. It will be noted from the diagram that the coil is wound at the bottom end of its former and not along the full length. This ensures that the coil is well immersed in the hottest part of the liquid, thus minimizing errors due to radiation and conduction losses in the bulb. A calibrating or balancing coil is normally provided so that a standard constant temperature/resistance characteristic can be obtained, thus permitting interchangeability of sensing elements. In addition the coil compensates for any slight change in the physical characteristics of the element. The coil, which may be made from Manganin or Eureka, is connected in series with the sensing element and is adjusted by the manufacturer during initial calibration.

Air Temperature Sensors

Air temperature is one of the basic parameters used to establish data vital to the performance monitoring of aircraft and engines,
e.g. true airspeed measurement, temperature control, thrust settings, fuel air ratio settings, etc. of turbine engines, and it is therefore necessary to provide a means of in-flight measurement. The temperature which overall would be the most ideal is that of air under pure static conditions at the various flight levels compatible with the operating range of the particular aircraft concerned. The measurement of static air temperature (SAT) by direct means is, however, not possible for all types of aircraft or, in many instances, for one type of aircraft, for the reason that measurements can be affected by the/adiabatic compression of air gesulting from increases in airspeed3 In general, the boundary layer at the outside surfaces of an aircraft flying at speeds below
0.2 Mach number is very close to the SAT. However, at higher Mach numbers the boundary layer can be slowed down or stopped relative to the aircraft, and thereby produce adiabatic compression which will raise the air temperature to a value appreciably higher than SAT. Friction of high speed flow along the aircraft surfaces will also raise the air temperature. This increase is commonly referred to as 'ram rise', and the temperature indicated under such conditions is known as ram air temperature (RAT) i.e. SAT plus the ram rise.
The ram rise due to full adiabatic compression is always pre calculated mathematica lly as a function of Mach



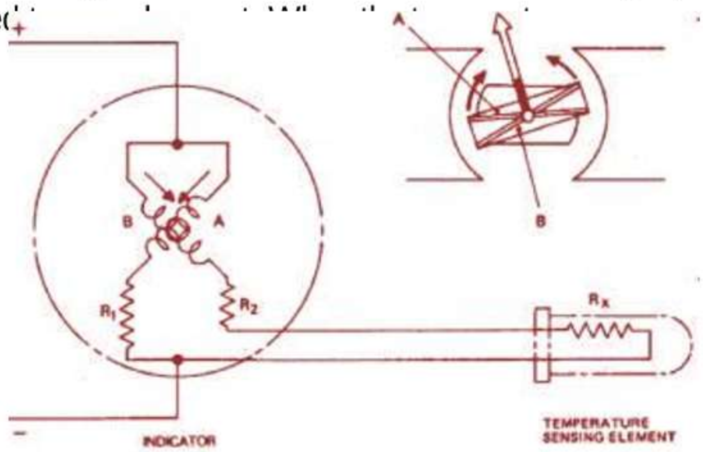
number, and for each type of aircraft values are presented in either tabular or graphical form in the operating manual or the flight manual for the type. Thus, for air temperature sensors subjected to ram rise; the RAT readings of the associated indicators can always be corrected to obtain SAT, either by direct subtraction of tabulated ram rise values, conversion charts, or in the case of air data computers by the automatic application of a correction signal. The proportion of ram rise is dependent on the ability of the sensor to Sense or recover the temperature rise, the sensitivity in this case being expressed as a percentage and termed the recovery factor. If, for example, a sensor has a recovery factor of 0.80, it will measure SAT plus 80% of the ram rise.

Electrical Temperature Indicators

The measuring elements employed in the indicators of resistance- type and a majority of thermoelectric temperature-indicating systems, depend for their operation on the fact that electric current flowing through a conductor produces a magnetic field in and around the conductor. In order to utilize this effect as a method of measurement, it is necessary to have a free-moving conductor in a magnetic field which is both permanent and of uniform strength. In this manner, as will be shown, advantage can be taken of the interaction between the two magnetic fields and the resultant forces to move the conductor and its indicating element to definite positions.

Ratiometer System

A ratiometer-type temperature-indicating system consists of a sensing element and a moving-coil indicator, which unlike the conventional type has two coils moving together in a permanent- magnet field of non-uniform strength. The coil arrangements and methods of obtaining the non-uniform field depends on the manufacturer's design As in any moving-coil indicator, rotation of the measuring element is produced by forces which are proportional to the product of the current and field strength, and the direction of rotation depends on the direction of current relative to the magnetic field. In a ratiometer, therefore, it follows that the force produced by one coil will always tend to rotate the measuring element in the opposite direction to the force produced by the second coil, and furthermore, as the magnetic field is of non-uniform strength, the coil carrying the greater current will always move towards the area of the weaker field, and vice versa When the temperature at the sensing element R, increases, then in accordance with the temperature/resistance relationship of the material used for the element, its resistance will increase and so cause a decrease in the current flowing in winding B and a corresponding decrease in the force created by it. The current ratio is therefore altered and the force in winding A will rotate the measuring element so that both windings are carried round the air gap; winding B is advanced further into the stronger part of the magnetic field while winding A is being advance



at the sensing element stabilizes at its new value the forces produced by both windings will once again balance, at a new current ratio, and the angular deflection of the measuring element will be proportional to the temperature change. When the measuring element is at the mid-position of its rotation, the currents in both windings are equal since this is the only position where the two windings can be in the same field strength simultaneously, In a conventional moving-coil indicator, the controlling system is made up of hairsprings which exert a controlling torque proportional at the sensing element stabilizes at its new value the forces produced by both windings will once again balance, at a new current ratio, and the angular deflection of the measuring element will be proportion al to the temperature change. When the measuring element is at the mid- position of its rotation, the currents in both windings are equal since this is the only position where the two windings can be in the same field strength simultaneously, In a conventional moving-coil indicator, the controlling system is made up of hairsprings which exert a controlling torque proportional to the current flowing through the coil. Therefore, if the current decreases due to a change in the power supply applied to the indicator, the deflecting torque will be less than the controlling torque of the springs and so the coil will move back to a position at which equilibrium between

torques is again established. The pointer will thus indicate a lower reading. A ratiometer system, however, does not require hairsprings for exerting a controlling torque, this being provided solely by the appropriate coil winding and non-uniform field arrangements. Should variations in the power supply occur they will affect both coils equally so that the ratio of currents flowing in the coils remains the same and tendencies for them to move to positions of differing field strength are counterbalanced. In practical applications of the ratiometer system, a spring is, in fact, used and so at first sight this may appear to defeat the whole object of the ratiometer principle. It is, however, essential that the moving-coil former and pointer should take up an off-scale position when the power supply is disconnected, and this is the sole function of the spring. Since it exerts a very much lower torque than a conventional moving-coil indicator control spring, its effects on the ratiometer controlling system and indication accuracy, under power supply changes normally encountered, are very slight,. The power of the output signals from resistance type temperature sensing elements is very limited, so that the moving- coil mechanisms of indicators need to be of delicate construction in order to provide the necessary accuracy and response. For some applications, however, it may be required for indicators to operate in extreme environmental conditions which call for a more robust form of mechanism.

Thermocouple Principle

Thermoelectric temperature-measuring instruments depend for their operation on electrical energy which is produced by the direct conversion of heat energy at the measuring source. Thus, unlike resistance thermometers, they are independent of any external electrical supply. The arrangement of two dissimilar metal wires joined together in this manner is called a thermocouple, the junction at the higher temperature being conventionally termed the hot or measuring junction, and that at the lower temperature the cold or reference junction. (In practice, the hot junction is in the form of a separate unit for sensing the temperature and this is regarded generally as the thermocouple proper.

Thermocouple Materials and Combinations

Group	Metals and composition		Maximum temperature	Application
	Positive wire	*Negative wire*	°C (continuous)	
Base metal	Copper (Cu)	Constantan (Ni, 40%; Cu, 60%)	400	Cylinder-head temperature measurement
	Iron (Fe)	Constantan (Ni, 40%; Cu, 60%)	850	
	Chromel (Ni, 90%; Cr, 10%)	Alumel (Ni, 90%; Al, 2% + Si + Mn)	1,100	Exhaust-gas temperature measurement
Rare metal	Platinum (Pt)	Rhodium-platinum (Rh, 13%; Pt, 87%)	1,400	Not utilized in aircraft temperature-indicating systems

Cr, chromium; Ni, nickel; Al, aluminium; Si, silicon; Mn, manganese

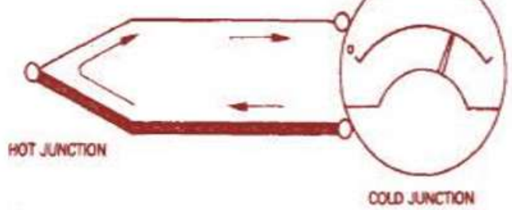
The choice of a particular thermocouple is dictated by the maximum temperature to be encountered in service. Thermocouples required for use in aircraft are confined to those of the base metal group. In order to utilize the thermoelectric principle for temperature measurement, it is obviously necessary to measure the e.m.f.'s generated at the various temperatures. This is done by connecting a moving-coil milli voltmeter, calibrated in degrees Celsius, in series with the circuits so that it forms the cold junction. The introduction of the instrument into the circuit involves the presence of additional junctions which produce their own e.m.f.'s and so introduce errors in measurement. However, the effects are taken into consideration when designing practical thermocouple circuits, and any errors resulting from 'parasitic e.m.f.'s', as they are called, are eliminated.

Types of Thermocouple

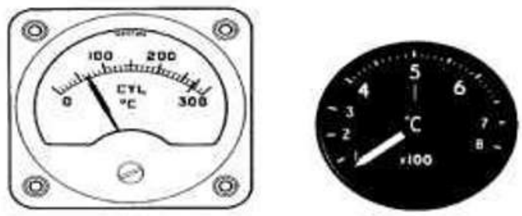
The thermocouples employed in aircraft thermoelectric indicating systems are of two basic types: (i) surface contact and (ii) immersion.

The surface- contact type is designed to measure the temperature of a solid component and is used, as the temperature- sensing element of air-cooled-engine cylinder-head temperature-indicating systems. The copper/constantan or iron/constantan element may be in the form of a 'shoe' bolted in good thermal contact with a cylinder head representative of the highest temperature condition or in the form of a washer bolted between the cylinder head and a sparking plug.

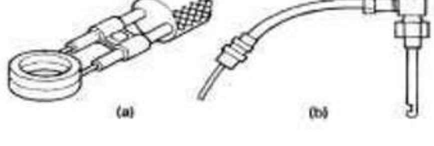
The immersion type of thermocouple is designed for the measurement of gases and is therefore used as the sensing element of turbine engine



gas temperature- indicating systems. The chromel/alumel hot junction and wires are usually encased in ceramic insulation within a metal (typically, Inconel) protection sheath, the complete assembly forming a probe which can be immersed in the gas stream at the points selected for measurement. The techniques for assembly of thermocouple elements include vacuum brazing, induction brazing and argon are welding as well as the technique of electron beam welding. Immersion-type thermocouples are further classified as stagnation and as rapid response, their application depending upon the velocity of the engine exhaust gases. In pure jet engines the gas velocities are high, and so in these engines stagnation thermocouples are employed. The reason for this will be clear from Fig (c), which shows that the gas entry and exit holes, usually called sampling holes, are staggered and of unequal size, thus slowing up the gases and causing them to stagnate at the hot junction, thus giving it time to respond to changes of gas temperature.

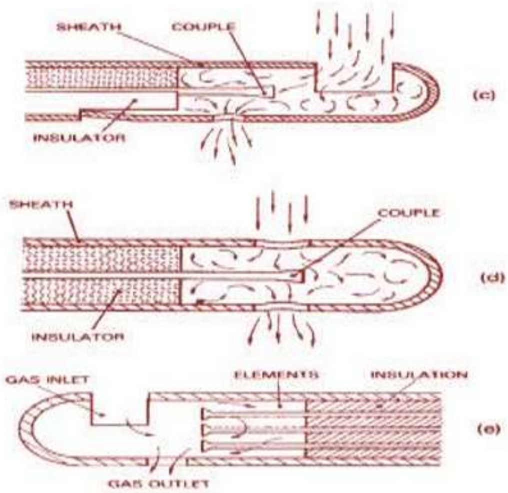


(a) (b)



(a) (b)

Rapid-response thermocouples are employed in turboprop engines since their exhaust-gas velocities are lower than those of pure jet engines. The sampling holes are diametrically opposite each other and of equal size; therefore the gases can flow directly over the hot junction enabling it to respond more rapidly. Typical response times for stagnation and rapid response thermocouples are 1 to 2 seconds and 0.5 to 1 second, respectively.



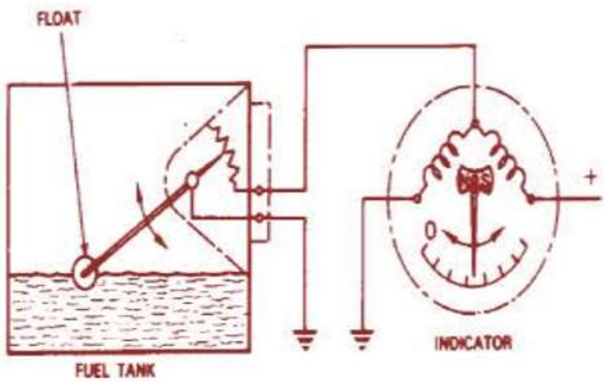
Thermocouple probes are also designed to contain double, triple and in some cases, up to eight thermocouple elements within a single probe. A triple element arrangement is shown schematically in Fig (e). The purpose of such multi-arrangements is to provide additional temperature signals for engine systems which utilize separate facilities, such as exhaust-gas temperature control and engine combustion analysing. Insulation of the thermocouple elements from each other is provided by compacted magnesium oxide (MgO) which also serves to maintain the elements in position. When the hot junctions of immersion-type thermocouples are in contact with the gas stream, it is obvious that not only will the stream velocity be reduckd, but also that the gas will be compressed by the expenditure of kinetic energy, resulting in an increase of hot-junction temperature. It is in this connection that the term recovery factor is used, defining the proportion of kinetic energy of the gas recovered when it makes contact with the hot junction, This factor is, of course, taken into account in the design of thermocouples so that the 'heat transfer', as we may call it, makes the final reading as nearly as possible a true indication of total gas temperature.

FUEL QUANTITY INDICATING SYSTEM

The measurement of the quantity of fuel in the tanks of an aircraft fuel system is an essential requirement, and in conjunction with measurements of the rate at which the fuel flows to the engine or engines permits an aircraft to be flown at maximum efficiency compatible with its specified operating conditions. Furthermore, both measurements enable a pilot or engineer to quickly assess the remaining flight time and also to make comparisons between present engine performance and past or calculated performance. Fuel quantity indicating systems vary in operating principle and construction, the application of any one method being governed by the type of aircraft and its fuel system. Two principal methods currently applied utilize the principle of electrical signal transmission from units located inside the fuel tanks. In one method, mainly employed in the fuel systems of small and light aircraft, the tank units consist of a mechanical float assembly which controls an electrical resistance unit and varies the current flow to the indicating element. The second method, employed in high-performance aircraft fuel systems, measures fuel quantity in terms of electrical capacitance and provides a more accurate system of fuel gauging. Fuel-flow measuring systems also vary in operating principle and construction but principally they consist of two units: a transmitter or meter, and an indicator. The transmitter is connected at the delivery side of the fuel system, and is an electromechanical device which produces an electrical output signal proportional to the flow rate which is indicated in either volumetric or mass units. In some systems an intermediate amplifier/computer is included to calculate a fuel-flow/time ratio and also to transmit signals to an indicator which presents integrated flow rate and fuel consumed information.

Float-Type Fuel Quantity Indicating Systems

The components of a float-type system are shown together with the methods of transmitting electrical signals. The float may be of cork specially treated to prevent fuel absorption, or it may be in the form of a lightweight metal cylinder suitably sealed. The float is attached to an arm pivoted to permit angular movement which is transmitted to an electrical element consisting of either a wiper arm and potentiometer, or a Desynn type of transmitter. As changes in fuel level take place the float arm moves through certain angles and positions the wiper arm or brushes to vary the resistance and flow of direct current to the indicator. As a result of the variations in current flow a moving coil or rotor within the
indicator is deflected to position a pointer over the scale calibrated in gallons.



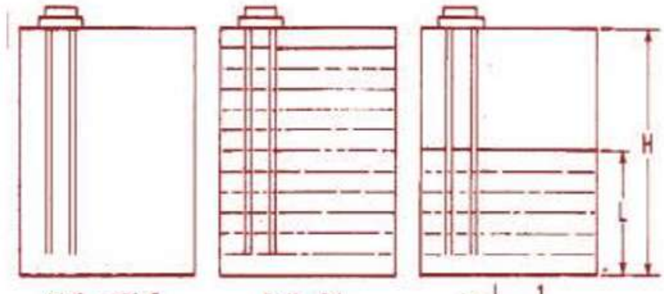
Capacitan ce-Type Fuel- Gauge System

In its basic form, a capacitanc e-type
fuel-gauge system consists of
a variable capacitor located in the fuel tank, an amplifier and an indicator. The complete circuit forms an electhcal bridge which is continuously being rebalanced as a result of differences between the capacitances of the tank capacitor and a reference capacitor. The signal produced is amplified to operate a motor, which positions a pointer to indicate the capacitance change of the tank capacitor and thus the change in fuel quantity. Before going into the operating details of such a system, however, it is first necessary to discuss some of the fundamental principles of capacitance and its effects in electrical circuits.
Air -- 1.00059
Water -- 81-07 - Water vapour -- 1-007
Aviation gasolene -- 1.95 Aviation kerosene -- 2-10

Basic Gauge System

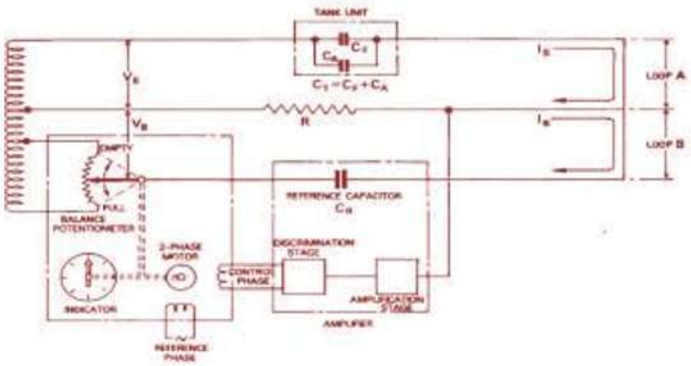
For fuel quantity measurement, the capacitors to be installed in the tanks must differ in construction from those normally employed in electrical equipment. The plates therefore take the form of two tubes mounted concentrically with a narrow air space between them, and extending the full depth of a fuel tank. Constructed in this manner, two of the factors on which capacitance depends are fixed while the third factor, dielectric constant, is variable since the medium between the tubes is made up of fuel and air.

Changes in capacitance due to fuel and air.

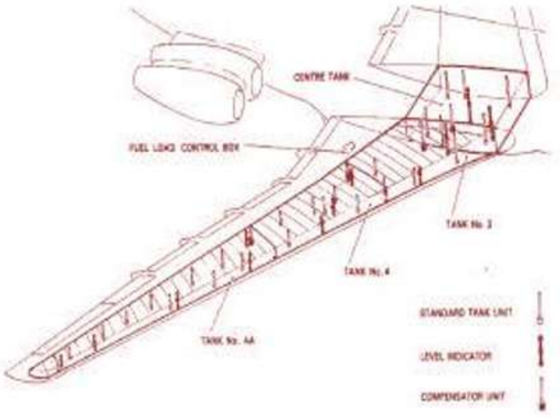


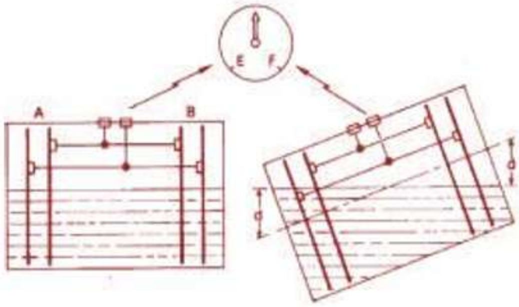
(a) C_A =100pF (b) K = 2·1 (c) $\frac{L}{H} = \frac{1}{2}$

It is divided into two section s or loops by a resistance R, both loops being connected to the secondary winding of a power transformer. Loop A contains the tank capacitor CT and may therefore be considered as the sensing loop of the bridge since it detects current changes due to changes in capacitance. Sensing loop voltage Vs remains constant. Loop B, which may be considered as the balancing loop of the bridge, contains a reference capacitor CR of fixed value, and is connected to the transformer via the wiper of a balance potentiometer so that the voltage VB is variable. The balance potentiometer is contained within the indicator together with a two-phase motor which drives the potentiometer wiper and indicator pointer. The reference phase of the motor is continuously energized by the power transformer and the control phase is connected to the amplifier and is only energized when an unbalanced condition exists in the bridge. The amplifier, which is based on solid-state circuit techniques has two main stages: one for amplifying the signal produced by bridge unbalance, and the other for discriminating the phase of the signal which is then supplied to the motor. Let us consider the operation of the complete circuit when fuel is being drawn off from a full tank. Initially, and at the constant full tank level, the sensing current Is is equal to the balancing current B; the bridge is thus in balance and no signal voltage is produced across R. As the fuel level drops, the tank capacitor has less fuel around it; therefore the added capacitance (CF) has decreased. The tank unit capacitance decreases and so does the sensing current Is, the latter creating an unbalanced bridge condition with balancing current IB predominating through R. A signal voltage proportional to I s is developed across R and is amplified and its phase detected before being applied to the control phase of the indicator motor.



The output signal is a half-wave pulse, a feature of transistors in discriminator circuits, and in order to convert it into a full-wave signal, a capacitor is connected in parallel with the control winding. A capacitor is also connected in series with the reference winding to form what is termed a series resonant circuit. This circuit ensures that the currents in both phases are 90‘ out of phase, the current in the control phase either leading or lagging the reference phase depending on which loop of the bridge circuit is predominating. In the condition we are considering, the balancing current is predominating : therefore, the control-phase current lags behind that of the reference phase causing the motor and balance potentiometer wiper to be driven in such a direction as to decrease the balancing current IB . When the current IB equals the current Is, the bridge is once again in balance, the motor stops rotating and the indicator pointer registers the new, lower value.

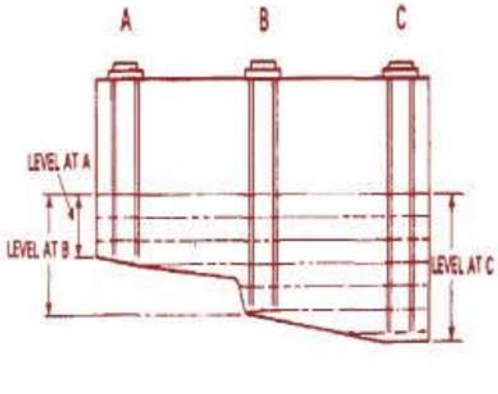




Location and Connection of Tank Unit   Attitude compensation.

Characterized Tank units

The fuel tanks of an aircraft may be separate units designed for installation in wings and in centre sections, or they may form an



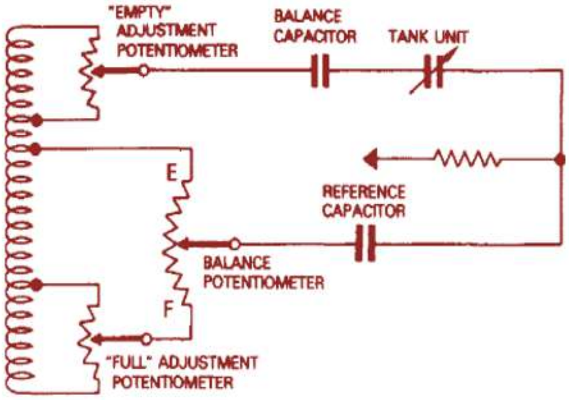
integral sealed section of these parts of the structure. This means, therefore, that tanks must vary in contour to suit their chosen locations, with the result that the fuel level is established from varying datum points.

The contour of a tank located in an aircraft

wing, and as will be noted the levels of fuel from points A, B and C are not the same. When standard tank units are positioned at these points the total capacitance will be the sum of three different values due to fuel (CF), and as the units produce the same change of capacitance for each inch of wetted length, the indicator scales will be non-linear corresponding to the non-linear characteristic of the tank contour. The non-linear variations in fuel level are unavoidable, but the effects on the graduation spacing of the indicator scale can be overcome by designing tank units which measure capacitance changes proportional to tank contour. The non-linear tank units so designed are called characterized tank units, the required effect being achieved either by altering the diameter of the centre electrode or by varying the area of its conducting surface at various points over its length, to suit the tank contour.

Empty and Full Position Adjustments

In order that an indicator pointer shall operate throughout its range between the two principal datums corresponding to empty- and full tank conditions, it is necessary during calibration to balance the current



and voltage of the circuit sensing and balancing loops at these datums. The 'empty' potentiometer is connected at each end to the supply transformer and its wiper is connected to the tank units via a balance capacitor. When a tank is empty, due to the empty capacitance of the tank units, current will still flow through them. The balance potentiometer wiper will also be at its empty position, but since it is earthed at this point, no current will flow through the reference capacitor. However, current does flow through the balance capacitor and it is the function of the empty potentiometer to balance this out. The balancing signal from the potentiometer is fed to the amplifier, the output signal of which drives the indicator and servomotor to the empty position.

The full adjustment may be regarded as a means of changing the position of the point on the balance potentiometer at which the balance voltage for any given amount of fuel is found, and also of determining the voltage drop across the balance potentiometer. if the full potentiometer wiper is set at the bottom, the full transformer secondary voltage will be applied to the balance potentiometer. With the wiper at the top, resistance is introduced into the circuit so that a smaller voltage is applied to the balance potentiometer. Therefore, the distance the balance potentiometer wiper needs to move to develop a given balance voltage can be varied.

GYROSCOPE INTRODUCTION

A knowledge of higher mathematics is required if the subject of gyro- dynamics is to be fully comprehended. However, this is unnecessary for an adequate understanding of the basic principles of the gyroscopic flight instruments. This chapter aims to provide sufficient background knowledge for the study of the Artificial Horizon, Directional Gyro Indicator, and the Rate of Turn Indicator.

THE GYROSCOPE

Any rotating body exhibits gyroscopic phenomena. The earth is a gyro, spinning about the axis between the geographic poles. The road wheel ofa car is a gyro when it is turning, and so is a child's 'top'. The rotor in an aircraft gyro may be little more than an inch in diameter, spinning at perhaps 25,000 r.p.m.

GIMBAL RINGS

These are the supports for the rotor Of a gyroscopic instrument. Gimbal rings, are known briefly as gimbals, and sometimes spelled gimbals. Figure shows a spinning rotor mounted in two gimbal rings, the outer one being supported by a fixed frame. The rotor itself is a metal disc rotating about the axis indicated as X-X and usually called the rotor spin axis. The rotor shaft (or spindle) is supported by bearings in a ring called the inner gimbal. The inner gimbal is in turn supported by bearings mounted inside the Outer gimbal which can rotate on bearings in the frame (or instrument case). The arrangement in Figure allows movement of the gyro in

It follows from the above, that the gyro in Figure has 'three degrees of freedom in rotation that is to say the frame can be rotated in three mutually perpendicular planes without

b) If the frame is rotated about the YY axis, the inner and outer gimbals will obviously not remain at 900 to each other, but the XX axis can remain pointing in the same direction.
c) If the frame is rotated about the ZZ axis (by rotating the base) the alignment of the XX axis Will still be unchanged but the frame and the outer gimbal Will no longer be at right angles to each other.

three mutually perpendicular planes; this gyro is said to have
complete freedom of rotation in three planes at right angles to each other (or freedom of rotation about three axes at right angles to each other).

The Rotor and Gimbal
a)If the frame is rotated about the XX axis, it is being moved either with or against the direction of the rotor spin, so the position of the XX axis is undisturbed.

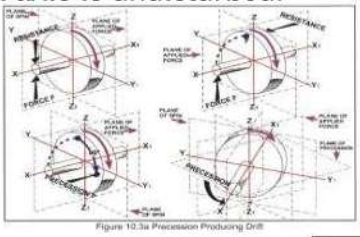
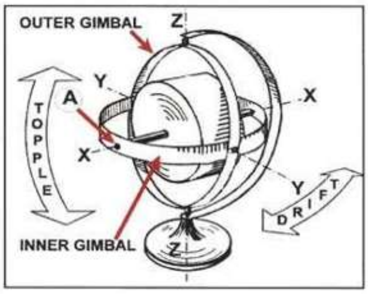


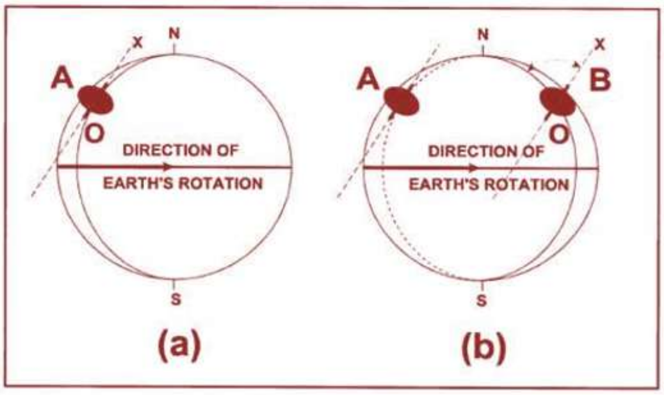
Figure 10.3a Precession Producing Drift



disturbing the spin axis. Such a gyro is known as a space gyro and its chief virtue is that it allows the spinning rotor to exhibit its fundamental property of rigidity.

THE FUNDAMENTAL PROPERTIES OF A GYROSCOPE.

Rigidity. A spinning rotor maintains its axis pointing in a fixed direction in space, unless
subjected to an external force. This property is called rigidity in space or gyroscopic inertia
Precession.
If an external force (or more correctly, torque) is applied to change the direction of the rotor axis, the gyro resists angular movement in the plane of the torque applied and instead moves in a plane at right angles to that of the torque, the resulting movement being called "precession". This is the second fundamental property of a gyroscope.



Rule Of Precession. The direction of presessional movement can be found by the following rule:-
The gyro will precess in a direction at 900 to the applied force, measured round the circumference of the rotor in the direction of spin. The force applied appears to have moved 900 in the direction of spin.
PRECESSION RATE
The formula for the rate of precession (P) is:- P = T/Iw Where; T is the applied torque;
I is the moment of inertia of the rotor; w is the angular velocity of the rotor.
Thus the rate of precession (P) is proportional to T, or the greater the applied force (torque) the greater the rate of precession The rate of precession is inversely proportional to 1m, the rigidity or gyroscopic inertia, so the greater the rigidity the smaller the rate of precession produced by a given applied force.
WANDER
If the axis of a gyro rotor departs from its chosen direction it is said to wander. If the axis of the gyro rotor wanders in the

horizontal plane for instance, as shown by the horizontal arrow in Figure it is said to drift. A weight hung on the gimbal at A in Figure thus produces drift. If the rotor axis wanders in the vertical plane (as shown by the vertical arrow in Figure, it is said to topple. Real Wander. Manufacturing imperfections in a gyroscope cause small rates of 'random'
precession. Other terms given to this precession are 'balance wander' or, if the precession is in the horizontal plane, 'mechanical drift'. The imperfections concerned are uneven rotor bearing friction, unbalanced gimbals, and friction in gimbal bearings. In- flight turbulence may increase the effect of these imperfections. The precession caused by application of an external force to a gyro is another example of real wander.
Apparent Wander. Consider a space (or 'free') gyro at A in above figure with the rotor axis horizontal and aligned with the geographic meridian. The rotor axis is indicating the direction Of true North on the earth and is also aligned with a point at an infinite distance in space.
Apparent Wander

Some time later when point A on the earth has rotated to B in figure, the gyro rotor axis is still aligned with the same fixed point in space (assuming no other disturbing forces) but no longer indicates the direction of North on the earth. It has therefore changed its alignment, according to the earthbound observer, by the angle NOX. This is apparent wander or apparent drift, due Only to the rotation of the earth.
TIED GYROS
A gyro which relates its rigidity to a particular attitude or direction rather than to a space
direction has more applications. Such a gyro is called a tied gyro. A tied gyro can be described as one having freedom in three planes mutually at right angles but with the rotor kept (Or 'tied') in some desired position by a control system. As an example, the DGI rotor axis is maintained in the yawing plane of the aircraft. An earth gyro (as in the Artificial Horizon) is a tied gyro which utilizes the force Of gravity to monitor a control system keeping the rotor axis in the earth's vertical.

RATE GYROS

If a gyro has Only One gimbal ring, with consequently only two planes Of freedom, it can be adapted for use as a rate gyro to measure a rate of angular movement. If the gyro frame is rotated in the plane in which the gyro has no freedom, the rotor will precess, unless restrained, until its plane of rotation coincides with the plane in which the frame is being turned. If this precession is restricted by a spring, as shown in Figure, the resultant tilt of the rotor will be a measure of the rate Of angular movement Of the instrument

Gyroscopic Inertia or Rigidity in Space is the property of a spinning body to maintain a fixed direction in space unless acted on by an outside force.

Precession is the movement of a gyro, resulting from the application of an outside force, about an axis perpendicular to both the spin axis and the applied force.

Real Wander is a precession caused by mechanical imperfections (such as uneven bearing friction) or by any applied force.

Apparent Wander is the observed wander of a gyro rotor axis relative to a datum on the earth.

Topple is rotor axis wander - real or apparent - in the vertical plane.

Drift is rotor axis wander - real or apparent - in the horizontal plane.

Space Gyro has complete freedom in three mutually perpendicular planes, and so has gyroscopic inertia relative to a point in space.

A Tied Gyro has the rotor axis maintained (tied) in a desired position, so having gyroscopic inertia relative to the reference to which it is tied.

An Earth Gyro is a gravity-controlled tied gyro, so having gyroscopic inertia relative to the earth's vertical.

THE APPLICATION OF THE PROPERTIES OF A GYRO.

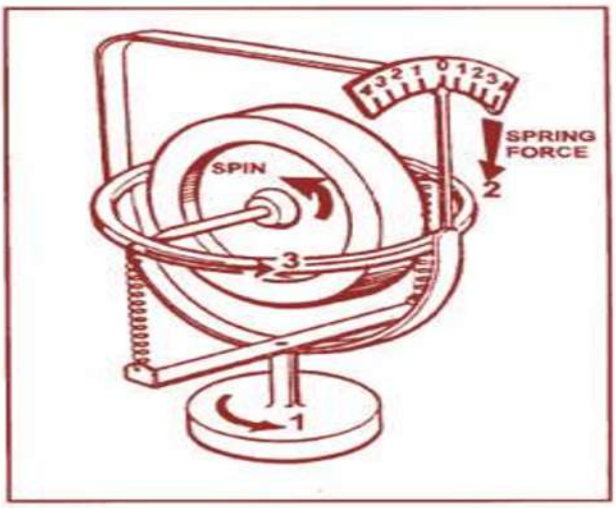
Rigidity. This property is used to provide a directional datum. In the Directional Gyro Indicator (DGI), the rotor spin axis is horizontal and so provides a reference for the measurement of direction in azimuth (aircraft heading). The gyro in the DGI is sometimes known as an azimuth gyro. In the Artificial Horizon (AH), gyroscopic rigidity provides a pitch and roll attitude reference, the spin axis defining the earth's vertical. Such a gyro may be called a vertical axis gyro or vertical reference gyro.

Precession. As already explained, the application of an external force to a gyro produces movement of the rotor axis called precession. This property is used:-

a) In the Rate of Turn Indicator - to measure angular velocities in the yawing plane.

b) In the DGI control system - to maintain the rotor axis in the yawing plane. Additionally, to compensate DGI wander by means of a 'latitude nut'.

c) In the Artificial Horizon control system - to maintain the rotor axis vertical.



SUCTION AND ELECTRIC GYROS

Gyroscopic flight instruments may be either air-driven or electric. With the former, an engine-driven vacuum pump, or venturi tube On some light aircraft, reduces the pressure in the instrument case so that filtered replacement air is sucked in and led through a jet impinging on 'buckets' cut in the periphery Of the rotor to make it spin like a water wheel. The air also operates the control systems of tied gyros. With electric gyros, the rotor is an integral part -
comprising the rotor windings - of an AC motor. The control System Of an electric tied gyro is operated by limit switches
and torque motors.

There are three designs of Gyro currently in use in aviation: Tuned Rotor
Gyro - a mechanical device using a spinning rotor which is engineered to be as balanced and frictionless as possible and which may be found in a number of different instruments.

Laser Gyro - more accurately described as a Rate Sensor this device uses Laser Light set in a rotational pattern to discern movement - it has no moving parts

Fiber-optic Gyro - an extension of the Ring Laser Gyro but lighter, smaller and less expensive but also less accurate. They may be used in a number of applications, that do not require maximum long term accuracy and they may also be employed as a back up to another main system.

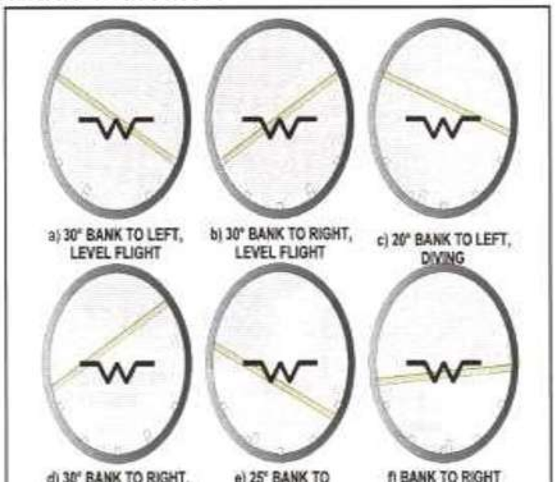
SUCTION AND ELECTRIC TYPES - COMPARISON

The advantages and disadvantages of Electric and Suction gyros: a)Suction Gyros. These are independent of electric power and so are not vulnerable to the risk of total electric failure. Unfortunately, moisture, dust, oil, and nicotine in the airflow penetrate the filter, reducing bearing life, unbalancing gimbals, and so impairing accuracy. At high altitude, adequate suction cannot be
maintained. With a venturi tube, suction is insufficient on the ground and in flight such a tube is subject to icing risk.

b) Electric Gyros. These are a later development than the suction types and in general are more expensive, heavier, and require AC power supplies. They have, however, important advantages over the air-driven types without the latter's disadvantages. The electric rotor can be constructed with higher moment of inertia and spin speed, giving greater rigidity and consequently improved stability. Rotor speed can, at any altitude, be accurately maintained, giving improved accuracy. Operating r.p.m. is more quickly reached when first switching on. The instrument case is sealed, for longer life and greater accuracy.

ARTIFICIAL HORIZON INTRODUCTION.

The artificial horizon (AH) provides the pilot with information in terms of the aircraft's attitude both in pitch and roll. It is a primary instrument, replacing the natural horizon in poor visibility. The attitude display consists of a miniature aircraft shape or 'gull-wing' (tail view) painted or engraved centrally on the inside Of the glass face Of the instrument, and therefore fixed to the instrument case and the actual aircraft. Behind this representation of the aircraft is the horizon bar, linked to the gyro in such a way that the bar is gyro-stabilised parallel to the true horizon. The artificial horizon may be suction or electrically driven. It is also known as a gyro horizon and attitude indicator.



CONSTRUCTION

The artificial horizon uses an Earth gyro in which the spin axis is maintained in, or tied to, the vertical by Earth's gravity. This means that the plane of the rotor rotation is horizontal, so providing the stable lateral and longitudinal references required.

ARTIFICIAL HORIZON INDICATIONS

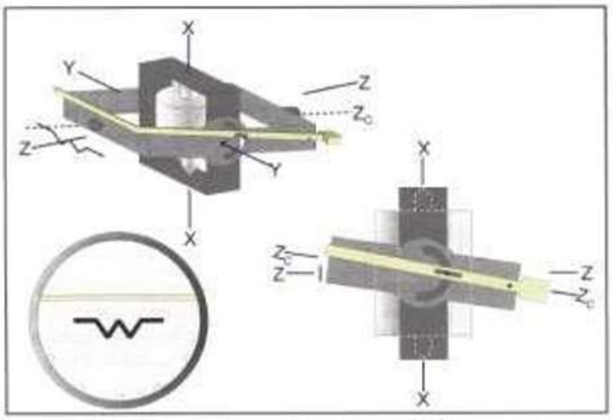


Figure 12.3. A Nose Down Attitude.

Figure shows the three axes of the gyro; XX, YY and ZZ. Because the gyro is tied to the vertical note that the axis XX (the spin axis) will remain Earth vertical and therefore the axis YY will be Earth horizontal when the aircraft is straight and level.

Pitch. the level-flight attitude display and two views Of the instrument with the case removed. In Figure a nose-up attitude (of 10 degrees) is shown. The pitch-up movement has rotated the case together with the attached outer gimbal ring about the lateral axis As this occurs, a guide pin protruding from the stabilized inner gimbal forces the horizon bar arm down. The horizon bar is now below the gull-wing producing the nose-up indication. 12.3. relate to a pitch-down situation. Note that the angle of pitch may be selected using the pitch markers shown.

Roll. In roll, as with pitch, the rigidity of the vertical gyro provides the stable attitude reference. As the aircraft rolls (about the
longitudinal axis - ZZ in the diagrams) the instrument case and the gull-wing will rotate about the stabilized gyro rotor and
gimb al syste m.
The gyro scop ic
rigidity of the spinning rotor holds the horizon bar in the rolling plane so that the amount and
direction of bank are displayed by the gull-wing relative to the horizon bar. A more accurate indication of the amount of bank is
given by a pointer attached to the outer gimbal
and showing bank angle on a scale painted on the face of the instrument. LIMITATIO
NS

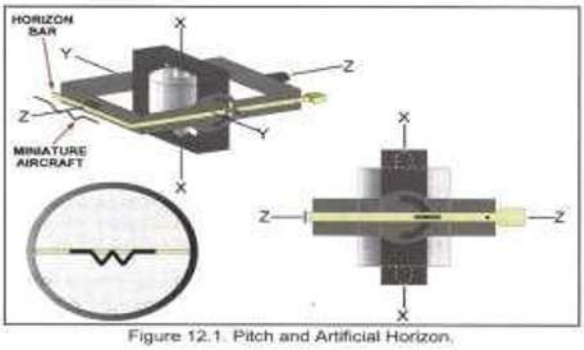


Figure 12.1. Pitch and Artificial Horizon.

The amount the case can move relative to the gyro is controlled by fixed stops. With older designs, typical limits are ± 600 in pitch and 1100 each way in roll. In modern instruments there is complete freedom in roll and up to 850 (plus or minus) in pitch. If the limits are exceeded, the gyro 'topples', giving violent and erratic movements of the horizon bar. Unless a fast erection system is incorporated, accurate indications will not be obtained until the gyro has re-erected itself over a period Of 10 to 15 minutes.

CONTROL SYSTEMS

The rotor assembly is made very slightly bottom-heavy in order to keep down the time taken for initial erection when the gyro is first started up, but a complex control system is required to maintain the rotor axis vertical in flight. A suction or air driven artificial horizon exhausts air through four slots which are normally half covered by four pendulous vanes. Electric artificial horizons use leveling / mercury switches and torque motors.

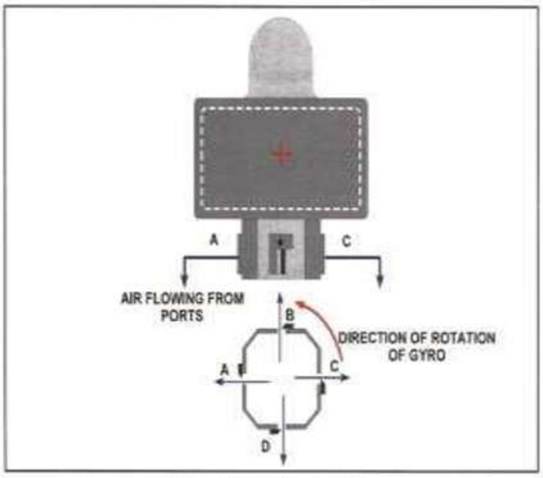


Figure 12.6. Equilibrium.

THE AIR DRIVEN ARTIFICIAL HORIZON

In the air driven artificial horizon an engine-driven suction pump (or venturi tube in some light aircraft) is used to create a suction of about 4 inches of mercury in the instrument case. Replacement air, drawn in by this suction Via a filter, is ducted through the outer and inner gimbals to enter the rotor case as a jet which spins the rotor at up to 15,000 rpm. After driving the rotor, the air passes into the instrument Case through slots at the base Of the rotor housing.

Control System.

The control system of the air driven artificial horizon consists of four slots and four pendulous (hanging) vanes at the base of the rotor housing. The vanes hang down so that when the rotor axis is vertical each slot is half covered by its vane, and four equal jets of air emerge from the slots, fore an aft and left and right, as in Figure. Because the four jets are of equal strength but in opposite directions no force is exerted On the gyro and therefore no precession occurs - the gyro rotor remaining vertical. However the opposing vanes are fixed to a common spindle so that the four vanes operate as two pairs. The positioning of the vanes is such that if the rotor axis wanders from the vertical, one vane will hang clear of its slot, allowing unrestricted airflow, while the opposite slot is completely obstructed by its vane. The resulting unbalanced airflow precesses the gyro and corrects the tilt, returning the gyro axis to the vertical. Exactly how this correction is achieved is shown in Figure. The gyro has wandered from the vertical so that vanes A and C are not affected and remain half covering their slots. However, vanes B and D, on a common spindle, hang down so that slot B is now closed and D is wide Open. A strong jet exits through D causing an equal and opposite reaction _R' on the gyro. This reaction is precessed through 900 in the direction of rotor spin (anticlockwise hen viewed from the top) and acts in the direction of 'P' which restores the gyro axis to the vertical.

ACCELERATION ERROR IN THE AIR DRIVEN ARTIFICIAL HORIZON.

The control system of the air driven artificial horizon depends on the pendulous vanes being affected by the Earth's gravity.

However, the vanes will be affected by any acceleration, not just that due to gravity. When an aircraft

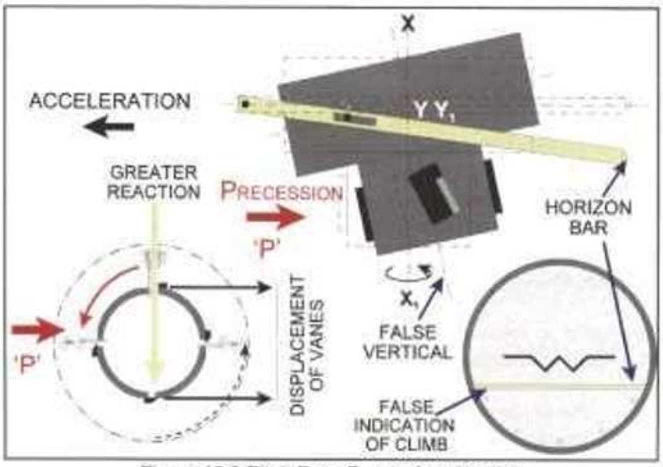


Figure 12.8 Pitch Error Due to Acceleration

accelerates in a level attitude (such as during the take-off run) a false nose up, right wing down, or climbing right hand turn indication will result. The pitch error is due to the effect of acceleration on the lateral pendulous vanes. The roll error is due to the inertia of the bottom-heavy rotor housing. These effects are now considered in more depth.

a) Pitch Error. During acceleration, the lateral vanes lag, swinging back towards the pilot, opening the starboard slot and closing the port slot. This results in a reaction 'R' which acts to port By the rule of precession the effect on the gyro is

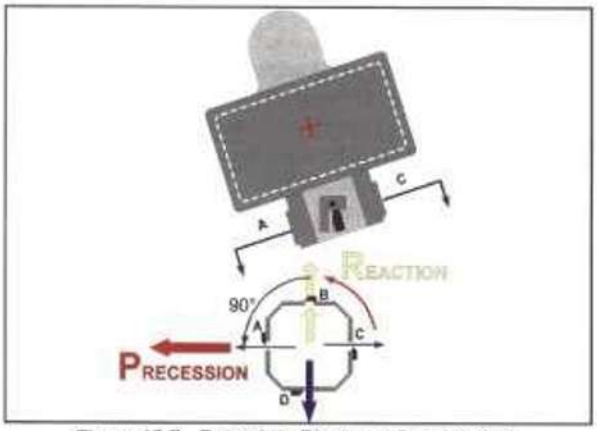


Figure 12.7. Rotor Axis Displaced from Vertical.

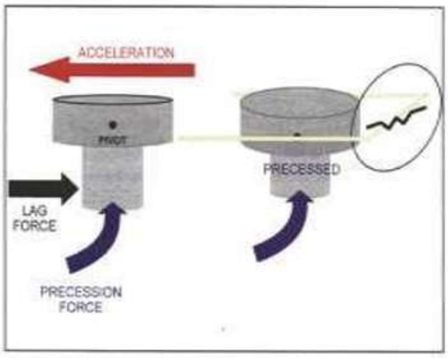


Figure 12.9. Roll Error Due to Acceleration.

as if the direction of application of R had been moved 900 in the direction of rotor spin (anticlockwise). The gyro will now be precessed out of vertical with the base moving backwards towards the pilot. This movement is transmitted via the guide pin and horizon bar arm to bring the horizon bar below the gull-wing giving a nose-up indication.

b) Roll Error. Due to inertia, the weighted base Of the rotor housing tries to lag during acceleration. However, this force will be precessed, resulting in the base of the rotor housing moving to starboard and the gyro axis precessing out of the vertical. This rotates the whole rotor/ gimbal assembly about the longitudinal axis to give a right wing down indication.

Deceleration will cause a nose down, left wing error, the opposite of the acceleration error indication. These errors assume that the rotor is rotating anticlockwise When Viewed from the top. which is the for British air driven artificial horizons. Most electric horizons and Some American air driven horizons have clockwise rotor spin. giving opposite errors

THE ELECTRIC ARTIFICIAL HORIZON

The main advantage of electric artificial horizon over the air driven horizon is its greater rigidity due to its faster spin rate. This greater rigidity results in increased accuracy due to reduced errors. The basic principle of the instrument is the same as the air driven horizon. The vertical gyro is still tied by Earth's gravity, but by
mercury / leveling switches and torque motors rather than the pendulous vanes of the air driven horizon.

ELECTRIC ARTIFICIAL HORIZON CONTROL SYSTEM.

The gravity-operated control system consists Of mercury / leveling switches (which are fixed to the base of the rotor) and electric torque motors. If a leveling switch is not level the mercury liquid ball

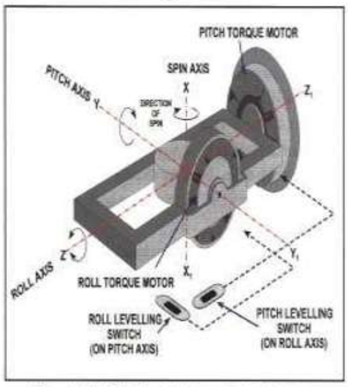


Figure 12.10. The Electric Horizon Control System.

moves from its central position and closes the circuit to drive its torque motor. The torque motor provides the force which is precessed to return the gyro axis to the vertical. There are two leveling switches, one to sense pitch and one to sense roll. Because of the 90* precession rule, the torque motor on the side of the inner gimbal corrects wander in the rolling plane (applying torque round the lateral axis to produce rotation about the longitudinal axis). Likewise the pitch torque motor is on the outer (longitudinal) gimbal so that the precession is about the lateral axis to correct for pitch. This control system, like that of the suction horizon, is designed to compensate for turning errors by maintaining the rotor axis slightly tilted away from the true vertical and having the horizon bar compensated by a similar amount. The amount and direction of this tilt depends on the particular model of instrument in use.

ACCELERATION ERRORS IN THE ELECTRIC HORIZON

Acceleration errors are minimal in the electric artificial horizon for the following reasons;

a) The high rotor speed of an electric artificial horizon, results in very high gyro rigidity and therefore very low precession rates. There is therefore less potential for the gyro to move out of the Earth's vertical.

b) The rotor housing is less bottom heavy in the electric artificial horizon and therefore roll error is reduced while accelerating. c)Pitch and roll cut-out switches. When an aircraft in a level attitude accelerates the pitch levelling switch will falsely complete the circuit as the mercury 'ball moves back in its tube(due to inertia). As this would then result in the pitch torque motor falsely precessing the gyro out of the vertical, a pitch cut-out switch is included in the circuit which activates when an acceleration of 0.18G or greater is detected.

d)Similarly in a turn the roll mercury switch would falsely activate the roll torque motor. A cut out is therefore incorporated in the circuit which is activated at 10 degrees angle of bank.

FAST ERECTION SYSTEM.

In many electric horizons a fast erect system is included to give rapid initial erection and quick re-erection should the instrument have toppled due to exceeding the operating limits. Quoting typical figures, the normal erection rate of 40 per minute is increased to 1200 per minute by pushing the fast erection knob on the face of the instrument. This action increases the voltage to the erection torque motors. One advantages of having a fast erection system is that the pendulosity (bottom-heaviness) Of the gyro can be reduced, so decreasing the turning and acceleration errors.

Note: When airborne, the fast erection knob can only be used successfully in level flight with no acceleration. During acceleration or a turn, the liquid level switches would be 'off- centre', and operation of the fast-erection system would align the rotor axis with a false vertical.

TURN AND SLIP INDICATOR THE RATE OF TURN INDICATOR.

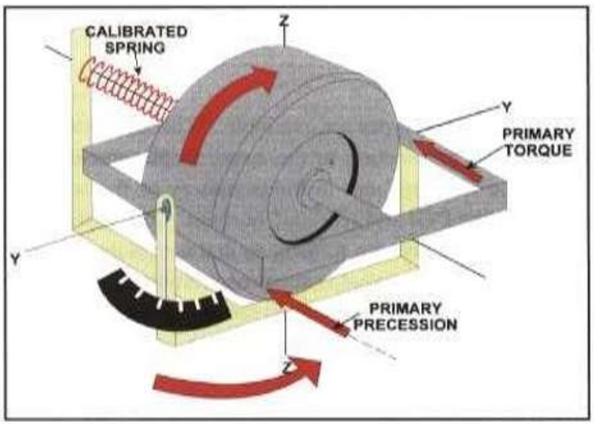
This instrument incorporates two measuring devices, both indicating on the same instrument face. One of these, the rate of turn indicator, (commonly shortened to _turn‘ indicator), uses a rate gyro to measure rate of turn about a vertical axis. The other, the slip indicator, is a very simple pendulous device which is used mainly to show whether or not a turn is balanced, (whether the angle of bank is correct for the TAS and rate of turn), and if not, to indicate the extent of slip or skid.

THE RATE GYRO.

The turn indicator employs a rate gyro which, having only one gimbal, has freedom about only two axes. There is, of course, complete freedom of rotation about one of these - the rotor spin axis - which in level flight lies athwart ships (laterally). There is restricted freedom about the fore/aft (longitudinal) gimbal axis. There is no freedom about the aircraft's vertical axis, so any torque applied about this axis - as in a turn - will cause the gyro to precess.

OPERATION.

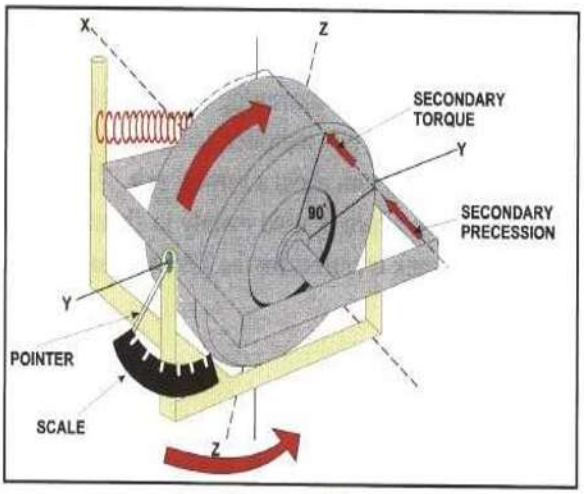
If the aircraft turns, the rotor is subjected to a primary torque acting about the ZZ axis. This produces a primary precession about the YY axis, the direction Of this precession being as if the applied force were moved 90* in the direction of rotor spin. As the rotor tilts about the VY axis it causes a spring between gimbal and frame to be extended. The resultant spring tension subjects the rotor to a secondary torque acting about the YY axis. This secondary torque, with the precession will continue until the gimbal has tilted just the right amount to give the spring tension required to generate a rate of secondary precession equal to the rate of turn of the aircraft. This gives equilibrium. It should be emphasised that the chain of events is virtually instantaneous - as the aircraft goes into a turn, the gimbal takes up the appropriate angle of tilt.



If the rate of turn changes, the tilt of the gimbal will also change, to re-establish the balance of torques on the gyro. The angle of tilt is thus a measure of the rate of turn. A pointer fixed to or linked with the gimbal indicates the tilt on a scale on the face of the instrument. The scale is calibrated to indicate rates of turn either side of the Centre zero so that the first graduation corresponds to a Rate 1 turn with the aircraft turning 30 per second. A second mark for Rate 2 corresponds to 60 per second. There may be further graduations for higher rates Of turn. It is desirable that the angle of tilt Of the gimbal should equal the angle of bank in a turn, so that the rotor axis remains horizontal and the instrument measures rate of turn about the earth's vertical. By spinning the rotor up and away from the pilot and by having a suitable spring sensitivity, the direction and amount of processional tilt of the gimbal will keep the rotor axis approximately horizontal in a moderately banked turn. For a given sensitivity, the tilt of the rotor depends on the rate of turn, but the bank angle, even in a balanced turn, depends on rate of turn and TAS. It follows that only at particular TAS, specified in the calibration, will the rotor axis stay horizontal and the rat
indications be substantially accurate. In practice the errors produced by TAS deviations are not serious. One manufacturer quotes a maximum error of 5% over a speed range of 85 to 350 knots, the calibration value being 260 knots.

THE SLIP INDICATOR

It is desirable that turns should be properly balanced, with no side slip or skid. This implies that the angle of bank should be correct for the TAS and rate of turn. The correct bank angle could be accurately calculated, or rules of thumb may be used. For instance, with Rate I turns, we can take one tenth of the TAS and add seven to give approximately the required bank angle. For example; Rate 1, TAS 150 knots; bank angle should be 15+7 = 22*. This rule gives reasonable accuracy for Rate 1 turns with TAS between 100 and 250 knots. During instrument flight however, the fewer the calculations that have to be made, the better. The slip indicator gives a direct indication of the state of balance of the turn.



CONSTRUCTION

Early types Of Slip indicator employed a simple metal pendulum suspended in the instrument case, its oscillations being controlled by a piston-in-cylinder damping device. The modern version is usually a 'ball-in-tube inclinometer'. This comprises a solid ball in

a curved tube containing liquid with damps out the unwanted oscillations. The heavy ball behaves like a pendulum, with the centre of curvature of the tube acting as the effective point of suspension.

OPERATING PRINCIPLES.

Consider first the aircraft in level flight with lift L balancing weight W viewed in Figure 13.3a. The weight

W of the ball in the tube acts downwards and is exactly balanced by the equal and opposite reaction of the base of the tube on the ball, acting upwards towards the centre of Curvature of the tube. If the wings are level, the ball will lie just between the two vertical lines etched on the tube, as indicated in Figure 13.3b,


Figure 13.3a.
Aircraft in Level Flight

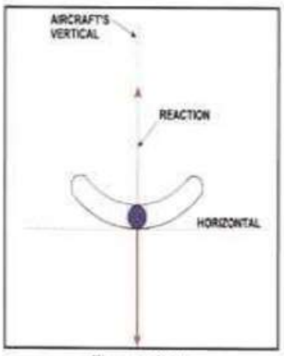


Figure 13.3b.
Ball-in-tube (Level Flight)

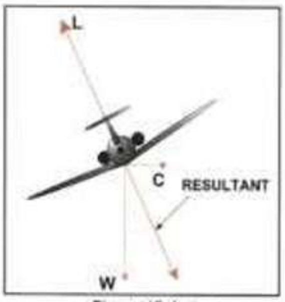


Figure 13.4a.
Balanced Turn To Port

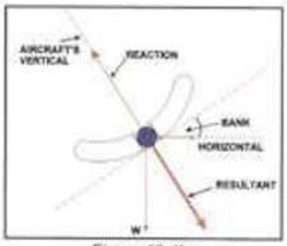


Figure 13.4b.
Ball-in-Tube (Balanced Turn Port)

Now let us consider a balanced turn to the left. Figure 13.4a. shows the aircraft with lift L equal and opposite to the resultant of aircraft weight W and centrifugal force C, the latter being proportional to TAS and rate of turn. The ball is also subject to a centrifugal force depending on TAS and rate Of turn, so it rolls Outwards, taking up a new equilibrium position such that the reaction of the base of the tube on the ball is again exactly balanced, this time by the resultant of ball weight W and centrifugal force C (Figure 13.4b.) Because both aircraft and the ball are experiencing the same TAS and rate Of turn (and so the same acceleration towards the centre Of the turn) it can be proved that the resultant weight and centrifugal force for the aircraft will be parallel to the resultant of weight and centrifugal force for the ball. Now if the ball is laterally central in the tube, (between the two etched lines), the resultant and reaction forces of the ball must lie in the aircraft's vertical (see Figure 13.4b.). These forces are parallel, as stated above, to the resultant of aircraft weight and centrifugal force with must therefore also lie in the aircraft's vertical and will thus be in the same line as the lift L (Figure 13.4a.) which means that the turn is balanced. Unbalanced turns are most simply considered as follows. Let us assume that the TAS and rate of turn are the same as in Figures 1 3.4a. and 13.4b., so that the ball will not have moved laterally. Now if too much bank is applied, (for the TAS and rate of turn), the tube will have been rotated too far in the rolling plane so that the ball appears as in Figure 13.5., no longer central, but correctly indicating the aircraft

to be 'slipping in' to the turn, the radius of which will be less than it should be. If on the other hand insufficient bank has been applied, the instrument will be indicating that the aircraft is 'skidding Out' of the turn (see Figure 13.6.), the radius of turn this time being greater than it should be.

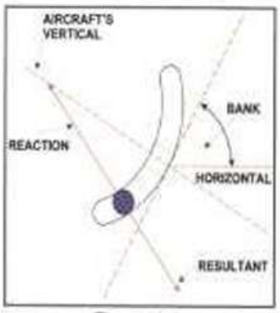


Figure 13.5.
Unbalanced Turn Port (Slipping)

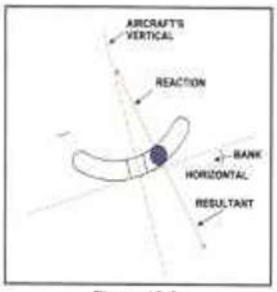


Figure 13.6.
Unbalanced Turn Port (Skidding)

TURN AND SLIP DISPLAYS.

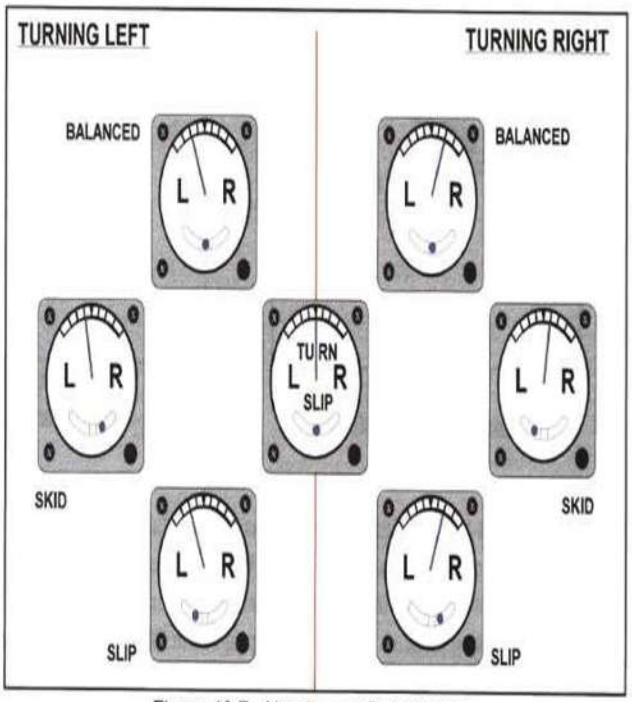


Figure 13.7. Needle and Ball Displays

DIRECTIONAL GYROSCOPE INTRODUCTIO N.

The directional gyro indicator (DGI), often called the 'direction indicator' (DI) provides a stable directional reference in azimuth for maintaining accurate headings and for executing precise turns. There is no magnetic element in the DI, so it is not North- seeking and must initially be synchronized with the magnetic compass.

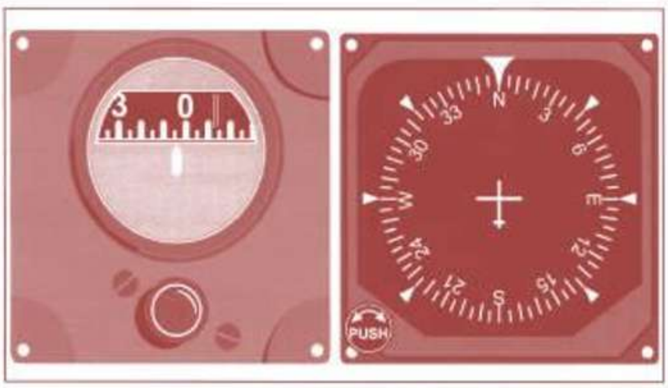


Figure 11.1. Two Directional Gyro Indicators.

The synchronization must be checked at regular intervals because of real and apparent gyro wander (drift). The DGI does not therefore replace the compass; its stable, dead-beat indications are complementary to the North- seeking capability of the compass. Having no magnetic element, the DGI does not suffer from the compass turning and acceleration errors produced by the vertical component of the earth's magnetic field.

THE PRINCIPLE AND CONSTRUCTION OF THE DGI.

The DI employs a tied gyro, that is to say, a gyro having freedom of movement in three planes mutually at right angles but with the rotor avis maintained in the yawing plane of the aircraft. This means that the rotor axis is horizontal in level flight, and because of gyroscopic rigidity it provides the datum from which heading can be measured. The rotor is mounted in the inner gimbal (on bearings mounted in the outer gimbal) which has restricted freedom to turn. The outer gimbal can rotate through 360* about the aircraft's vertical axis, on bearings in the case.

Note that the rotor axis, the inner gimbal axis, and the outer gimbal axis are mutually at right angles. During a turn, the aircraft and instrument case turn on the vertical axis bearings Of the outer gimbal whilst the gyro rotor, gimbals and indicating scale all remain fixed in azimuth because of gyroscopic rigidity. I-leading is indicated on the scale by a lubber line painted on a glass window in the instrument case. Some designs have a circular vertical-card indicating scale geared to the outer gimbal, in place of the cylindrical scale fixed to the outer gimbal

DGI LIMITATIONS.

If the aircraft exceeds the pitch or roll limits of 85* (55* in air driven gyro DI's) the gyro will topple as the inner gimbal comes up against the stops, the precession causing the outer gimbal and scale to spin rapidly. Exceptions to this can occur:-

a) If the rotor axis is athwart ships – 360* of aircraft rotation in the looping plane then being possible without toppling the gyro.

b) If the rotor axis is fore and aft – 360* of roll then being possible without toppling. The actual indications on the scale at which these two situations can arise depend on the vintage and manufacture of the instrument.

DGI ERRORS.

There are several reasons why it is virtually impossible for a DGI to remain synchronized with the compass. The most significant errors are listed below.

a) Gimbal errors.
b) Random wander.
c) Apparent wander due to earth's rotation.
d) Errors resulting from varying rotor rpm.

Apparent wander due to change Of aircraft position (transport wander).

GIMBALLING ERRORS.

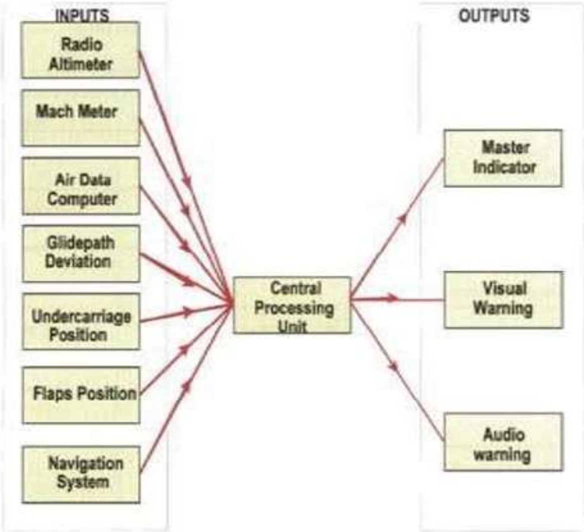
These are errors in the indications of the DGI which occur when bank is applied. If the errors during a 360* turn are plotted, an approximate double sine curve results, with zero error on four headings (90* apart) spaced between alternate positive and negative peaks (two of each). The curve becomes more complex if pitch changes are made during the turn. The actual readings on the DI at which the maximum errors occur depend on its make and mark.

The errors are small, provided deviations in attitude from the level position are only moderate, and they disappear as soon as level flight is resumed. They occur because of the geometry of the gimbal system, in that unless the instrument case (and the aircraft bolted to it) are able to rotate about one of the axes Of the gyro, the outer gimbal itself must move - giving an error - if the rotor axis is to maintain its fixed direction.

GPWS

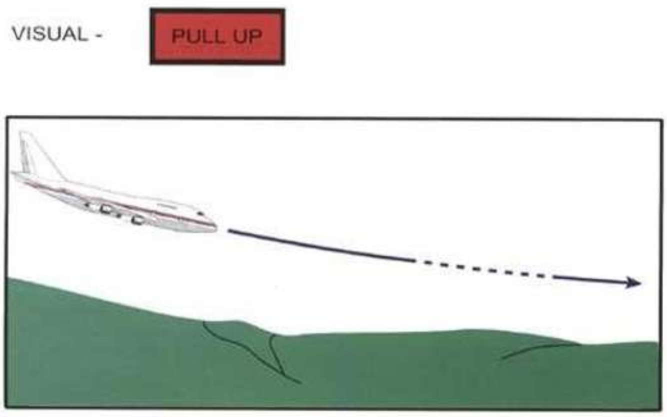
INTRODUCTION

The aim of the system is to give visual and audible warning signals to a pilot when the aircraft's proximity



to the terrain poses a potential threat to its safety. Although not a full proof means of preventing a collision with the earth's surface,

EGPWS enhances night safety and can prevent those accidents which could result from crew errors or distraction, malfunction or misinterpretation of navigational equipment, or inappropriate ATC instructions. Figure shows the three elements Of a GPWS: inputs, outputs and a central processing unit. The Central Processing Unit will also indicate a computer failure and any failures of the six input signals. The system operates between 50' and 2450' actual height above the surface and automatically selects the correct mode of operation.



DEFINITIONS

a)ALERT: A caution generated by the EGPWS equipment. WARNING: A command generated by the EGPWS equipment. b)Types Of Warnings/Alerts

i) Genuine

The equipment provides a warning in accordance with its technical specification.

ii) Nuisance

The equipment provides a warning in accordance with its technical specification, but the pilot is flying an

accepted safe procedure. iii)False
A fault or failure in the System causes the equipment to provide a warning that is not in accordance with its technical specification.

A TABLE OF THE EGPWS OPERATING MODES.

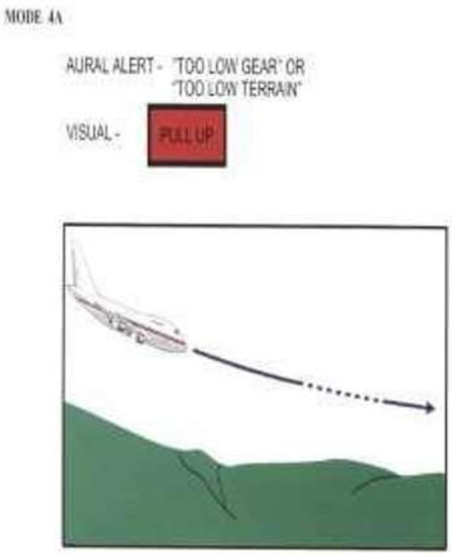
GPWS MODE		ADVANCED EQUIPMENT Alert	Warning
1. Excessive descent rate		'Sink Rate'	'Whoop Whoop Pull Up'
2. Excessive terrain closure rate		'Terrain Terrain'	'Whoop Whoop Pull Up'
3. Altitude loss after take-off or go-around		'Don't Sink'	-
4. Unsafe terrain clearance while not in the landing configuration	4A. Proximity to terrain - Gear not locked down	'Too Low gear'	'Whoop Whoop Pull Up'
	4B. Proximity to terrain - Flaps not in a landing position	'Too Low Flaps'	'Too Low Terrain' (see note below)
5. Descent below glide-slope		'Glideslope'	-
6. Descent below 'minimums'	6A.	'Minimums'	-
	6B.	'Bank angle'	-
7. Windshear warning	-	-	'Wind Shear'

TERRAIN CLEARANCE FLOOR
This alerts the crew to possible premature descent for non- precision approaches regardless of aircraft configuration. It uses the present aircraft position with respect to the runway. It is speculated that in the future the database for this and the EGPWS as a whole will be merged with that for the FMS. This will mean that the EGPWS can work with the latest information and include temporary obstacles normally notified by NOTAM

MODE 1 - EXCESSIVE BAROMETRIC DESCENT RATE
Mode I has two boundaries and is independent of aircraft configuration. Penetration of the first boundary generates an aural alert Of "SINK RATE" repeated each 1.5 seconds. Penetrating the second boundary causes the repeated warning of "WHOOP, WHOOP PULL UP", until the rate of descent has been corrected.

MODE 1
AURAL ALERT - SINK RATE, SINK RATE
AURAL WARNING - "WHOOP WHOOP PULL UP"

MODE 2 - EXCESSIVE TERRAIN CLOSURE RATE
Mode 2 monitors Mach number, radio altitude rate of change, barometric altitude and aircraft configuration. Mode 2 has two boundaries. Penetrating the first boundary causes an aural alert of "TERRAIN, TERRAIN", followed by the repeated aural warning "WHOOP, WHOOP PULL UP'. After leaving the PULL UP area, the repeating TERRAIN message will again be heard while in the terrain portion of the envelope. If both boundaries are penetrated while in the landing configuration, only the repeating TERRAIN aural alert will occur.
The terrain message is repeated each 1.5 seconds. As Mach number increases from 0.35 to 0.45 with gear up, the highest radio altitude at which Mode 2 alert warning will occur is increased to 2450 feet. This higher portion of the envelope is inhibited with the flap override switch in the FLAP OVRD position.
MODE 2
AURAL ALERT - 'TERRAIN, TERRAIN' AURAL WARNING - 'WHOOP WHOOP PULL UP'

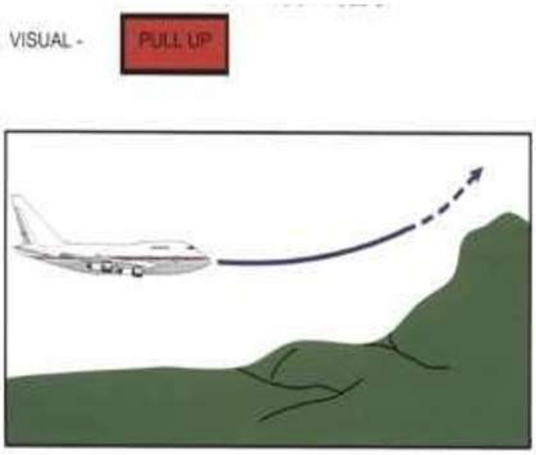


MODE 3 - ALTITUDE LOSS AFTER TAKE- OFF OR GO-AROUND

Mode 3 provides an alert if a descent is made during initial climb or go-around. The aural alert is a voice message Of "DON'T SINK", repeated each I .5 seconds until the flight condition is corrected.

Mode 3 is effective between 50 and 700 feet radio altitude and generates the alert when the accumulated barometric loss equals approximately 10 percent of the existing radio altitude. Mode 3 does not arm during the descent until below 200 feet radio altitude. MODE 3

AURAL ALERT - "DON'T SINK, DON'T SINK"



MODE 4A - UNSAFE TERRAIN CLEARANCE WITH LANDING GEAR NOT DOWN

The terrain clearance mode with gear retracted, is armed after take- off upon climbing through 700 feet radio altitude.

When this envelope is penetrated at less than 0.35 Mach, the aural alert —TOO LOW GEAR" is sounded. When the envelope is penetrated at more than 035 Mach, the aural alert "TOO LOW TERRAIN" is sounded and the upper boundary of the envelope is increased to 1000 feet radio altitude. The applicable voice message is repeated each 1.5 seconds until the flight condition has been corrected.

MODE 4A

AURAL ALERT - "TOO LOW GEAR" OR "TOO LOW TERRAIN"

MODE 4B - UNSAFE TERRAIN CLEARENCE WITH FLAPS NOT

IN LANDING

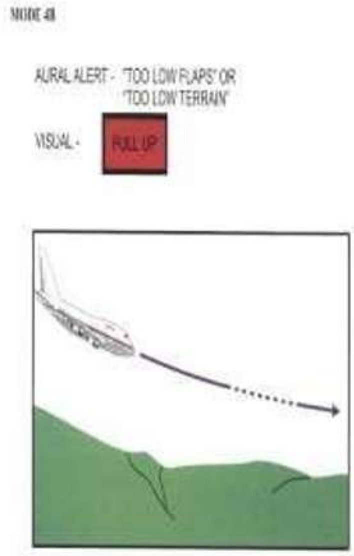
CONFIGURATION

This mode provides an alert when the gear is down and the flaps are not in the landing position. If the envelope is penetrated at less than

0.28 Mach with the flaps not in the landing position, the aural alert of"T00 LOW FLAPS" is sounded. When the envelope is penetrated at more than 0.2* Mach, the aural alert of "TOO LOW TERRAIN" is sounded and the upper boundary of the envelope is increased to 1000

feet radio altitude. The applicable voice message is repeated each

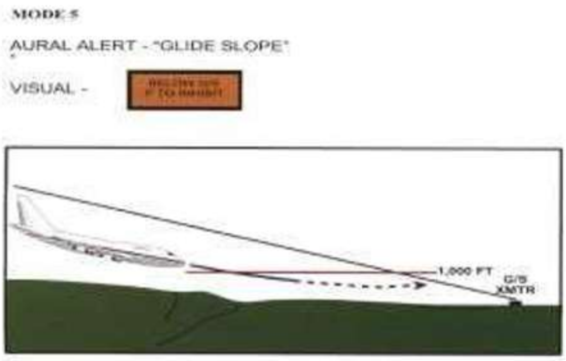
1.5 seconds until the flight condition has been corrected. The "TOO LOW GEAR" alert takes priority over the "TOO LOW FLAPS". The too low flaps alert and associated too low terrain alert are inhibited with the flap inhibit switch in the FLAP OVRD position.



MODE 5 - BELOW GLIDE SLOPE DEVIATION ALERT

This mode alerts the flight crew of a descent of more than 1.3 dots below an ILS glide slope.

The envelope has two areas of alerting, soft and loud. In both

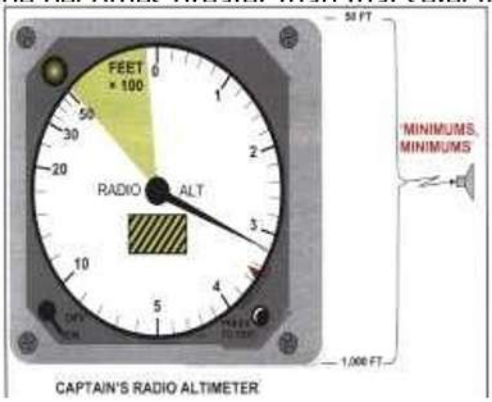


areas, the alert is a repeated voice message of "GLIDE SLOPE",

and illumination of both pilots_BELOW G/S‘ lights. The voice message amplitude is increased when entering the loud area. In both areas, the voice message repetition rate is increased as the glide slope deviation increases and the radio altitude decreases. The mode is armed when a valid signal is being received by the captain's glide slope receiver and the radio altitude is 1000 feet or less. The mode may be cancelled or inhibited by pressing either pilot' s below GIS light while below feet radio altitude. The mode will re- arm when climbing above 1000 feet radio altitude. Mode 1 to 4 aural alerts and warnings have priority over mode 5 aural alerts, however both PULL UP and BELOW G/S lights could be illuminated at the same time.

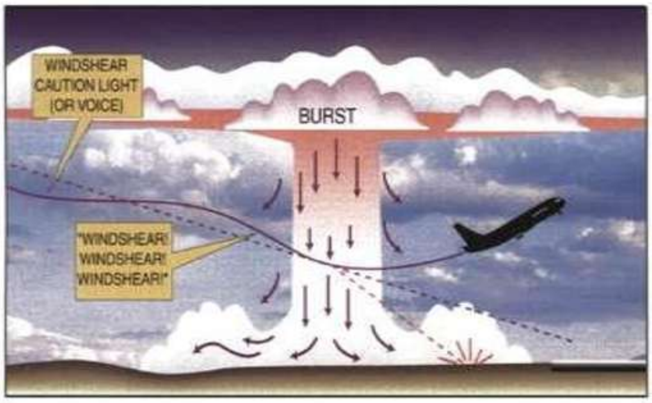
MODE 6A - BELOW SELECTED MINIMUM RADIO ALTITUDE

Mode 6A provides an aural alert if a descent is made below the minimum decision altitude cursor in the captain's radio altimeter. This mode operates between 50 and 1000 feet of radio altitude. This alert is aural only and consists of "MINIMUMS, MINIMUMS" sounded once. The mode is reamed when the radio altitude becomes greater than that selected with the captain's altitude cursor.

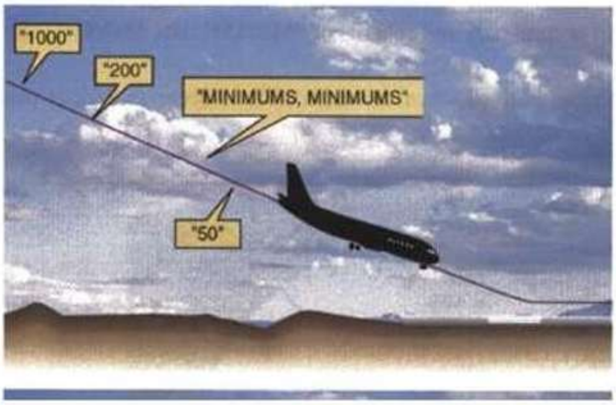


CAPTAIN'S RADIO ALTIMETER

The call outs used are a customer option but for example may consist of calls at 200ft and 100ft to decision height, or absolute height call outs from the radio altimeter with respect to the ground. "BANK ANGLE" can be used to alert crews of excessive roll angles. The bank angles will be specific to each aircraft. The bank angle limit reduces with proximity to the ground due to the reduced wing tip clearance to prevent wing tip or engine damage during take off and landing.



MODE 7 :WINDSHE AR ALERTING Visual and aural wind shear warnings are given when several parameters such as ground speed, airspeed, barometric height and rate of descent and radio altitude, indicate the initial conditions of entering an area of wind shear. Again as with the terrain threat display there is no scanning beam looking ahead to avoid the condition entirely. Rather the benefit from the system is derived from the fact that it allows the pilot to initiate the wind shear go- around procedure earlier, giving the aircraft a greater probability of avoiding an accident.



COMPASS SYSTEM THE MAGNETIC COMPASS.

A compass is an instrument designed to indicate direction on the surface of the earth relative to Some known datum. The magnetic compass uses the horizontal component of the earth's field as its directional datum. Unfortunately, the earth's field is normally not aligned with the true meridian - the most desirable datum from which to measure direction. The angular difference between true and magnetic meridians is called the magnetic variation The purpose of a magnetic —steering‖ compass in an aircraft is to indicate heading, the direction in which the aircraft is pointing.

Magnetic influences - iron/steel components, electric currents - distort the earth's field so that the compass magnet assembly deviates from the magnetic meridian. This is called compass deviation. The rules for applying variation and deviation to the compass heading indication in order to determine true heading

DIRECT INDICATING MAGNETIC COMPASS.

It deals with the direct indicating or direct reading magnetic compass, where the pilot directly reads his heading in relation to the pivoted magnet assembly. There are two basic types of direct reading magnetic compasses used in aircraft, the vertical card and, less commonly, the grid ring compass.

THE VERTICAL CARD COMPASS.

The vertical card compass - which is also known as the B-type or E-type - is the direct reading compass in general use. It is usually the main magnetic heading reference in light aircraft and the standby compass in larger aircraft. It consists of a circular compass card attached directly to the magnet assembly. This combined unit is suspended in liquid within the compass bowl. A vertical lubber line on the glass window of the bowl, enables the heading to be read off the compass card.

THE GRID RING COMPASS.

The P-type compass or grid ring compass is found on older aircraft. It is more accurate than the vertical card compass and is more stable. It is however heavier, bulkier and more expensive. In addition it can only be read in straight and level flight, as the grid ring has to be unclamped and aligned with the north reference before a reading can be taken against the lubber line. The grid ring compass also differs from the vertical card compass in that it achieves a greater periodicity by the addition Of damping wires which also rotate through the compass liquid.

COMPASS REQUIREMENTS.

The direct reading magnetic compass contains a pivoted magnet which must be able to align itself, and remain aligned, with the horizontal component of the earth's magnetic field. For the compass to succeed, certain requirements must be satisfied. The most important of these are that the magnet system must be:

a. Horizontal
b. Sensitive
c. Aperiodic HORIZONTALITY.

In order to measure direction in the horizontal, the magnets must lie as nearly as possible in the horizontal plane during normal straight and level flight. A freely suspended magnet assembly would align itself with the earth's total field so the magnets would only be horizontal at the magnetic equator. To achieve horizontality, the magnet assembly is 'pendulously suspended' the centre of gravity of this assembly being lower than its supporting pivot

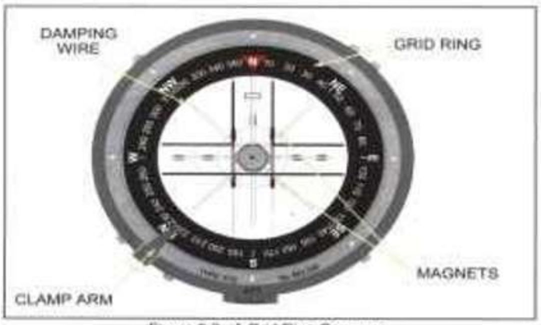


Figure 9.2. A Grid Ring Compass.

SENSITIVITY.

The magnet System is required to seek the horizontal component H of the earth's field in all areas except near the magnetic poles - where the horizontal component is inadequate. The notes on magnetism show that the ability of a pivoted magnet to align itself with an external field - its sensitivity - depends on the strength of the external field and on the magnetic moment of the magnet. The weak external field (H) at a place cannot be changed, but the magnetic moment of the magnet can be increased - by increasing the magnet's length and/or pole strength. It is however undesirable to increase the magnet length so the pole strength is increased by using two, four or Six short magnets or a circular magnet, made of an alloy which will accept and retain the high degree of magnetism required. Sensitivity is further increased by reducing friction. This is achieved in three ways:

a) By using an iridium-tipped pivot in a jeweled cup

b) By lubricating the pivot with the liquid which fills the compass bowl.

c) By reducing the effective weight of the magnet assembly acting down through the pivot, because the liquid that the magnet assembly is displacing is denser than air

APERIODICITY.

The magnetic assembly is required to be aperiodic or 'dead beat', which means that it should Settle down quickly on a steady indication after being displaced by turbulence or manoeuvres. Any tendency to oscillate must be quickly 'damped out'. The desired aperiodicity is achieved as follows:

a. Several short magnets are used instead of one longer one. This keeps the mass Of the assembly near the centre, so reducing the moment of inertia and consequently making any oscillations easier to damp out. Light alloy is utilised wherever possible in order to minimise the weight of the assembly framework.

b. The primary purpose of the liquid in the compass bowl is to act as a damping liquid on the compass assembly. The grid ring compass dampens oscillations more rapidly than the vertical card compass, due to addition of damping wires.

THE COMPASS LIQUID

The liquid mentioned earlier is essential to the design of the compass. Two difficulties may arise. Firstly, the liquid is likely to expand or contract with change Of temperature; this is overcome by incorporating an expansion chamber or 'Sylphon tube'. Secondly, errors occur in sustained turns as a result of liquid swirl'. Liquid swirl occurs due to the viscosity Of the liquid, because Of this the liquid chosen should have a low viscosity to minimize liquid swirl. Various liquids, including alcohol have been used. The main properties required of a compass liquid are: Various liquids, including alcohol have been used. The main properties required of a compass liquid are:

a) Low coefficient of expansion
b) Low viscosity
c) Transparency
d) Low freezing point
e) High boiling point
f) Non-corrosiveness

SERVICEABILITY CHECKS.
Compass body. Check that there is no obvious damage such as dents or cracks. Any lighting system should be checked, as should the efficacy Of the luminous paint.
Compass Liquid. The compass liquid should be checked and be free from:
a) Sediment and discoloration - either of which would indicate corrosion which would result in increased pivot friction.
b) Bubbles - which would probably indicate a leaking seal.
Turbulence and manoeuvres
would cause any bubbles to move about, creating eddies which could disturb the magnet system.

ACCELERATION AND TURNING ERRORS.
Direct reading compasses are subject to large errors during linear acceleration or deceleration, or during a turn. Most manoeuvres which cause the centre of gravity of the magnet assembly to move away from its normal position, almost directly below the pivot, will produce an error. However, if the manoeuvre displaces the centre Of gravity North or South Of its usual position So that cg and pivot are still in the plane of the magnetic meridian, the magnet assembly merely changes its North-South tilt angle, with no rotation in azimuth and consequently no error.

Note also that turning and acceleration errors only occur where there is a significant vertical component (Z) in the earth's field, so that except for a small liquid swirl effect in turns, the errors are non-existent near the magnetic equator. The north seeking end of the compass magnet should remain pointing in the same direction (i.e. Magnetic north) whether the aircraft is moving in a straight line or turning Acceleration and turning errors occur however when the north seeking end of the magnet is displaced from Magnetic north and therefore an incorrect heading will be shown on the compass card which is attached to the magnet. Figure shows a pendulously suspended magnet (with residual dip) in the northern hemisphere.

TURNING ERRORS.
Turning errors are maximum when turning through north and south, and ignoring liquid swirl zero when turning through east and west. The basic theory of turning errors is much the same as that for linear acceleration errors. Due to the earth's vertical component of the magnetic field, Z, the compass's cg will be displaced from almost beneath the pivot point away from the nearer pole. In a turn, the aircraft accelerates towards the centre of the turn, and therefore an acceleration force acts through the pivot towards the centre of the turn, while the opposing centrifugal force due to inertia acts outward through the cg. This results in the magnet assembly tending to 'swing out' from the turn, rotating the magnet assembly around the pivot point and producing a turning error.
Turning errors are usually more significant than acceleration errors for the following reasons: -
a) They are inherently of greater magnitude because greater displacement of the magnet assembly is likely in turns.
b) Tums occur more often and are likely to be more prolonged than linear accelerations.

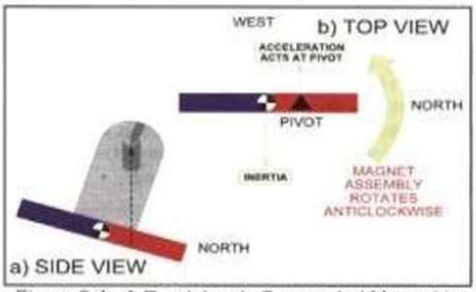


Figure 9.4. A Pendulously Suspended Magnet in the Northern Hemisphere.

FDR

INTRODUCTION

Commercial aircraft have a flight recorder which records various aircraft parameters during the entire duration Of the flight The main function of the flight data recorder (FDR) is to preserve the aircraft data in order to determine the cause of any aircraft accident. It is also used to gather information for trend analysis and trouble shooting. In smaller aircraft the FDR may be combined with a cockpit voice recorder.

FDR DESIGNS

The FDR records the last 10 or 25 hours of aircraft data on a digital storage device housed in a fire and shock resistant box. The box that is painted red or orange and located at the rear of the aircraft, normally under the fin. On the front of the unit is an underwater locating device (ULD). The older type of FDR's are of non digital design while the JAR now states that on aircraft registered as of the I April 2000, all FDR's must be of the digital type.

FDR COMPONENTS

The FDR consists of the following components a)a recording System

a) a control unit On the overhead panel
b) a control unit on the pedestal

The recording system includes a digital flight data recorder (DFDR), a flight data interface unit (FDIU) and a 3-axis linear accelerator (LA). The control unit on the overhead panel also controls the cockpit voice recorder (CVR). A loaded switch labelled GND CTL can be selected ON or AUTO as follows:

a) ON : The CVR and the DFDR are energised and the ON light is lit
b) AUTO : The CVR and the DFDR are energised: i)on the ground whenever electrical power is applied

ii) on the ground with one engine running
iii) in flight (with engine running or stopped)

The control on the pedestal consists simply of push button labelled 'EVENT' which sets an event mark on the DFDR recording. This acts as a kind of bookmark to enable the "event" to be found rapidly on the recording at a subsequent analysis.

When On the ground the FDR is automatically stopped 5 minutes after the final engine shut- down. A block diagram Of a digital flight data recorder system for a Boeing 767 aircraft is shown in
Figure

AIRCRAFT INTEGRATED DATA SYSTEMS (AIDS)

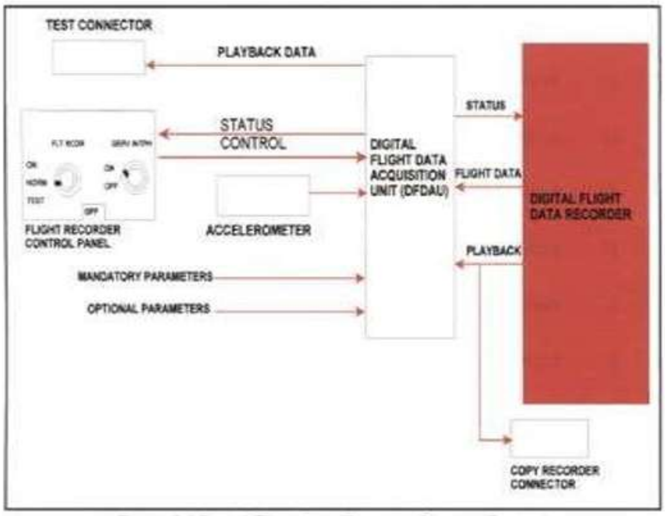
AIDS processes data for various aircraft systems to ease maintenance tasks. This is done via a data management unit (DMU) that collects and processes data to compile reports for

storage and for printing. Some of this information is sent to the FDR via the flight data interface unit (FDIU) for recording mandatory parameters of the flight. The rest of the information is recorded on a separate flight maintenance recorder from which data can be printed Out for the purpose of maintenance. Data stored on the DMU can be printed out both in flight as well as on the ground for the purpose of maintenance. It is also possible to transmit relevant data from AIDS to ground at certain intervals so that the aircraft performance can be monitored from the ground. This is done via ACARS (Airborne Communications and Reporting System) on a VHF data link.

PARAMETERS RECORDED

The mandatory aircraft parameters recorded on the FDR depend upon the age and size of the aircraft and are specified in JAR OPS.

The main parameters are: 1)altitude
2) airspeed 3)heading 4)acceleration 5)pitchand roll attitude 6)radio transmission keying 7)thrust or
power on each engine 8)configuratio n Of lift and drag devices 9)air temperature
10) use Of automatic flight control Systems and
11) Angle of attack



ELECTRONIC INSTRUMENTATION

ELECTRONIC INSTRUMENTS FOR ENGINE AND AIRFRAME SYSTEMS CONTROL.

The display of the parameters associated with engine performance and airframe systems control by means of Cathode Ray Tube type display units has, like those of flight instrument systems, become a standard feature of many types of aircraft.
The display units form part of two principal systems designated as
Engine Indicating and Crew Alerting System (EICAS), and Electronic Centralized Aircraft Monitoring (ECAM) system.

These systems were first introduced in Boeing 757 and 767 aircraft and the Airbus A310 respectively.
At the time of their introduction there were differing views on the approach to such operating factors as

flight deck layouts and crews' controlling functions, the extent to which normal, alerting and warning information should be displayed, and in particular, whether engine operating data was required to be displayed for the whole of a flight, or only at various phases

In respect of EICAS, engine operating data is displayed on its CRT units, thereby eliminating the need for traditional instruments. This data, as well as that relevant to other systems, is not necessarily always on display but in the event Of malfunctions occurring at any time, the flight crew's attention is drawn to them by an automatic display of messages in the appropriate colours. The ECAM system, on the other hand, displays systems' operation in checklist and schematic form, and as this was a concept based on the view that engine data needed to be displayed during the whole of a flight, traditional instruments were retained in the Airbus A310. It is of interest to note, however, that in subsequent types produced by this manufacturer, e.g. A320, the ECAM system is developed to include the display of engine data in one of its display

EICAS.

The basic EICAS system comprises two display units, a control panel, and two computers
supplied with analog and digital signals from engine and system sensors.

The computers are designated 'Left' and Right', and only one is in control at a time; the other is on 'standby', and in the event of failure it may be switched in either manually or automatically. Operating in conjunction with the EICAS system are discrete caution and warning lights, standby engine indicators and a remotely located panel for selecting maintenance data displays.

The system provides the flight crew with information on primary engine parameters (full-time), with secondary engine parameters and advisory/caution/warning alert messages displayed as required.

DISPLAY UNITS

The display units provide a wide variety Of information relevant to engine operation, and operation of other automated systems, and they utilize colour shadow mask CRTs and associated card modules whose functions are identical to those Ofthe EFIS units. The units are mounted one above the other as shown in Figure.

The upper unit displays the primary engine parameters, NI speed, EGT, and warning and caution messages. In some cases this unit can also display EPR, depending on the type of engines installed and on the methods used to process data by the thrust management control system. The lower unit displays secondary engine parameters, i.e. N2 speed, fuel flow, oil
quantity, pressure and temperature, and engine vibration. In addition, the status of non-engine systems, e.g. flight control surface positions, hydraulic system, APU, etc., can also be displayed together with aircraft configuration and maintenance data. The rows of 'V's shown On the upper display unit Only appear when secondary information is being displayed on the lower unit.

Seven colours are produced by the CRTs and they are used as follows:

White : All scales, normal operating range of pointers, digital readouts.

Red : Warning messages, maximum operating limit marks on scales, and digital readouts.
Green : Thrust mode readout and selected EPR / N, speed marks or target cursors.
Blue : Testing of system only.
Yellow : Caution and advisory messages, caution limit marks on scales, digital readouts.
Magenta : During in-flight engine starting, and for cross-bleed messages.
Cyan : Names of all parameters being measured (e.g. N I oil pressure, TAT etc) and status marks or cues.

DISPLAY MODES

EICAS is designed to categorize displays and alerts according to function and usage, and for this purpose there are three modes Of displaying information: (i) operational, (ii) status and (iii) maintenance. Modes (i) and (ii) are selected by the flight crew on the display select panel, while mode (iii) is selected on the maintenance panel which is for the use of ground engineering staff only.

OPERATIONAL MODE.

The operational mode displays the engine operating information and any alerts required to be actioned by the crew in flight. Normally only the upper display unit presents information, the lower one remains blank and can be selected to display secondary information as and when required.

STATUS MODE.

When selected, the status mode displays data to determine the dispatch readiness of an aircraft, and is closely associated with details contained in an aircrafts Minimum Equipment List. The display shows positions Of the flight control surfaces in the form of pointers registered against vertical scales, selected sub-system parameters, and equipment status messages on the lower display unit. Selection is normally done on the ground either as pan Of pre-flight checks of dispatch items, or prior to shut-down Of electrical power to aid the flight crew in making entries in the aircraft's Technical Log.

MAINTENANCE MODE.

This mode provides maintenance engineers with information in five different display formats to aid them in trouble-shooting and verification testing of the major sub-systems. The displays, which are presented on the lower display unit, are not available in flight.

DISPLAY SELECT PANEL

This panel, as indicated in Figure 2.3, permits control of EICAS functions and displays and can be used both in flight and on the ground. It is normally located on the centre pedestal of an aircraft's flight deck, and its controls are as follows:

a) Engine Display Switch. This is of the momentary-push type for removing or presenting the display of secondary information on the lower display unit.

b) Status Display Switch. Also of the momentary-push type, this is used to display the status mode information referred to earlier, on the lower display unit. The display is known as a 'status page'. c)Event Record Switch. This is of the momentary-push type and is used in the air or on the ground, to activate the recording of fault data relevant to the environment control system, electrical power, hydraulic system, performance and APC. Normally if any malfunction occurs in a system, it is recorded automatically (called an 'auto event') and stored in a non-volatile memory of the EICAS computer. The push switch also enables the flight crew to record a suspected malfunction for storage, and this is called a 'manual event'. The relevant data can only be retrieved from memory and displayed when the aircraft is on the ground and by Operating switches on the maintenance control panel.

d) Computer Select Switch. In the 'AUTO' position it selects the left, or primary, computer and automatically switches to the Other computer in the event of failure. The other positions are for the manual selection Of left or right computers.

e) Displays Brightness Control. The inner knob controls the intensity Of the displays, and the outer knob controls brightness balance between displays.

f) Thrust Reference Set Switch. pulling and rotating the inner knob positions the reference cursor On the thrust indicator display (either EPR or NI) for the engine(s) selected by the outer knob. g)Maximum Indicator Reset Switch. If any one of the measured parameters e.g. oil pressure, EGT, should exceed normal operating limits, this will be automatically alerted on the display units. The purpose of the reset switch is to clear the alerts from the display when the excess limits no longer exist.

ALERT MESSAGES.

The system continuously monitors a large number of inputs (typically over 400) from engine and airframe systems' sensors and will detect any malfunctioning of systems. If this should occur , then appropriate messages are generated and displayed On the upper display unit in a sequence corresponding to the level of urgency Of action to be taken. Up to 11 messages can be displayed, and at the following levels:
Level A - Warning requiring immediate corrective action. They are displayed in red. Master warning lights are also illuminated, and aural warnings (eg fire bell) from a
central warning system are given.

Level B - Cautions requiring immediate crew awareness and possible action. They are displayed in amber, and also by message caution lights. An aural tone is also repeated twice.

Level C - Advisories requiring crew awareness. Also displayed in amber. No caution
lights or aural tones are associated with this level.

DISPLAY UNIT FAILURE.

If the lower display unit should fail when secondary information is being displayed on it, an amber alert message appears at the top left of the upper display unit, and the information is The format of this display is referred to as 'compact', and it may be removed by pressing the 'ENGINE' switch on the display select panel. Failure of a display unit causes the function of the panel 'STATUS' switch to be inhibited so that the status page format cannot be displayed.

DISPLAY SELECT PANEL FAILURE.

If this panel fails the advisory message 'EICAS CONTROL PANEL' appears at the top left of the upper display unit together with the primary information, and the secondary information automatically appears on the lower display unit. The 'CANCEL/RECALL' switches do not operate in this failure condition.

MAINTENANCE CONTROL PANEL.

The maintenance control panel is for use by maintenance engineers for the purpose of displaying maintenance data stored in system computer memories during flight or ground operations. The layout of the panel and the principal functions of each of the controls are shown in Figure

ELECTRONIC CENTRALIZED AIRCRAFT MONITORING (ECAM).

The units comprising this system, and as originally developed for the Airbus A310, are shown in the functional diagram of Figure As far as the processing and display of information are concerned, the
ECAM system
differs significantly from EICAS in that data
relates essentially to the primary systems of the aircraft, and is displayed in check-list and pictorial or
synoptic format. Engine operating data is displayed by conventional types of instruments as noted in the introduction to this chapter. Other differences relate to display locations and selection of system Operating modes.

DISPLAY UNITS.

These units may be mounted side-by-side; the left- hand unit is dedicated to information on the status of systems, warnings and corrective action in a sequenced check-list format, while the right-hand unit is dedicated to associated information in pictorial or synoptic format.

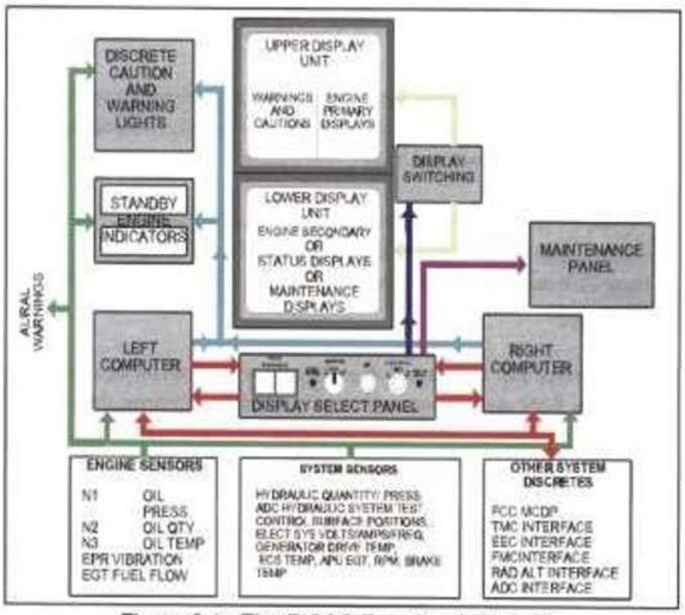


Figure 2.1. The EICAS Functional Diagram.

DISPLAY MODES.

There are four display modes, three of which are automatically selected and referred to as:-
a) Flight Phase-related
b) Advisory (mode and status) c)Failure-related modes.
The fourth mode is manual
d) Aircraft System Display.

THE 'ECAM CONTROL PANEL.
The layout of the 'ECAM' control panel is shown in Figure, all switches, with the exception Of those for display control, are of the push-button, illuminated caption type.

a) SGU Selector Switches. These control the respective symbol generator units, and the lights are off in normal operation of the system. The 'FAULT' caption is illuminated amber if a failure is detected by an SGU's internal self-test circuit. Releasing a switch isolates the corresponding SGU , and causes the 'FAULT' caption to extinguish, and the 'OFF caption to illuminate white.

b) Synoptic Display Switches. These permit individual selection of synoptic diagrams corresponding to each of 12 systems, and
illuminate white when pressed. A display is automatically cancelled whenever a warning or advisory occurs.
c) CLR Switch. The light in the switch is illuminated white whenever a warning or status message is

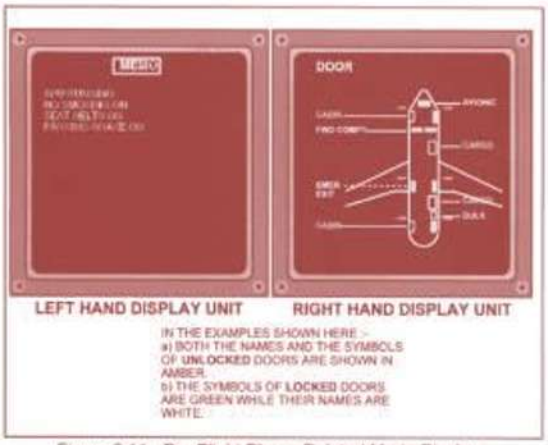


Figure 2.11. Pre-Flight Phase-Related Mode Display.

displayed on the left-hand display unit. The Switch is pressed to clear messages.

The displays are selected according to an appropriate display selection mode.

Figure 2.2. EICAS Engine Data Displays.

d) STS Switch. The Status Switch permits manual selection of an aircraft status message If no warning is displayed; illuminated white. Pressing the switch also causes the 'CLR' switch to illuminate. A status message is suppressed if a warning occurs or if the 'CLR' switch is pressed.

e) RCL Switch. The Recall Switch enables previously cleared warning messages to be recalled provided the failure conditions which initiated them still exist. Pressing the switch also causes the CLR switch light to illuminate. If a failure no longer exists the message 'NO WARNING PRESENT' is displayed on the left-hand display unit.

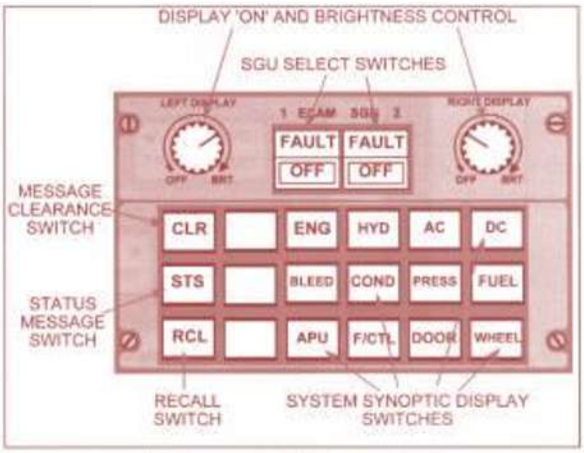


Figure 2.15. The ECAM Control Panel.

THE ELECTRONIC FLIGHT INSTRUMENT SYSTEM (EFIS).

The Electronic Flight Instrument System presents attitude and navigation information to the pilot on two electronic display units in a format that is easier to read and less likely to be misinterpreted than some older mechanical instruments. As far as the pure basic functions and number of display units are concerned, this system, (which is generally referred to as 'EFISV), is fully integrated with digital computer-based navigation systems, and utilizes colour

Cathode Ray Tube (CRT) or Liquid Crystal Display (LCD) types of Attitude Director Indicator (ADV) and Horizontal Situation Indicator (HSI). The System is therefore extremely advanced, not only in terms of physical construction, but also in the extent to which it can present attitude and navigational data to the flight crew of an aircraft.

THE UNITS OF A SYSTEM.
As in the case of a multi crew conventional flight director System, a complete EFIS installation is made up of left (Captain), and right (First Officer), systems.

Each system in turn is comprised of:
a) Electronic Attitude Director Indicator (EADI) or Primary Flight Display (PFD).
b) Electron ic Horizontal Situation Indicator (EHSI) or
Navigation Display (ND).
c) Control Panel. d)Symbol Generator (SG).
e)Remote Light Sensor Unit.
A third (centre) Symbol Generator is also incorporated so that its drive signals may be switched to either the left or right display units in the event of failure Of their corresponding Symbol Generators. The signal switching is accomplished within the left

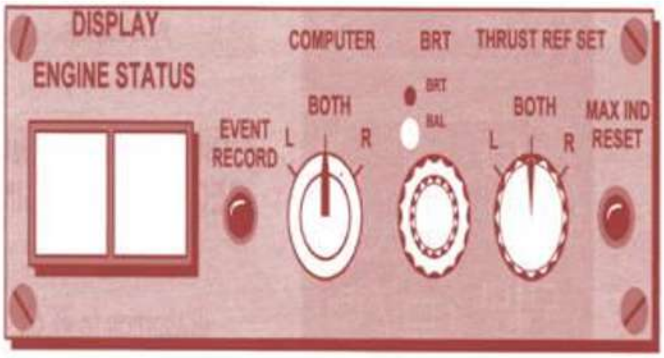


Figure 2.3. EICAS Display Select Panel.

and right Symbol Generators, using electromechanical relays powered from an aircraft's DC power supply, via pilot-controlled switches.

SYMBOL GENERATORS (SGS)
Symbol Generators provide the analogue, discrete, and digital signal interfaces between an aircraft's systems, the display units, and the control panel, and they also perform symbol generation monitoring, power control and the main control functions of the 'EFIS' overall.
DISPLAY UNITS.
The display units may be Cathode Ray Tubes (CRT's) or Liquid Crystal Displays (LCD's). LCD's have the advantage of being smaller and generate less heat therefore need
less cooling. The PFD and ND are usually identical units to facilitate spares commonality and are often interchangeable with the systems display units (EICAS or ECAM).

THE COLOUR DISPLAY SYSTEM.

There is no set colour standard and so colour displays may vary slightly. In a typical display system, 5 colours are usually assigned for the display of the many symbols, failure annunciators, messages and Other alphanumeric information, with a sixth colour (RED) for weather (WXR):

White: Display Of present situation information.
Green: Display of present situation information where contrast with white symbols is required, or for data having lower priority than white symbols. Engaged autoflight modes

Magenta: All 'fly to' information such as flight director commands, deviation pointers, active flight path lines.
Cyan: Sky shading on an EADI and for low-priority information such as non active flight plan map data.
Yellow: Ground shading on an EADI, caution information display such as failure warning flags, limit and alert annunciators and fault messages.
Red: For display of heaviest precipitation levels as detected by the weather radar (WXR).
THE REMOTE LIGHT SENSOR.
The Remote Light Sensor is a photodiode device which responds to ambient light conditions on the flight deck, and automatically adjusts the brightness of the CRT displays to an acceptable level.

THE CONTROL PANEL.
A control panel is provided for each system, and are typically, as shown in Figure, the switches are grouped for the purpose of controlling the displays Of their respective EADI and EHSI units.

THE 'EADI' SECTION OF THE CONTROL PANEL

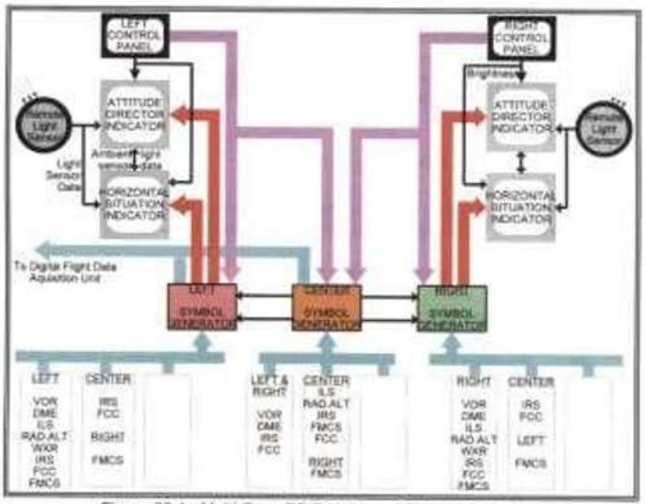
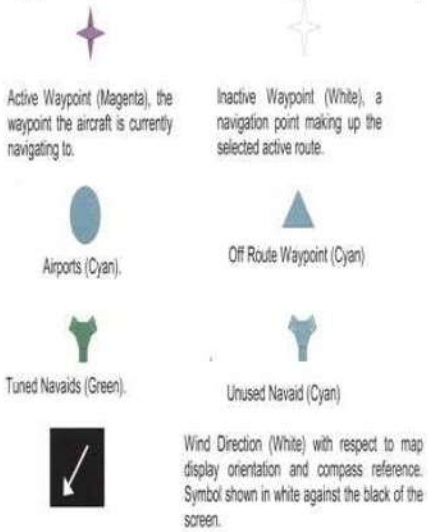


Figure 22.1. Multi-Crew EFIS Units and Signal Interfacing

THE SYSTEM SYMBOL
DECISION HEIGHT (DH).
Decision height is the wheel height above the runway elevation by which a go-around must be initiated unless adequate visual reference has been established and the aircraft position and approach path have been assessed as satisfactory to continue the approach and landing in safety.



THE —EHSI‖ SECTION OF THE CONTROL PANEL

INSTRUMENT WARNING SYSTEM INCLUDING MASTER WARNING SYSTEMS AND CENTRALIZED WARNING PANELS

INTRODUCTION

The purpose of the Flight Warning System (FWS) is to produce cautions and warnings for the crew to increase their situation awareness and to give them suitable indications of the action necessary to avoid impending danger. The proliferation of various warning systems in today's aircraft poses a severe problem in that the crew could be confused by the multiplicity of warnings. It is therefore necessary to install an integrated flight warning system

that will prioritize the warnings. By producing warnings relevant to a particular stage Of flight and inhibiting Other warnings the system enables the crew to respond to the warning posing the most immediate threat to safety.

LEVELS OF ALERTS

The alerting and warning system produces the following levels of alerts:

A) Warnings or Level A alerts. These require immediate crew action. Warnings must attract the pilot's attention in sufficient time for appropriate action to be taken.

B) Cautions or Level B alerts. These require immediate crew alertness and possible future action.

C) Advisories or Level C alerts. These require crew alertness.

WARNINGS IN GENERAL

The alerting and warning messages are presented to the crew in visual, aural and sensory forms.

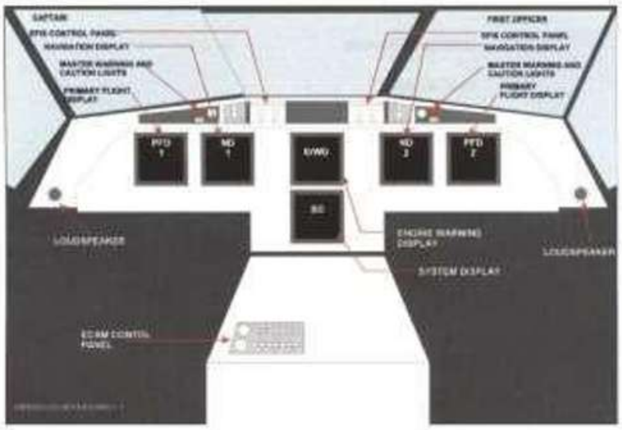
a) VISUAL. The level of alert is indicated by colours as follows:

1)Warnings are presented in Red 2)Cautions are shown in Amber or yellow

3) Advisories are also shown in Amber or yellow

These visual indications can be presented in two different forms: i)Electronic Screens. Alerts and warnings appear in coloured text or symbols on various electronic screens (flight, navigation, engine and aircraft system displays).

ii) Lights or Flags. Red lights or reflective flags signify warnings and require remedial action if flight is to continue. An amber light or flag is used to indicate that a system or equipment is approaching a limit of normal function and that corrective action is necessary to prevent further deterioration and consequent failure.



Additionally, master warning and caution lights are normally provided and are located near the centre of scan in front of each pilot.

b) AURAL An
audible warning is mandatory if the pilot is required to assume control.
This can be in a variety of forms depending upon the type of aircraft. The alert can be in the form of sounds or synthetic voice messages or a combination of both.

* Warnings
Boeing aircraft produce the following aural warnings: 1)A bell accompanies Fire messages
2) A siren accompanies warnings on Cabin Altitude, Configuration and Overspeed
3) A Wailer accompanies Autopilot disconnect
4) Synthetic Voice messages for ground proximity, windshear, airborne collision avoidance.
Airbus aircraft produce :
1)continuous repetitive chimes (red warnings) 2)cavalry charge (autopilot disconnect) 3)cricket sound (stall warning)
4)synthetic voice (GPWS, TCAS warnings)

ii) Cautions : Beepers with various tones or chimes or musical chords are used to caution the crew to potential threats to safety.

c) SENSORY: A vibratory mode on the controls is used to indicate stall approach and demands immediate action to avert loss of control. In some aircraft a stick-pusher provides guidance to prevent a further deterioration of the situation that demanded the vibratory warning.

To rationalize warnings systems, a Master Warning Indicator light is often provided near the centre of scan. In Older systems the crew member would then refer to a Master Warnings Panel where warnings were assembled in a rational order and annotated. In the modern Electronic Instrumentation Systems most of the alerts and warnings appear on appropriate electronic screens together with associated aural messages and master warning lights.

The cockpit displays and warnings of Airbus 320
The Flight Warning System generates alerts and warnings for the following situations:

1) Engine and Airframe systems malfunctions 2)Aerodynamic limits exceeded
3)Presence of external Hazards.

Aerodynamic limits
If aerodynamic limits are exceeded the FWS provides the following alerts to the crew:
1) Altitude Alerting 2)Overspeed Warning 3)Stall Warning
External Hazard Warning

The external hazards that constitute a threat to aircraft safety are proximity to terrain and to other aircraft. These hazards can be avoided by the use of:
1) the Ground Proximity Warning System and 2)the Airborne Collision Avoidance System.

FWS COMPONENTS

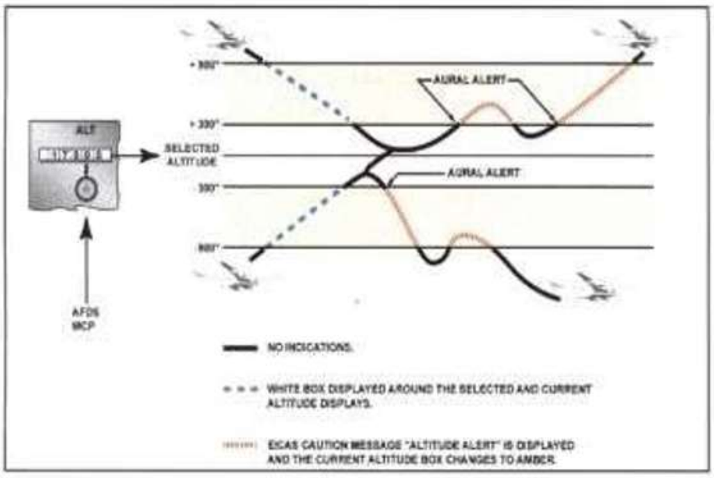
The FWS system comprises:

a) Inputs. There are inputs from various sources including hundreds of engine and airframe sensors, air data sensors, GPWS and ACAS systems.

b) A processing unit. This is made up of one or two flight warning computers.

c) Outputs. The outputs are classified either as alerts or as warnings and are generated according to the nature of the malfunction or threat to safety. Alerts can be visual (amber lights or text on VDU's) or aural (chimes or tones). Warnings are given in the form of red lights or by red text on electronic screens (steady or flashing) as well as aural signals (siren, bell, hooter). Additionally there are red and amber lights on the glaresheild in front of the pilots to act as attention getters.

STALL WARNING AND ANGLE OF ATTACK SYSTEM



INTRODUCTION

The Flight Warning System alerts the crew if there are deviations from certain aerodynamic parameters like altitude, airspeed and angle Of attack. The system therefore provides the following alerts and warnings: a)Altitude Alerting System b)Overspeed Warning c)Stall Warning.

ALTITUDE ALERTING SYSTEM

a) Function

The function of the Altitude Alerting System is to warn the pilots that the aircraft is approaching or deviating from the altitude selected on the Autopilot control panel. It does this in certain height bands above and below the selected altitude.

b) Operation

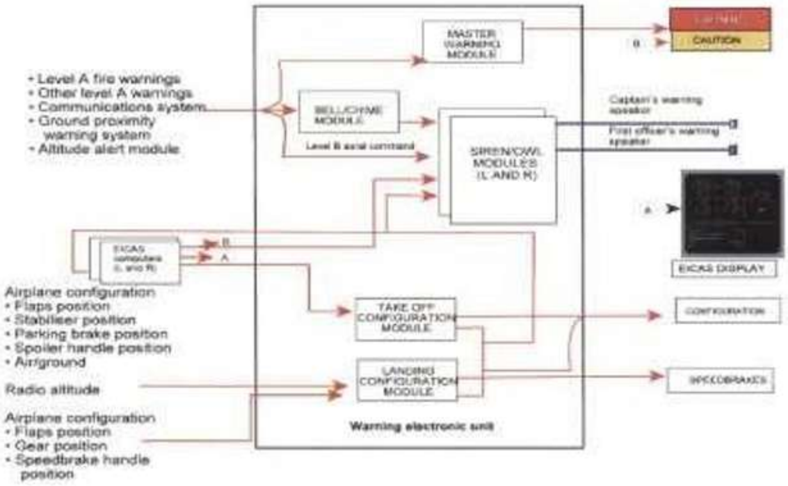
The height bands within which altitude alerting operates are typically 300 feet to 900 feet for Boeing aircraft and 250 feet to 750 feet for Airbus aircraft.

OVERSPEED WARNING

a) Function

The purpose Of the Overspeed warning System is to alert the flight crew if the airspeed exceeds the Vmo / Mrno limits calculated by the air data computer (ADC).

b) Operation



Whenever an Overspeed situation occurs in an aircraft with electronic instrumentation the system:

i) sounds the siren or horn

ii) illuminates the red master WARNING lights

iii)displays the message OVERSPEED on the EICAS upper display in red.

The warning continues while the overspeed situation exists and cannot be cancelled by depressing the red master WARNING light switch. The system obtains its input from the air data computers (ADC's) via the flight warning System. It can be tested on the ground before flight by pressing a test switch which would then sound the siren Or horn as appropriate for that aircraft. In case of System failure the pilot would receive no warning if Vmo or Mmo is exceeded.

c) Displays

The maximum allowable speed is Shown on the airspeed indicator by means of a barber's pole on a conventional meter and on the

airspeed tape On the primary flight display or EADI Of an EFIS display. The barbers pole indicates the Vrno up until the Mmo (when
expres sed in terms Of an indica ted air speed) becom es limitin
g. The barber s pole will
the move counter- clockwise to indicate the maximum allowable speed. As altitude increases when climbing at a constant indicated airspeed the Mmo when expressed as an indicated
airspeed will decrease.

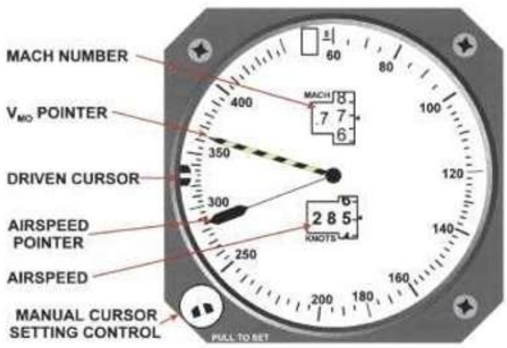


Figure 2.3 Conventional ASI with Vmo pointer

STALL WARNING SYSTEM

a) Function

The purpose of the stall warning system is to warn the pilot of an impending stall. It does so when the aircraft approaches the stalling angle Of attack for the current speed and configuration Of the aircraft.

b) Stall Warning Systems.

The simplest form of system, and one which is adopted in several types of small aircraft, consists of a hinged-vane-type sensor mounted in the leading edge of a wing so that the vane protrudes into the airstream.

In normal level flight conditions, the airstream maintains the vane in line with the relative air flow. If the aircraft's attitude changes such that a increases, then, by definition, the airflow will meet the leading edge at an increasing angle, and so cause the vane to be deflected.

When a reaches that at which the warning unit has been preset, the vane activates a switch to complete a circuit to an aural warning unit in the cockpit. In larger types of aircraft, stall warning and prevention systems are designed to perform a more active function, such as 'stick-shaker' or 'stick-push or nudger' type.

c) The warning

The regulatory margin between the stall and the stall warning is 5 knots or 5% of the CAS whichever is the greater. The warning provided can be in the form of tactile, aural or visual or a combination of these signals. Most aircraft have warning provided by stick-shakers which vibrate the control column as well as produce a rattling noise. In fly-by-wire systems the warning consists of a cricket (insect) sound, a synthetic voice STALL message and the red master WARNING light illumination. The stall warning must continue until the angle of attack is reduced to approximately that at which the stall warning is initiated. d)Operation

The stall warning module processes the signals from the various inputs to produce appropriate stall warning output signals. The system has the following inputs:
1) angle of attack 2)flap and slat positions
3) landing gear weight-on position 4)airspeed.

The angle Of attack sensors are usually located on either side of the front fuselage. Sensing relays denote the positions of the flaps and slats. Since the pitch attitude of the aircraft is also changed by the extension of flaps or slats the angle of attack signal has to be modified when these are extended. During take-off when the nosewheel lifts off, microswitches operate to make the stall warning system active. The airspeed is usually derived from the ADC.

The output signals from the system can be applied to : 1)a stick-shaker motor
2)an angle Of attack indicator 3)aural warning
4)synthetic voice warning 5)red master WARNING light

e)Component

swept or delta wings.

ii)Alpha Probes

The two types in current use are the conical slotted probe and the vane detector; the conical slotted probe The vane detector is a counter-balanced aerodynamic vane which positions the rotor of a synchro. Both types are protected against ice formation by a heater.

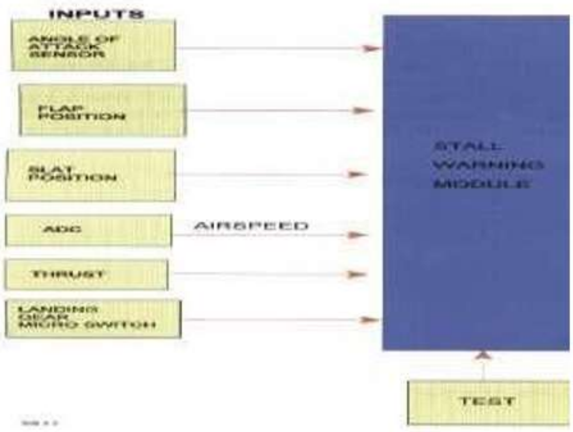


Figure 2.5 Components

i) Angle of Attack Sensing.

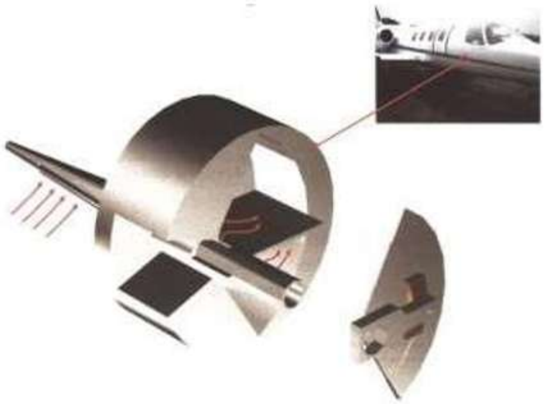
The angle of attack (AOA), or alpha (a) angle, also known as the aerodynamic incidence, is the angle between the chord line of the wing of an aircraft and the direction of the relative airflow, and is a major factor in determining the magnitude of lift generated by a wing. Lift

increases as a increases up to some critical value at which it begins to decrease due to separation of the slow- moving air (the boundary layer) from the upper surface of the wing, which, in turn, results in separation and turbulence of the main airflow. The wing, therefore, assumes a stalled condition, and since it occurs at a particular angle rather than a particular speed, the critical AOA is also referred to as the stalling angle. The angle relates to the design of aero foil section adopted for the wings of any one particular type of aircraft, and so, of course, its value varies accordingly; typically it is between 120 and 180 for straight wings but maybe as high as 300 or 400 for

The conical probe extends through the aircraft skin perpendicular to the flow of air. The probe is attached to a paddle inside the transmitter housing. The probe and paddle are free to rotate. Two sets of slots in the probe allow pressure variations, caused by changes in airstream direction, to be transmitted through separate air passages to opposite sides of a paddle chamber. When the pressure acting on one side of a paddle is greater than the pressure on

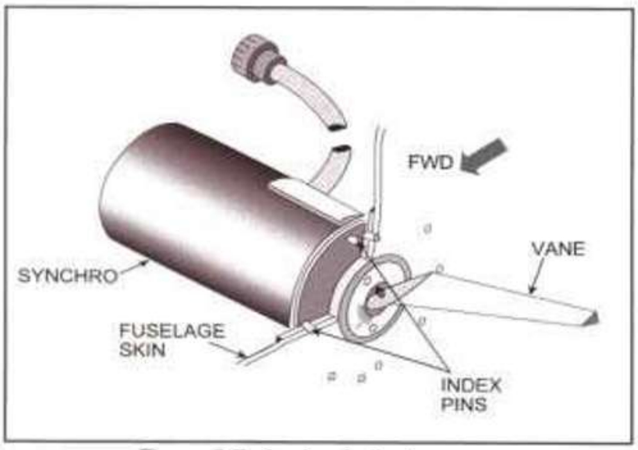
the other side, the paddle and probe rotate until the pressures are equal. The probe thus positions itself to determine the angle of attack of the aircraft. The probe also drives the electrical pick offs such as potentiometers or synchros.

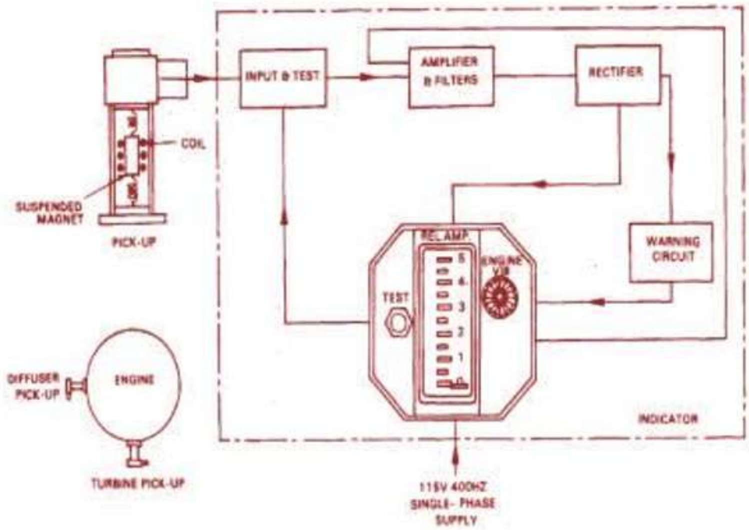
Angle of attack sensor

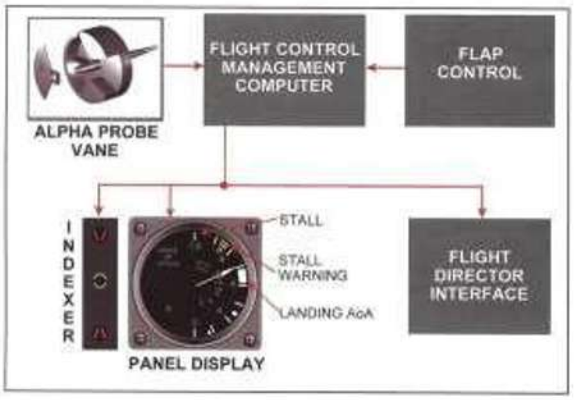


iii) Angle of Attack Indicators.

These may be fitted in addition to the stall warning system. A simple schematic lay-out of the installation is shown in figure







Engine Vibration Monitoring And indicating Systems

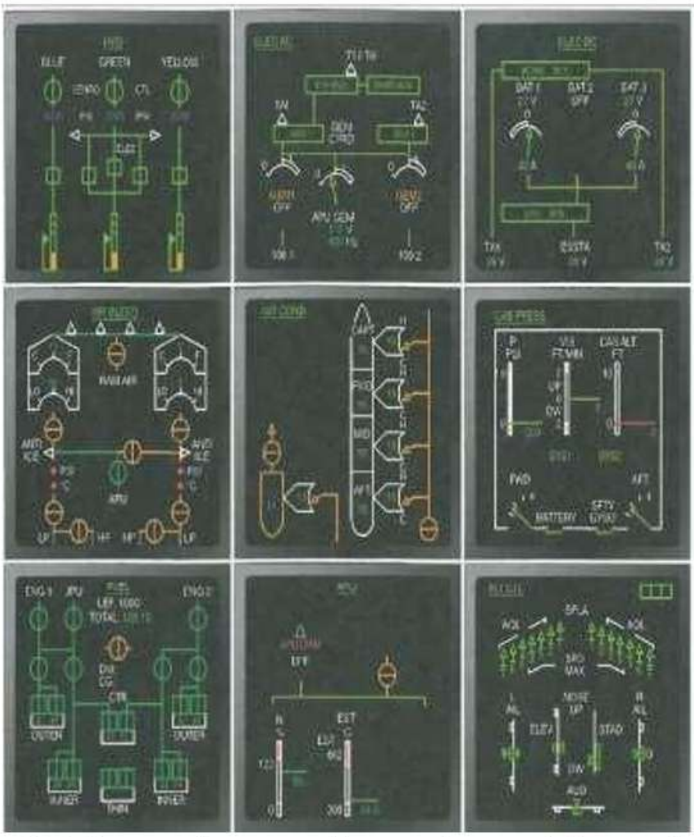
Engine vibration is, of course, something which is unwanted, but unfortunately it cannot be entirely eliminated even with turbine engines,
which have no reciprocating parts. It can only be kept down to the lowest possible level. During operation, however, there is always the possibility of vibration occurring in excess of acceptable levels, as a result of certain mechanical troubles. For example, a turbine
blade may crack or creep' or an uneven temperature distribution around turbine blades and rotor discs may be set up: either of these troubles will give rise to an unbalanced condition of the main rotating assemblies. In order, therefore, to monitor vibration and to indicate when the maximum amplitude on any engine exceeds a
preset level, systems have been developed which come within the engine control group of instrumentation. A system consists essentially of a vibration pick-up unit mounted on the engine at right angles to its axis, an amplifier monitoring unit and a moving-coil micro ammeter calibrated to show vibration amplitude in thousands of an inch (mils).

The pick-up unit is a linear-velocity detector that converts the mechanical energy of vibration into an electrical signal of proportional magnitudes. It does this by means
of a spring-supported permanent magnet suspended in a coil attached to the interior of the case.

As the engine vibrates, the pick-up unit and coil move with it; the magnet, however, tends to remain fixed in space because of inertia. The motion of the coil causes the turns to cut the field of the magnet thus inducing a voltage in the coil and providing a signal to the amplifier unit. The signal, after amplification and integration by an electrical filter network, is fed to the indicator via a rectifying section. An amber indicator light also forms part of the system, together with a test switch. The light is supplied with direct current from the amplifier rectifying section and it comes on when the maximum amplitude of vibration exceeds the preset value. The test switch permits functional checking of the system's electrical circuit. In some engine installations, two pick-up units may be fitted to an engine, one monitoring vibration levels around the turbine section and the other around the diffuser section. In this case, a two-position switch is included in the monitoring system so that the vibration level at each pick-up may be selected as required and read on a common indicator.

GLASS COCKPIT

In an effort to increase the safety of operating complicated aircraft, computers and computer systems have been incorporated. Flight instrumentation and engine and airframe monitoring are areas particularly well suited to gain advantages from the use of computers. They contribute by helping to reduce instrument panel clutter and focusing the pilot's attention only on matters of imminent importance. —Glass Cockpit" is a term that refers to the use of flat-panel display screens in cockpit instrumentation. In reality, it also refers to the use of computer produced images that have replaced individual mechanical gauges. Moreover, computers and computer systems monitor the processes and components of an operating aircraft beyond human ability while relieving the pilot of the stress from having to do so. Computerized electronic flight instrument and maintenance systems have additional benefits. The solid-state nature of the components increases reliability. Also, microprocessors, data buses, and LCDs all save space and weight. Technicians interface with EICAS (Engine Indicating and Crew Alerting System) and ECAM (Electronic Centralized Aircraft Monitoring) Systems through control panels to gather operating and maintenance data. These systems have been developed and utilized on aircraft for a number of years. New systems and computer architecture



development is ongoing. Detains on the operation and use of these glass cockpit maintenance aids are located in the manufacturer's maintenance manual.

13.9 LIGHTS (ATA 33)

LIGHTING SYSTEM INTRODUCTION

Lighting is installed on aircraft for a number of reasons including: safety, operational needs, servicing and for the convenience of passengers. The applications of aircraft lights can be broadly grouped into four areas: flight compartment (cockpit), passenger cabin, exterior and servicing (cargo and equipment bays).

Lighting technologies

Aircraft lighting is based on a number of technologies:

- Incandescence
- Light-emitting diode (LED)
- Electro-luminescent
- Fluorescence
- Strobe.

Incandescence is the radiation of light from an electrical filament due to an increase in its temperature. The filament is a small length of wire, e.g. tungsten, which resists the flow of electrons when a voltage is applied, thereby heating the filament.

Electro-luminescence is a combined optical and electrical phenomenon that causes visible light to be emitted. This can be achieved with electron flow through a semi-conductor material, or by a strong electric field applied across a phosphor material. Electro-luminescence is the effect of recombining electrons and holes in a light-emitting diode (LED) or phosphor material.

Fluorescent lamps are gas-discharge devices formed from a sealed tube of glass that is coated on the inside with phosphor; the glass

tube contains mercury vapor mixed with an inert gas,
e.g. argon or neon. The lamp uses a
high voltage to energize the mercury vapor; these results in an ionized gas where the electrons are separated from the nucleus of their atoms creating plasma. The release of energy causes the phosphor coating to fluoresce, i.e. it produces visible light. Fluorescent lamps require a ballast resistor to regulate the flow of energy in the tube.

Strobe lights are formed from small diameter (typically 5 mm) sealed quartz or glass envelope/tube filled with xenon gas. Power from the aircraft bus is converted into a 400 V DC supply for the strobe. The tube is formed into the desired shape to suit the installation is a wing-tip anti-collision light, normally located behind a clear plastic protective cover. Xenon is an inert (or noble) gas, chemically very stable, and has widespread use used in light- emitting devices, e.g. aircraft anti-collision lights. The emission of light is initiated by ionizing the xenon gas mixture by applying a high voltage across the electrodes.

Flight compartment lights

Lighting is needed for the illumination of instruments, switches and panels.

Dome lights located on the ceiling provide non-directional distribution of light in the compartment; it typically contains an incandescent lamp and is powered from the battery or ground services bus.

Flood lighting in the flight compartment from incandescent lamps and/or fluorescent tubes provides a general illumination of instruments, panels, pedestals etc. Fluorescent tubes located beneath the glare shield provide overall illumination of the instrument panels.

Emergency lights are installed in the flight compartment for escape purposes. The colour of flight compartment lights is normally white; this reduces the power and heat, improves contrast on the instruments, and reduces eye fatigue.

Instruments

Is normally Internal instrument lighting from incandescent lamps integrated within individual instruments; lighting must be shielded from causing any direct glare to the pilot and must be dimmable. External instrument lighting is provided by pillar (or bridge) lights positioned on the panels for individual instruments. The light intensity can be dimmed by a simple rheostat device as illustrated in Fig above; this typical circuit is for flight instruments, engine instruments and switch panels.

Transistor lighting control circuit

A transistor circuit provides electronic control as illustrated in

Fig above; variable resistor R V1 varies the (relatively low) base current into the base of a PNP transistor; this controls the (relatively high) current through the collector and lamp. The relatively low base currents in the respective transistors can now control a variety of lighting circuits:

- Radio navigation systems
- Compass
- Fuel panels
- Engine indications

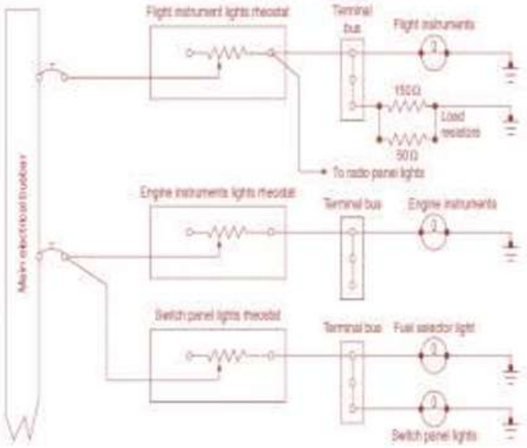
Master warning

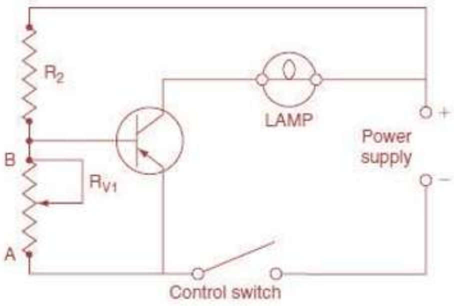
An increasing number of systems are being designed into aircraft; this leads to more warning lights and larger panels with an increased possibility of a warning light being missed by the crew. This has led to centralized _attention getters', or master warning and caution light panels.

Warning and caution lights affecting system operation and aircraft safety are defined by specific colors:

- Warning, red, an unsafe condition exists
- Caution, amber, an abnormal condition exists, but it is not unsafe.
- Advisory, green or blue, a safe condition exists, or for information e.g. gear down.

Passenger cabin lights Interior lighting installations for the passenger cabin vary depending on the size of aircraft; this ranges from a small quantity of roof- mounted incandescent lamps, through to integrate lighting concealed within the interior trim. These

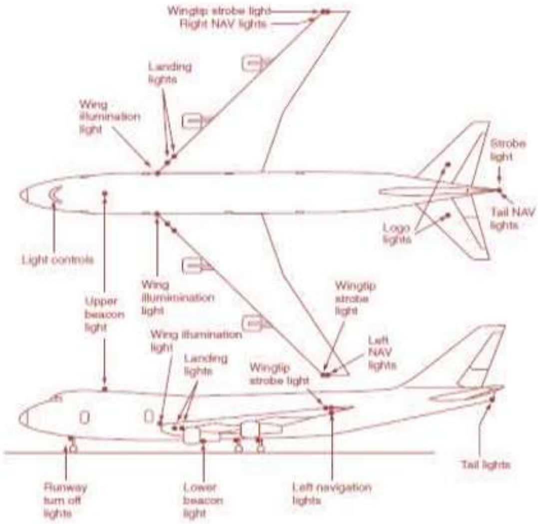




lights are controlled from the flight attendants' station. LED illumination is being specified on business and passenger aircraft that have pre-programmed settings for specific flight phases and time zones. The systems are automatically controlled to customize the mixing of colors and lighting levels; this is intended to help passengers combat the fatigue of long-distance travel.

Exit lights are located adjacent to the emergency exits and are clearly visible, irrespective of whether the door is open or closed.

Floor path lighting is used to in emergency situations to provide visual identification of escape routes along cabin aisle floor. These systems have sufficient energy to enable passengers to identify aisle boundaries. The system guides the passengers to designated emergency exits.



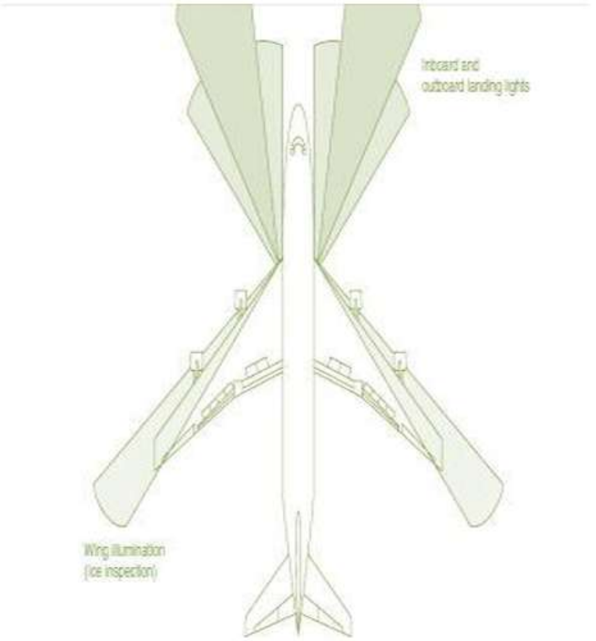
Exterior lights

An overview of the exterior lighting arrangement on a large passenger aircraft is illustrated in Fig above.

Exterior lighting is used for:

- Logo illumination
- landing/taxiing
- Wing illumination
- anti-collision/navigation.

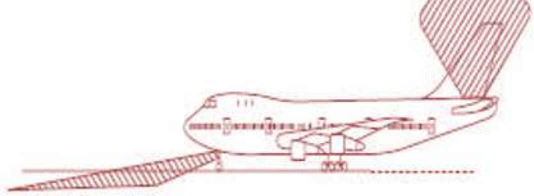
Logo lights



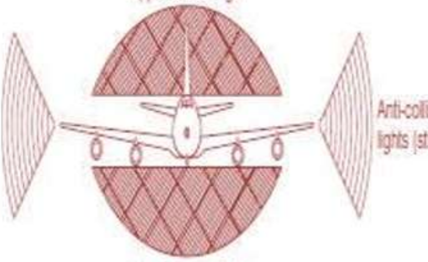
Logo lights are used to illuminate the tail fin; this is primarily for promotional purposes, i.e. for the airline to highlight their logo during night operations at an airport. Apart from the advertising value at airports, they are often used for additional awareness in busy airspace.

Taxi lights (or runway turn off lights) are sealed beam devices with250 W filament lamps located on the nose, landing gear or wing roots. They are sometimes combined with the landing light and used when approaching
or leaving the runway. Taxi lights improve visibility during ground operations; they are directed at higher angle than landing lights.

Landing lights are located on the wing tips, or on the



front of the fuselage, usually at fixed angles to illuminate the runway. They are sealed beam devices with 600–1000 W filament lamps; a parabolic reflector concentrates light into a directional beam. The high current requirement is controlled via a relay. Some landing light installations have a retractable assembly located on the underside of the wing. This has a reversible motor and gear mechanism to drive the light out against the airflow. The alternative location for a landing light is in the wing leading edge; this has a transparent cover to provide aerodynamic fairing. Inboard and outboard landing lights provide extended illumination of the landing area.



Wing illumination
Ice inspection lights are often installed to check ice formation on wing leading edges and engine intakes. Typical lights are the sealed beam type with filament lamps of 50–250 watts. They are recessed into the fuselage or engine nacelle with a preset direction that illuminates a section of the wing that can be viewed from the flight compartment.

Service lights
Service lights are provided throughout the aircraft. These lights are powered from the aircraft ground servicing bus. Examples include:

- Cargo bays
- Wheel wells
- Equipment bays
- fuelling panels

Navigation lights

The navigation (or position) lights are a legal requirement for night flying. Navigation lights are normally based on filament lamps, providing steady illumination. They are located at the extremes of the aircraft and provide an indication of the aircraft's

direction and maneuvers. Navigation lights are based on regulations that define the colour, location and beam divergence such that the aircraft is visible from any viewing angle; these colors and divergence angles are:

- Green, starboard wing, divergence of 110 degrees
- Red, port wing, divergence of 110 degrees
- Clear (white), tail, divergence of 70 degrees either side of aircraft centerline (140 degrees total).

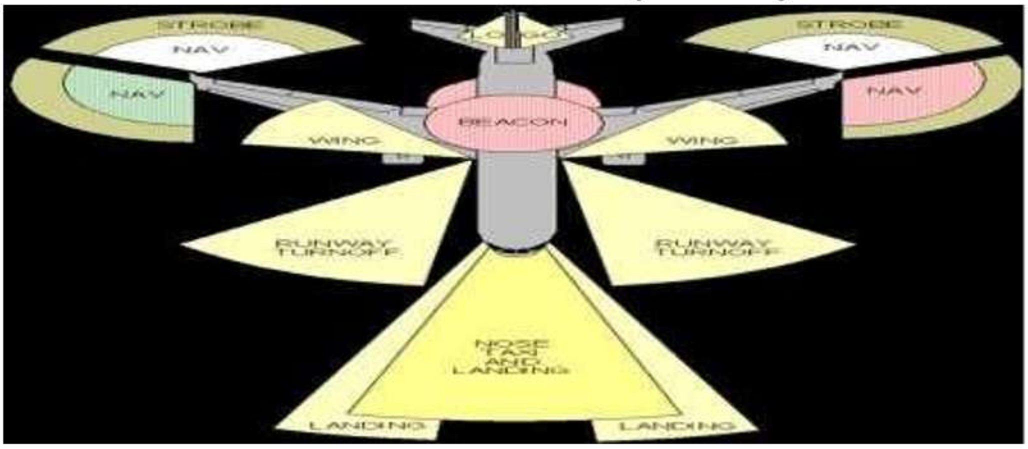
Anti-collision lights
Anti-collision lights often supplement navigation lights; these can be provided either by a strobe light, rotating beacon or a combination of both, . Anti-collision lights are also used as a warning that the engines are running or are about
to be started. They are typically not switched off until it is considered safe for ground personnel to approach the aircraft. Strobe lights are typically located on the:

- vertical stabilizer
- wing tips
- tail/lower wing surfaces
- fuselage.

These anti-collision lights are controlled by a single switch, with a single protection device. Anti-collision lights used in conjunction with the navigation lights enhance situational awareness for pilots in nearby aircraft, especially during night-time or in low-visibility conditions.

The rotating beacon comprises a filament lamp, reflector, motor and drives mechanism that gives the effect of a light through a red filter those flashes 40–50 times per second. They are located on tail fins and the upper and lower fuselage (or tail boom on a helicopter).

Strobe lights are wing-tip and tail-fin mounted to supplement navigation lights. The strobe light produces a high intensity white flash of 1 mS duration at approximately 70 fl ashes per minute through a white or red filter; these provide light that can be seen from several miles. Many external lights are based on sealed beams. A sealed beam combines an incandescent filament lamp and reflector into a single assembly. The reflector concentrates the light from the lamp into a predetermined beam shape; the assembly is fittedwith a clear glass front cover that is permanentlysealed to the reflector and cannot be removed. Thefilament lamp is inserted through a hole in the rear ofthe reflector and retained by a locking mechanism.



13.10 On Board Maintenance Systems (ATA 45)

Central Maintenance System Introduction

Acquisition

The acquisition of aircraft system data is performed by 4 major electronic systems

the Electronic Centralized Aircraft Monitoring (ECAM) system which monitors the operational data in order to display warnings and system information.

The Flight Data Recording System (FDRS) which is mandatory and records aircraft operational parameters for incident investigation purpose.

The Central Maintenance System (CMS) which monitors the BITE data in order to record the system failures,

The Aircraft Condition Monitoring System (CMS) which records significant operational parameters in order to monitor the engine & the aircraft performance and to analyse specific aircraft problems.

Consolidation

In normal operation. the ECAM permanently displays normal aircraft parameters and Me ACMS and FDRS permanently record aircraft system parameters.

When an anomaly is detected by an aircraft system, the ECAM displays the abnormal parameter or function and its associated warning and the CMS records the failure information detected by the system BITE

Retrieval

All the information can be retrieved through.

The cockpit multipurpo se control display unit,

The ECAM displays.

The cockpit printer.

The down loading system.

A ground station via acars.

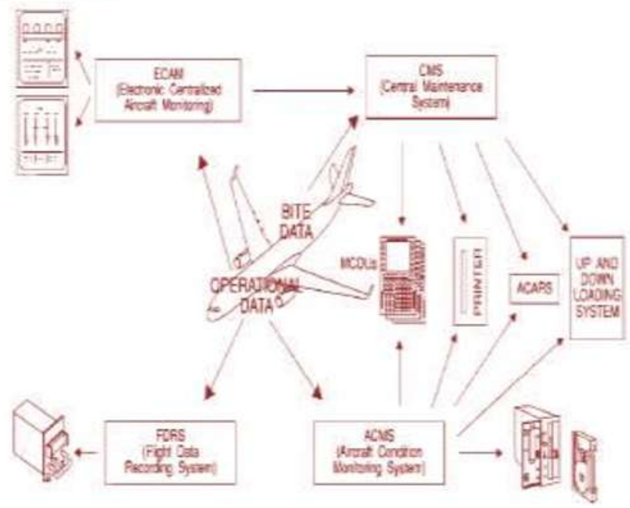
And the recorders

Analysis

Maintenance operations can be divided into 3 groups:

- minor trouble shooting which is performed with the help of the ECAM and the CMS through the MCDUs and the printed or ACARS down- linked reports.
- In depth trouble shooting which is performed with the help of the CMS and the ACMS through the MCOUs and printed reports.
- long term maintenance which is performed with the help of the ACMS and the FORS through printed. ACARS down. linked and down- loaded reports or recorded tapes

Figure 1: On Board Maintenance Facilities



BITE Philosophy General

A system is composed of LRUS which can be computers, sensors actuators. probes etc. Most of these Line Replacable Units (LRUs) are controlled by digital computers For safety reasons these LRUs are permanently monitored. They can be tested and troubleshooting can be performed In each system a part of a computer is dedicated to these functions it is called Built In Test Equipment In some multi- computer systems. One computer is used to concentrate the BITE (Built- In Test Equipment) data of the system

BITE

During normal operation, the system is permanently monitored internal monitoring. Inputs/outputs monitoring, link monitoring between LRUs within the system

FAULT DETECTION

If a failure occurs. it can be permanent (consolidated) or intermittent ISOLATION

After failure detection. the BITE is able to identify the possible failed LRUs and can give a snapshot of the system environment when the failure occurred

MEMORIZATION

All the information necessary for maintenance and trouble shooting is memorized in a Non Volatile Memory.

Concept

The BITE information stored in the system BITE memories is sent to a centralized maintenance device. The manual tests (SYSTEM TEST and SPECIFIC TESTS) can be initiated via this centralized maintenance device.

Its main advantages SW

single interface location (cockpit). easy fault identification reduction of the trouble shooting duration simplification of the technical documentation. standardization of the equipment.

Test

The test function can be divided into 4 groups POWER UP TEST

The power up test is first a safety test. The purpose of a safety test is to ensure compliance with the safety objectives It is executed only on ground after long power cuts (more than 200ms) Its duration is function of the system which is not operational during the power up test

If the aircraft is airborne. the power up test is limited to a few items to enable a quick return to operation of the system The typical tasks of a power up test are test of microprocessor. test of memories, test of ARINC 429 and various I/O or-cults.

configuration test

CYCLIC TESTS

These tests are carried out permanently They do not disturb system operation

The typical tasks of a cyclic test (also called IN OPERATION TEST) are Watchdog test (a watchdog is a device capable of restarting the microprocessor if the software fails). RAM test, Permanent monitoring is performed by the operational program (e g. ARINC 429 messages validity)

SYSTEM TEST

The purpose of this test is to otter to the maintenance staff the possibility to test the system for trouble shooting purposes This test can be performed after the replacement of a LRU in order to check the integrity of the system or sub-system. It is similar to the POWER UP TEST but it is more complete II is performed with all peripherals supplied

SPECIFIC TESTS

For some systems specific tests are available The purpose of these tests is to generate stimuli to various command devices such as actuators or valves
They can have a major effect on the aircraft (automatic moving of slats or flaps. engine dry cranking).

Architecture General

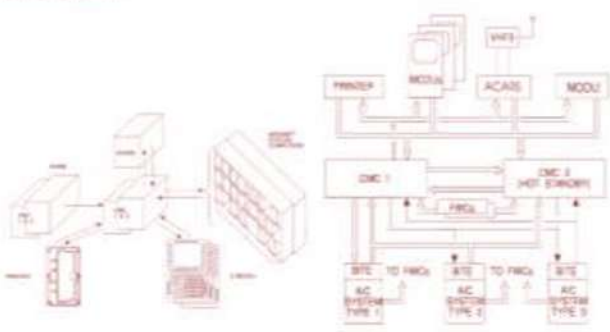
The Central Maintenance System or also called Centralized Fault Display System is composed of one or two Maintenance Computers and the aircraft system BITES. Build In Test circuits are build in most system LRU's

The CMC interfaces are M C D U S a printer the Aircraft Communication Addressing and Reporting System (ACARS). a data loader called Multifunction Disk Drive Unit (MDDU). Central Maintenance Computer (CMC) or Centralized Fault Display Interface Unit (CFDIU) The Central Maintenance Computers continuously scan the buses coming from the NC systems If a failure message from a system BITE is present on a bus. the CMC or CFDIU copy and store it They also store the Electronic Centralized Aircraft Monitoring (ECAM) messages generated by the Flight Warning Computers.
BITE

In each aircraft system computer, a BITE monitors the system and memorizes the failures. The A/ C systems are divided into three types. depending on their capabilities and their connection to the CMCs
Multipurpose Control and Display Unit MCDU

The Multipurpose Control and Display Unit is the operators interface with the Central Maintenance System. Any two of the three MCDUs may be operated simultaneously
Printer
Most of the Central Maintenance System reports may be printed. The printer provides the Post Flight Report (PFR) print which is the main maintenance tool.

Aircraft Communication Addressing and Reporting System ACARS
Data may also be transmitted to the ground through the Aircraft Communication Addressing and Reporting System (AGARS)
Multifunction Disk Drive Unit MDDU
Data may also be loaded into the CMCs through the Multifunction Disk Drive Unit



Condition Monitoring General
I he main functions of the Aircraft Condition Monitoring System (ALMS) are to perform engine condition. APU condition and aircraft performance monitoring as well as trouble shooting assistance It collects records and processes aircraft system data which can be retrieved through the MCOU, on a memory support or printed.

Architecture
The AC MS consists of.

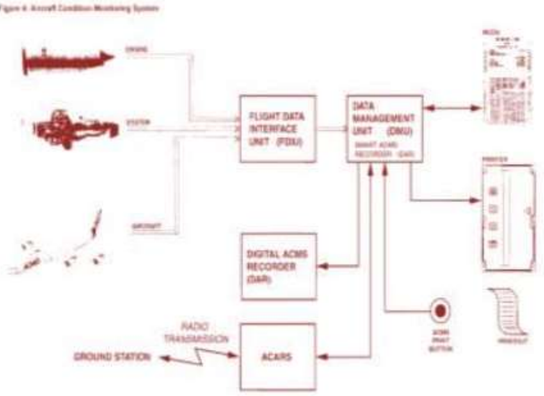
- The Dale Management Unit (DMU) including a Smart ACMS Recorder (SAR). The DMU may contain a Personal Computer Memory Card International Association (PCMCIA) interface.
- Flight Data Interface Unit (FDIU),
- an "on-ground' equipment called Ground Support Equipment (GSE).
- an optional Digital ACMS Recorder (OAR).

DMU
The DMU collects, stores and processes various aircraft system data. This data can be stored in the internal DMU memory the PCMCIA card or the DAR. if installed The collected data is used to generate various condition reports. These reports can be stored in the internal DMU memory or on the PCMCIA card
FDIU
The Flight Data Interface Unit (FDIU) part of the Flight Data Recording System. It Sends the same parameters as the Digital Flight Data Recorder (DFDR) to the DMU. These parameters will be recorded on the PCMCIA card.
Aircraft Systems
Various aircraft systems are connected to the FDIU. These input sources provide the FDIU with engine parameters, APU parameters and aircraft parameters
SAR
The SAR is a DMU function. This function allows the recording of compressed data programmable through the GSE. SAR data can be stored in the internal DMU memory or on the PCMCIA card
DAR
The purpose of the DAR is to store data on an optical disk for on ground performance. maintenance or condition monitoring tasks Preprogrammed selection of data can be done through the GSE.
DAR data can also be stored on the PCMCIA card
MCDU
Multipurpose Control and Display Units (MCDU) are connected to the DMU. to display data, program and also control the system.
Compared to the GSE the programming facilities offered by the MCOU are very limited.
The main functions of the MCDU within the ACMS are Online display of selected parameters.
display of the list of the stored reports and SAR files. manual request of reports and SAR/DAR recording
Printer
The printer is used to print reports generated by the DMU as well as most of the ACMS MCDU displays The printer can be automatically controlled by the DMU manually controlled from the MCDU or activated using the ACMS PRINT pushbutton
ACARS
The Aircraft Communication Addressing and Reporting System can be used to send reports and to broadcast parameters generated by the ACMS to a ground station via radio transmission

VHF Voice Data Radio.
Satellite Communication System or
- Gate-Link acting as a Wireless Local Area Network (LAN) when aircraft is ground

The download of reports can be automatically initiated by the DMU or manually initialed from the MCDU. The ATSU can also
receive and send to the FDIMU requests from the ground 00
Ground Support
Equipment GSE
The GSE is based on a compatible personal computer able to read

3.5 inch floppy disks and PCMCIA disks. The GSE software provides the following main functions: Reconfiguration function and Readout function. The reconfiguration function allows the configuration of the customer database (trigger conditions. layout of recording space). The readout function allows display, print out and analysis of recorded data.



Data Loader

The data loader is used to upload data into the ACMS (operational software. customer databases), download data on a 3.5 inch floppy disk for GSE analysis (reports, SAR data).

PCMCIA Interface

The PCMCIA interface accepts high capacity and removable PCMCIA disks. On the PCMCIA disk can be stored ALMS reports, SAR data, DFDR data and DAR data. The disk space ratio is programmable by the GSE. The PCMCIA interface can also be used as a portable data loader to upload ACMS software and data-bases or to download recorded data.

Structure Monitoring

Structure monitoring comprises the instrumentation of a structure with a sensor system that monitors how the surroundings load the structure. An advanced soft- ware system receives thedata, interprets it and reports the structure's condition to the operator. Real-time feedback to the operator means that overloading may be avoided and any damage to the structure may be discovered quickly. Comprehen- sive knowledge of the objective condition of the structure allows condition-based maintenance, black-box recorder functionality and investigation of incidents.

The technology is well suited for applications such as condition monitoring of oil platforms, wind turbines and bridges.

In civil aviation, such systems come into action only during development and certification of an aircraft.

Damage Tolerance Monitoring

The term "Damage Tolerance Monitoring" describes short and correct what it exactly does.

A good example is the permanent monitoring of the vertical acceleration of an aircraft during landing. A hard landing can make serious damage to the structure and must be avoided in any case.

If happened, the computer registers the excessive acceleration during touchdown. This is reported e.g. to the pilot or maintenance personnel and the necessary action can be carried out.

Data Loading General

The data loading system is an interface between Me aircraft computers and ground data processing equipment used to update software and data bases or to retrieve aircraft system data The MDDU can operate in two mottles automatic mode and manual mode

The manual mode is only used to down load data while the automatic mode is used to up and down load data According to the operation to be performed. the disk which is used has to contain speak information (e. 9 configuration fie)

The data loading system includes two rotary selectors for system selection. It also includes a Multi- purpose

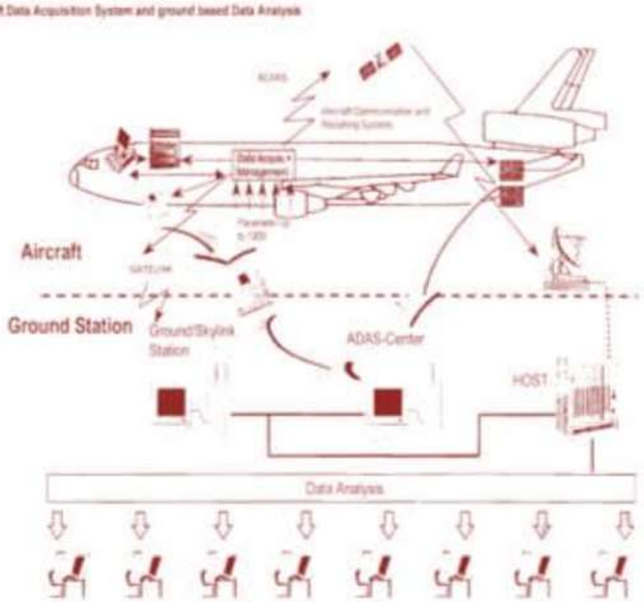
Disk Drive Unit (MDDU)

If the Multi- purpose Disk Drive Unit is not installed. Up and Down Loading functions can be performed through a connector by using a portable data loader (see module 5)

Before performing an up data loading operation, refer to the relevant procedure for the corresponding system in the Aircraft Maintenance Manual.

Up Loading

The aircraft system computers use the loading system to update their data base (for example the FMGEC Flight Management Guidance and Envelope Computer) or to modify pans of their operational software (for example the ACARS Management Unit}



Down Loading

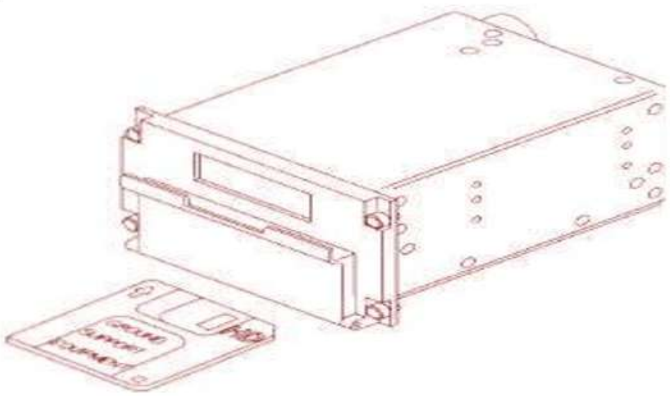
The down leading system is used to down load to a 3 5 inch disk, the data recorded by certain compilers during aircraft operation (for example the Artier% Condition Mon toeing System)

Components

The MDDU contains

- an electronic Jot composed of a pose supply. Input: output and CPU !Fleecy Disk Drive Cover boards
- a Disk Drive installed on shock mounts.
- a window with 16- character alphanumerical LCD display.
- a door protecting access to the Disk Drive

7-e data support is a
3 5 inch double face. high density
disk (1 44 megabytes) disk K vi M$- DOS format It can be read or written on ground by IBM- PC Ground Support Equipment (GSE)



Abnormal Operation

Other messages displayed on the MDDU window inform the operator of the transfer status
TRANSFER FAILURE: If the MDDU has to stop data transfer (up or down;owing) for any reason. This message is displayed
UNIT FAIL: The MDDU displays this message if a hardware failure is detected during the self- test. In this case the MDDU stops all operations.
DISK ERROR: If the MDDU cannot read or write data disk (incorrect formatting write- protected disk damaged etc) will interrupt operations and display this message

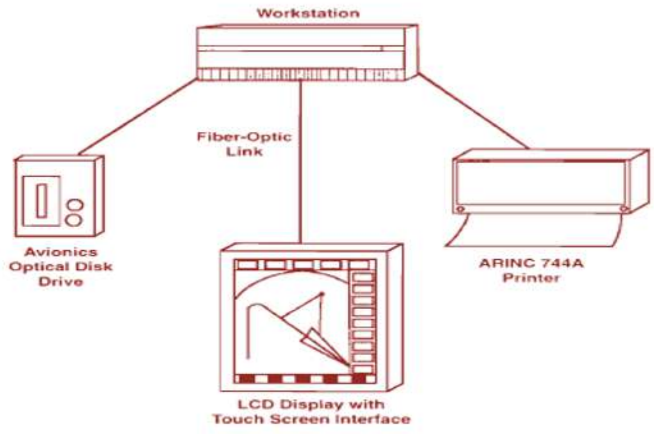
Electronic library

General
The Electronic Library System (ELS) is an information management system designed to provide airline personnel with timely access to the information necessary to operate and maintain an aircraft The system provides airline flight, maintenance, and cabin crews with instantaneous access to information contained in tens of thousands of pages of operational manuals, procedures, and navigation Charts. Wherever possible, the information is provided in a task-oriented manner so that it can be used more efficiently and accurately. The long-term objective of the ELS is to eliminate the airline's need to carry and update paper documentation on board the aeroplane
In addition to timely access to information, flight crews will also have enhanced avionics functionality due to ELS mass storage capabilities. Maintenance crews will benefit because the Onboard Maintenance Systems (OMS) will be combined with the Electronic Library System in an integrated maintenance and fault-reporting en- vironment. In addition to the benefits of cabin crew access to operational and procedural information, the mass storage and file server function contained within the ELS allows new levels of passenger service capabilities.
The airborne Electronic Library System is only a portion of the overall ELS solution Extensive ground applications using the information to be contained in the airborne system are already in place at most airlines today. The challenge for the industry is to provide the airlines with an airborne system that complements exist- ing and future ground-based systems. The goal is to have a common source database for all facets of airline operations (dispatch, night operations, maintenance, inventory/parts control, etc)
Information updates to the airborne system and communications between the ground-based and airborne systems are accomplished through the emerging Gate-Ink concept and other

existing and planned communication channels. In the long term, the very-high band-width Gate-link interface will allow the aircraft to park at the gate and log in as a node on the airline's computer network Bidirectional Gate-link information transfers between the aircraft and airline ground-based operations centers will ensure accurate and up-to-date information on the aircraft while significantly simplifying the update process.
System
The evaluation system consisted of a monochromatic active matrix liquid crystal display (LCD), an avionics-quality optical disk drive, a prototype ARINC 744A minter, and a workstation platform. The user input device to the liquid crystal display consisted of a capacitive touch screen overlay on the display surface. Fight crews accessed desired information by simply touching the display screen in the appropriate areas. The system did not include a keyboard for user input. Direct text entry for airport directory or word searches was implemented with a 'soft key- board function displayed on the screen. Integrating the input device directly with the display also helped conserve the space necessary for installing the display on the flight deck. This is particularly important for fleet-wide ELS Installations that may include
flight decks with limited space.

The integrated touch screen input device and a fibre optic

Figure 11: Electronic Library System



interface to the display simplifies the wiring installation. Software provides an accurate and complete functional representati on of the en- visioned final product. The system included simple "page turning" functions as well as more advanced retrieval techniques such as text and graphical hyperlinking Great care was taken in the design of the graphical user interface (GUI) to ensure intuitive and consistent operation.

A subset of all manuals and navigation charts carried in flight crew kit bags are stored digitally on the optical disk. Digital data was delivered and stored in multiple formats. Including compressed bitmap. Text, and vector graphic formats. This data set included sections from the Flight Crew Operating Manual. Aircraft Flight Manual, Airport Analysis Manual, Aircraft Maintenance Manual, Aircraft Schematic Manual. Illustrated Parts Catalogue and Minimum Equipment List (MEL). Digitized versions of airport terminal chards were provided by Jeppesen. Inc.

The electronic system would not only eliminate the bulky weight carried on the aircraft (up to 90 kilograms carried by international flight crews) but. More importantly. Significantly reduce the cost of producing, updating. Distributing. And maintaining the existing paper information. Furthermore, users recognized the po-tential benefit from task-oriented organization of information and that an ELS would allow this potential to be realized.

Airborne Printer General

The porter comprises the following functional sub- assemblies:

A front panel with pushbuttons and indicators

- an electronic part consisting of a Central Processing tint printer controller and power supply boards.
- • mechanical bock

The paler awes to printout of the Central Maintenance System reports and additions printouts from

the Aircraft Condition Monitoring System (ACAS. AIDS or ACMSE

the Aircraft Communication Addressing and Reporting System (AGARS)

the Flight Management. Guidance and Envelope System (FMGESI the Engine Interface Vibration Monitoring Unit EVIMU or EVMUI.

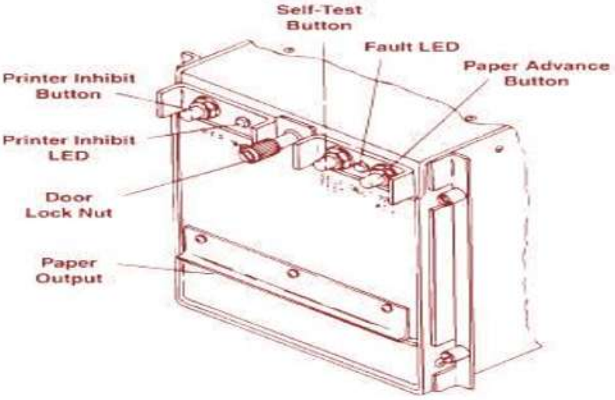
Paper

The printer works as a non- impact thermal printer vend, oily works on special neat sensitive paper. The paper can be inserted via an access door incorporated n the front panel. The printer is loaded with a correct size of paper roll (108 or 216 mm)

Controls

The printer face features pushbutton Switches and annunciator bunts. i.e.
TEST switch is used to perform a functional test SLEW switch is used to exit paper.
'PAPER ALARM'' pushbutton switch includes an AMBER caution light
'OFF' pushbutton switch includes a status Indicator light
Printer
This model shown below is often used as a versatile cockpit printer. Different controls located on the front panel:
PRINTER INHIBIT button and light Turning printer on/off. DOOR LOCK to open the printer for paper replacement.
SELF TEST button to start a self test and to print a test pattern. FAULT LED illuminates with internal failure or overheat
PAPER ADVANCE or SLEW button advances paper without printing.

Figure 13: Front face



Paper Loading

A red line marked on the paper indicates that the supply roll must be replaced. Open the printer and insert a new roll Verify that the heat sensitive side is facing against the print head. Close the printer and initiate a test print.

13.11 Air Conditioning and Cabin Pressurisation (ATA21)

Air conditioning

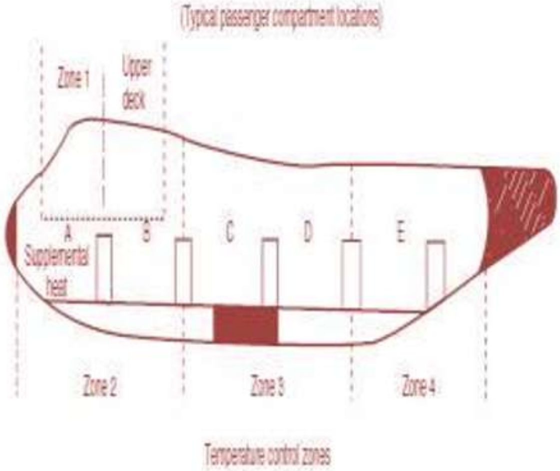
Air conditioning is provided in passenger aircraft for the comfort of passengers. The cabin on larger aircraft is divided into passenger compartment locations, or temperature control zones, Air conditioning, and the environmental control system (ECS), normally derive high-pressure air from the compressor stage of each turbine engine. The temperature and pressure of this bleed air varies, depending upon rotational speed of the engine. A pressure- regulating shutoff valve (PRSOV) restricts the flow as necessary to maintain the desired pressure for the ECS.

Environmental control system

To increase efficiency of the ECS, air is normally bled from two or three positions on the engine. Aircraft types vary, but the principles of air conditioning systems comprise five salient features:

- Air supply
- Heating
- Cooling
- Temperature control
- Distribution.

The air conditioning system is based on an air cycle machine (ACM) cooling device as illustrated in fig below.



Air conditioning passenger cabin zones

The air conditioning packs (A/C packs)

are located in varying places on aircraft, including:

- Between the two wings in the lower fuselage
- In the rear fuselage tail section
- In the front of the aircraft beneath the flight deck

Engine bleed air, with temperatures in the order of 150 and 200 ° C and a pressure of between 30–35 pounds per square inch (psi), is directed into a primary heat exchanger. External ram air (at ambient temperature and pressure) is the cooling medium for this air-to air heat exchanger. The cooled bleed air then enters the centrifugal compressor of the air cycle machine (ACM).

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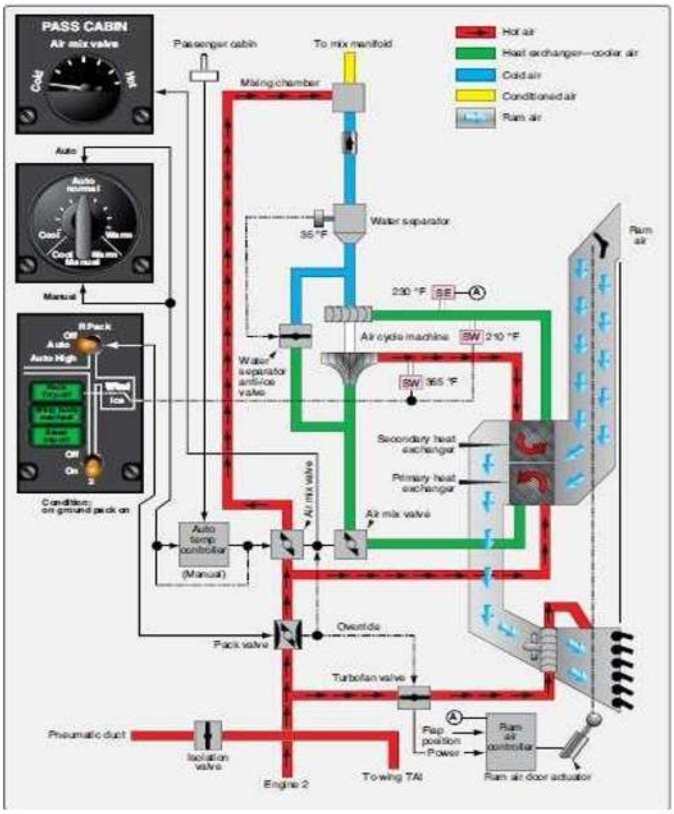
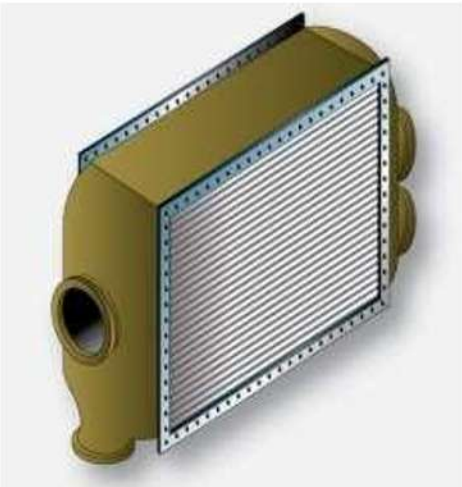
The cooled bleed air then enters the centrifugal compressor of the air cycle



machine (ACM). This compression heats the air (the maximum air temperature at this point is about 250°C) and it is directed into the secondary heat exchanger, which again uses ram air as the coolant. Pre-cooling through the primary heat exchanger increases the efficiency of the ACM by reducing the temperature of the air entering the compressor; less work is required to compress a given air mass (the energy required to compress a gas by a given ratio increases with higher temperature of the incoming air).
At this stage, the temperature of the heat exchanger air is higher than the outside air temperature; the temperature is sensed by an RTD and this is displayed on the air conditioning control panel and/or used as part of a control system input. The compressed, cooled air is then directed into the expansion turbine of the ACM; this extracts work from the air as it expands, cooling it to between
- 20°C and - 30°C. (The ACM can cool the air to less than 0°C even when the aircraft is on the ground, in high ambient temperatures.) The work extracted by the turbine drives a shaft to turn the ACM's centrifugal compressor, together with a ram air inlet fan that draws in the external air during ground running. A motorized bypass valve controls the ratio of air being directed into the turbine. An electrically driven ram air fan within the system provides air flow across the heat exchangers when the aircraft is on the ground. To assist ram air recovery, some aircraft use modulating vanes on the ram air exhaust. Power for the air conditioning pack is obtained by reducing the pressure of the incoming bleed air relative to the cooled air output of the system.

With the air now cooled, its water vapor condenses. To remove this, the moist air output of the expansion turbine is passed through a water separator; this uses centrifugal force to eject the water particles into a coalescing bag that absorbs the moisture. This condensate is sometimes fed back into the ram air entering the secondary heat exchanger to improve its performance. The cool dry air is now combined in a mixing chamber with a small amount of engine bleed air; the amount of trimming air mixed with cooled air is modulated to achieve the desired cabin air temperature before the air is ready for supply into the cabin. An RTD is installed to monitor the ACM outlet temperature; airflow into the cabin is monitored by a flow sensor.



Heat exchanger
Ram
air duct

The A/C pack outlet air for use in the cabin is mixed with filtered air from re- circulation fans, and then fed into the mixing manifold. On modern passenger aircraft, the airflow into the cabin is approximately 50% bleed air and 50% filtered air.

Vapor-Cycle Cooling System

To better understand the way heat is moved in a vapor-cycle cooling system, consider the events that take place when heat from the sun is absorbed in the water of a lake. When the sun shines on a lake during a hot summer day, some of the heat is absorbed by the water, which gets warmer. The warmed water on the surface evaporates, or changes from a liquid into a gas. When the water evaporates, it takes some of the heat from the air immediately adjacent to the surface, and this air is cooled. The water that evaporated from the surface of the lake is still water, only now it is in the form of invisible water vapor that is only slightly more than half as heavy as the air surrounding it. This water vapor still contains the energy from the sun that changed it from a liquid into a gas. The lightweight water vapor rises in the air, and because the temperature of the air drops as altitude increases, the water vapor cools. Soon its temperature becomes so low that it can no longer remain a vapor, and it changes back into a liquid, into tiny droplets that form clouds. When the water vapor reverts into liquid water, the heat it absorbed from the sun is released, and this heat raises the temperature of the air surrounding the cloud. Heat is moved in a vapor-cycle air cooling system in the same way it is moved from the surface of the lake to the air surrounding the clouds. Under standard conditions, water is a liquid. If heat energy is added to a pan of water on a hot stove, and the temperature of the water goes up until it reaches 212°F, then the water boils: As long as the water is allowed to boil, its temperature will never rise above 2l2°F. But, if a tight-fitting lid is placed on the pan and more heat is added to the water, its temperature will go higher.

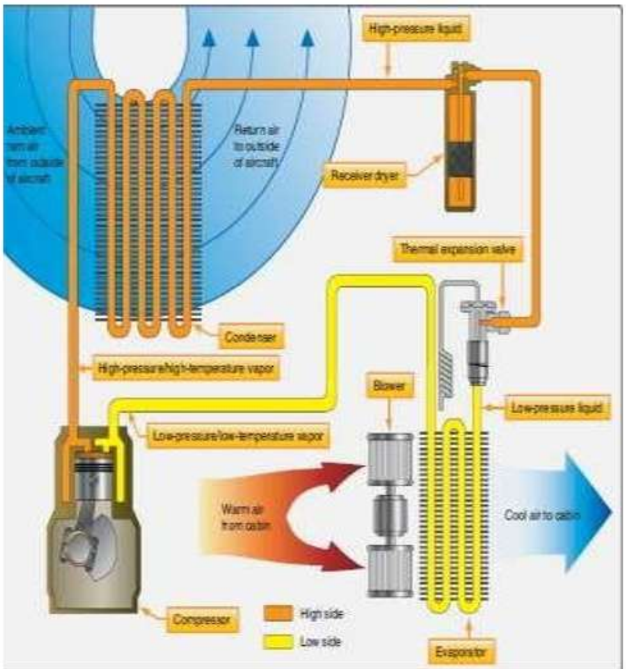
The Compressor

The compressor is the heart of an air c0nditioning system. It moves the refrigerant through the system, and it divides the system into its high side and low side. The compressor pulls the low-pressure

refrigerant vapor from the evaporator and compresses it. And when the vapor is compressed, its pressure and temperature both go up. The compressor carries a specified amount of special moisture-free refrigeration oil that lubricates and seals the compressor, and circulates through the system with the refrigerant.

The Condenser

The refrigerant leaves the compressor as a hot, high-pressure gas, and flows to the condenser mounted where outside air can pass through its fins.The condenser is made of high-pressure tubing wound back and forth, with thin sheet metal fins pressed over the tubes. The hot refrigerant gas enters one side of the condenser and gives up some of its heat to the air flowing through the condenser fins. When the system is working properly, about two thirds of the condenser is filled with refrigerant gas, and the rest contains liquid refrigerant.



Thermostatic Expansion Valves

The thermostatic expansion valve (TEV) is a metering device that measures the temperature of the discharge end of the evaporator to allow the correct amount of refrigerant to flow into the evaporator. All of the liquid refrigerant should be turned into a gas (it should evaporate) by the time it gets to the end of the evaporator coil. Several types of thermostatic expansion valves are installed in aircraft air conditioning systems. This section discusses both internally and externally equalized TEVs. Before discussing these valves, we must understand the term "superheat." Superheat is heat energy added to a refrigerant after it has changed from a liquid into a vapour. Refrigerant that has superheat in it is not hot, it is very cold.

The Evaporator

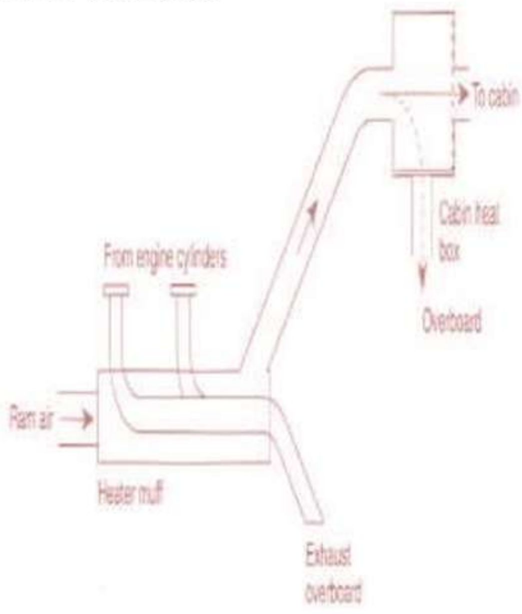
The evaporator is the part of the air condition ing system where the cold air is produced. It is made of a series of tubes over which thin sheet aluminium fins have been pressed. The area provided by the fins allows a maximum amount of heat to be picked up from the air inside the cabin and transferred into the refrigerant inside the evaporator tubing.

The evaporator is usually mounted inside a shroud in such a way that a blower can pull hot air from inside the cabin and force it through the evaporator fins. After the air leaves the evaporator, it blows over the occupants of the cabin. The blower is equipped with a speed control that allows the pilot to vary the amount of air blowing across the evaporator coils. The thermostatic expansion valve is mounted at the inlet of the evaporator, and it breaks the refrigerant up into a fine mist and sprays it out into the coils. The refrigerant flowing through the coils picks up heat from the fins, is warmed, and turns into a gas. The air passing through the fins loses some of its heat and is cooled.

The temperature-sensing bulb of the TEV is clamped to the discharge line of the evaporator, and it is insulated with tape so it is not affected by any temperature except that caused by the refrigerant vapors inside the evaporator.

Aircraft Heaters

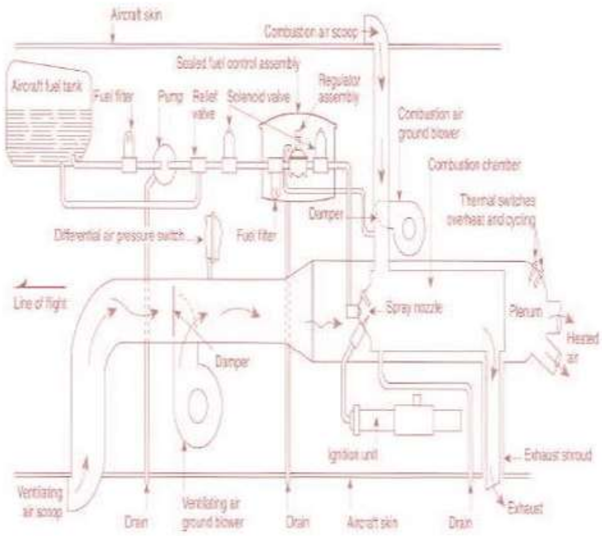
Aircraft environmental control systems include heaters, cooling systems pressurization systems, and supplemental oxygen. The most widely used environmental control devices are heaters, which are installed in almost all aircraft, from the smallest trainers to the largest transport aircraft. In this section we discuss exhaust system heaters and combustion heaters. The section on air-cycle air conditioning systems discusses cabin heat taken from engine compressor bleed air.



Exhaust System Heaters

Most of the smaller aircraft use jackets, or shrouds, around part of the engine exhaust system to provide heat

for the cabin. Air flows around the exhaust component and picks up heat before it is carried into the cabin. When the cabin heat valve is ON, the heated air is directed into the cabin. When it is OFF, this hot air is dumped overboard. Aircraft that use this type of heater should have their exhaust system regularly inspected for cracks or other leaks. One acceptable way of checking exhaust systems is to remove the heater shroud, pressurize the system with the pressure discharge of a vacuum
cleaner, and paint the outside of the system with a soap and water solution. Leaks will cause bubbles to appear.
Carbon monoxide detectors should be used in the cabin to detect any trace of carbon monoxide. These are simply small packets of crystals that are stuck to the instrume nt panel in plain sight of the occupant
s. These crystals are normally a bright color, but when they are exposed to carbon monoxid e, they darken .They turn black when exposed to a level of CO that could cause illness.



Combustion Heaters

Some aircraft arc heated with combustion heaters that use fuel from the aircraft fuel tanks. A typical combustion heater system schematic is shown in Figure. Fuel flows from the tank through a filter and an electric fuel pump and relief valve, then through an overheat solenoid valve into the fuel control assembly. In this assembly there is another filter, a fuel-pressure regulator, and a thermostat operated solenoid valve. From this assembly, the fuel flows to the spray nozzle inside the combustion chamber.

Combustion air is taken into the heater from the main air intake or from a separate outside air scoop, and the air pressure varies with

the airspeed. A differential-pressure regulator or a combustion-air relief valve prevents too much air from entering the heater as the airspeed increases. An electrically driven blower ensures a consistent flow of air into the combustion chamber. The heat produced by a combustion heater is controlled by a thermostat cycling switch that cycles the fuel on and off. When more heat is required, the fuel is turned on. When the correct temperature is reached, the fuel is turned off automatically. An overheat switch shuts the fuel off if the temperature at the discharge of the heater becomes too high.

Ventilation

A re-circulation fan is used to re-circulate filtered cabin air back into the cabin to reduce bleed air requirements. Up to 50% of the cabin air can recirculated for passenger comfort. The fan will switch off if either A/C pack is in high flow, giving a net reduction in the ventilation rate. Optimum cooling and reduced bleed air demand (hence reduced fuelconsumption) is achieved with a combination of air re-circulation and

automatic operation of the A/C packs. The ventilation rate is increased on larger aircraft with additional re-circulation fans for comfort levels.
Some aircraft are installed with a gasper fan ; this is an electric fan designed to increase pressure in the outlets above passenger seats. The gasper fan is typically used when there is a:

- low supply of air pressure
- high cold air demand
- high ambient temperature (on the ground on a hot day).

When the A/C packs are off, the gasper fan draws cabin air into the distribution ducts, into the main air manifold and mixing chamber where it is then blown (albeit not chilled) into the cabin.

Equipment cooling

Although not part of the cabin system for the comfort of passengers, some aircraft are installed with equipment cooling that takes air from the cabin system. Equipment cooling can be used for:

- flight compartment panels
- display units
- circuit breaker panels
- electrical and electronic (E & E) bay.

Replacing the warm air generated by equipment with cool air from the cabin is achieved with dedicated electric fans. On the ground, the air is directed overboard through the flow control or exhaust valves.
During flight, the warm air generated by equipment is sometimes used for heating the cargo holds by exhausting air around their walls.

Air Conditioning System Checks

With the system turned on and the engine running at a fast idle, a normally functioning air conditioning system will blow a stream of cold air out from the evaporator. All of the components in the high side of the system should feel hot or warm to the touch. All of the components in the low side of the system should feel cold or cool to the touch. The actual temperature of the air as it leaves the evaporator depends on the air's humidity and the ambient air temperature, but it should be in the range of 35° to 45°F.

Visual Inspection

The entire air conditioning system should be checked visually for its condition. Begin with one part of the system and check it through the entire system.

Check the evaporator to be sure it is mounted securely and that there is a clear airflow path through its shroud.

The fins must be free of lint and dirt, and there must not be any fins bent over to obstruct the air flowing through them. The blower must operate at all speeds and not rub against its housing.

The sensor for the thermostatic expansion valve must be securely taped to the discharge of the evaporator, and covered so it will not be affected by any temperature other than that of the evaporator coil.

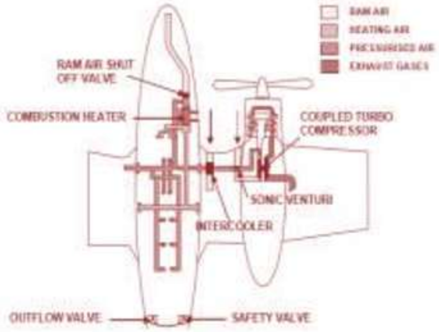
The thermostat switch must be secured in such a way that its sensor is in the fins of the evaporator so it can sense the temperature at the point the manufacturer specifies.

Check the compressor for security of mounting, for freedom of operation of the clutch, and for the proper belt ten-sion. The load the compressor places on its mounting as it cycles on and off puts a big strain on the castings, so you should carefully inspect the area around which the compressor is mounted. Check the mounting bolts to be sure none of them have vibrated loose.

Operational Check
After a careful visual check confirms that the air conditioning system is properly mounted in the aircraft, you can give it an operational check. This check consists of connecting a manifold gage set to the system and measuring the pressure of the refrigerant in the system.

Remove the protective cap from the service port in the high side of the system and, after checking to be sure the high-side valve on the manifold gage set is closed, connect the high-side service hose to the valve. Open the high side valve slightly and allow refrigerant to flow out of the center hose for about three to five seconds, then close the valve.

Remove the protective cap from the low-side service port and connect the low-side service hose. Open the low-side valve and



Small Reciprocating Engine Aircraft Pressurisation System
Figure 22

allow refrigerant to flow out of the center hose for three to five seconds, then close the valve.

Allow the system to operate with the engine running at a relatively fast idle for about five minutes, with the blowers operating at high speed and the air conditioning controls calling for maximum cooling. After the system has run five minutes, check the evaporator air discharge temperature and the high side pressures. The pressures are affected by the ambient temperature.

PRESSURISATION SYSTEMS
As aircraft became capable of obtaining altitudes above that at which flight crews could operate efficiently, a need developed for complete environmental systems to allow these aircraft to carry passengers. Air conditioning could provide the proper temperature and supplemental oxygen could provide sufficient breathable air.
The problem was that not enough atmospheric pressure exists at high altitude to aid breathing in and even at lower altitudes the body must work harder to absorb sufficient oxygen, through the lungs, to operate at the same level of efficiency as at sea level. This problem is overcome by pressurising the cockpit/ cabin area. Cabin pressurisation is a means of adding pressure to the cabin of an aircraft to create an artificial atmosphere that when flying at high altitudes it provides gives an environment equivalent to that below 10000 feet. The minimum quantity of fresh air supplied to each person on board must be at least 0.5lb/ minute.

Aircraft are pressurised by sealing off a strengthened portion of the fuselage. This is usually called the pressure vessel and will normally include cabin, cockpit and possibly cargo areas. Air is pumped into this pressure vessel and is controlled by an outflow valve located at the rear of the vessel.
Sealing of the pressure vessel is accomplished by the use of seals around tubing, ducting, bolts, rivets, and other hardware that pass through or pierce the pressure tight area. All panels and large structural components are assembled with sealing compounds. Access and removable doors and hatches have integral seals. Some have inflatable seals.

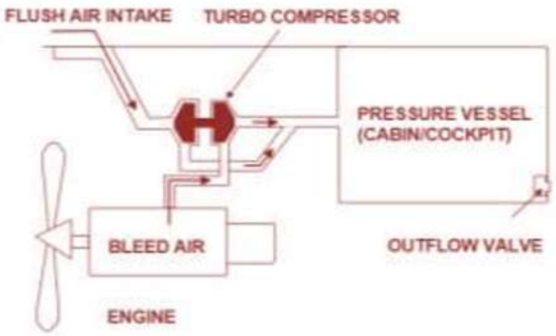
Pressurisation systems do not have to move large volume of air. Their function is to raise the pressure inside the vessel. Small reciprocating engine powered aircraft receive their pressurization air from the compressor of a coupled turbocharger. Larger reciprocating engine powered aircraft receive air from engine driven compressors and turbine powered aircraft use compressor bleed air

Small Reciprocating Engine Powered Aircraft

Turbochargers are driven by the engine exhaust gases flowing through a turbine. A centrifugal compressor is coupled to the turbine. The compressors output is fed to the engine inlet manifold to increase manifold pressure which allows the engine to develop its power at altitude. Part of this compressed air is tapped off after the compressor and is used to pressurise the cabin. The air passes through a flow limiter (or sonic venturi) and then through an inter- cooler before being fed into the cabin. A typical system is shown at Figure 22.

A sonic venturi is fitted in line between the engine and the pressurisation system. When the air flowing across the venturi reaches the speed of sound a shock wave is formed which limits the flow of air to the pressurisation system.

Large Reciprocating Engine Powered Aircraft



Turbo Compressor
Figure 23

These aircraft use engine driven compressors driven through an accessory drive or by an electric or hydraulic motor. Multi engine aircraft have more than one air compressor. These are interconnected through ducting but each have a check valve or isolation valve to prevent pressure loss when one system is out of action.

Turbine Powered Aircraft

The air supplied from a gas turbine engine compressor is contamination free and can be suitably used for cabin pressurisation (Figure 23). Some aircraft use an independent compressor driven by the engine bleed air. The bleed air drives the coupled compressor which pressurises the air and feeds it into the cabin.

Some aircraft use a jet pump to increase the amount of air taken into the cabin (Figure 24). The jet pump is a venturi nozzle located in the flush air intake ducting. High velocity air from the engine flows through this nozzle. This produces a low pressure area around the venturi which sucks in outside air. This outside air is mixed with the high velocity air and is then passed into the cabin.

Control And Indication

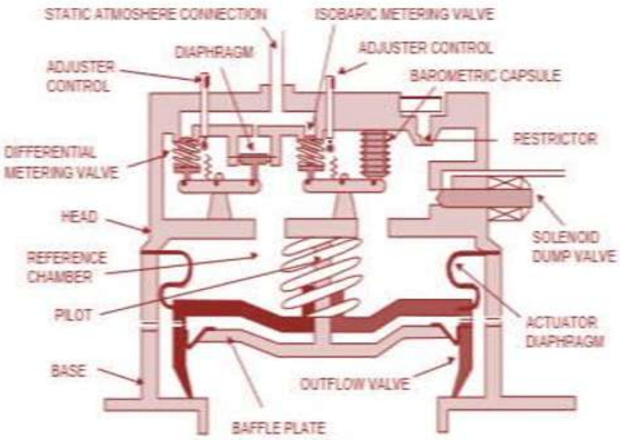
There are 3 modes of pressurisation, un- pressurised, the isobaric mode and the constant–differential pressure mode. In the

un-pressurised mode the cabin altitude remains the same as the flight altitude. In the isobaric mode the cabin altitude remains constant as the flight altitude changes and in the constantdifferential pressure mode, the cabin pressure is maintained at a constant amount above the outside ambient air pressure.

The amount of differential pressure is determined by the structural strength of the aircraft. The stronger the

aircraft structure the higher the differential pressure and the higher is the aircrafts operating ceiling.
The Un-Pressurised Mode
In this mode the outflow valve remains open and the cabin pressure is the same as the outside ambient air pressure. This mode is usually from sea level up to 5000` but does vary from aircraft to aircraft.

The Isobaric Mode
In this mode the cabin pressure is maintained at a specific cabin altitude as flight altitude changes. The cabin pressure controller begins to close the outflow valve as the aircraft climbs to a chosen
cabin altitude. The outflow valve then opens or closes (modulates) to maintain the selected cabin altitude as the flight altitude changes up or down. The controller will then maintain the selected cabin altitude up to the flight altitude that produces the maximum differential pressure for which the aircraft structure is rated. At this point the constant differential mode takes control.
The Constant-Differential Pressure Mode
Cabin pressurisation puts the aircraft structure under a tensile stress as the cabin pressure expands the pressure vessel. The cabin differential pressure is the ratio between the internal and external air pressures. At maximum constant-differential pressure as the aircraft increases in altitude the cabin altitude will increase but the internal/external pressure ratio will be maintained. There will be a

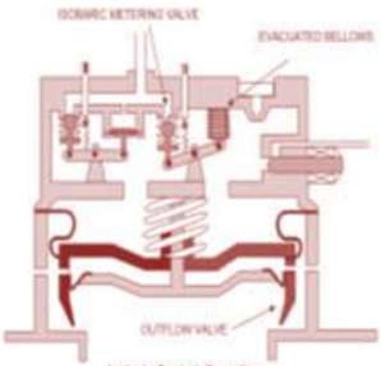


Cabin Pressure Regulator
Figure 25

maximum cabin altitude allowed and this will determine the ceiling at which the aircraft can operate.
Cabin Air Pressure Regulator
The pressure regulator maintains cabin altitude at a selected level in the isobaric range and limits cabin pressure to a pre-set pressure differential in the differential range by regulating the position of the outflow valve. Normal operation of the regulator requires only the selection of the desired cabin altitude and cabin rate of climb the adjustment of the barometric control.
The regulator shown in Figure 25 is a typical differential pressure type regulator that is built into the normally closed air operated outflow valve. It uses cabin altitude for its isobaric control and barometric pressure for the differential control. A cabin rate of climb controller controls the pressure change inside the cabin.
The actuator diaphragm extends outward from the outflow valve to the head assembly creating an air chamber between the head and the inner face of the outflow valve. Air from the head and reference chamber exert a force against the inner
face of the outflow valve helping the spring to hold the valve closed.
The position of the outflow valve controls the amount of cabin air that is allowed to flow from the pressure vessel and this controls the cabin pressure. The position of the outflow valve is determined by the amount of reference chamber air pressure that presses on the inner face of the outflow valve.

Isobaric Control System

The isobaric control system of the pressure regulator shown in Figure 26 incorporates an evacuated capsule, a rocker arm, valve spring and a ball type metering valve. One end of the rocker arm is connected to the valve head by the evacuated capsule and the other end of the arm holds the metering valve in a closed position. A valve spring located on the metering valve body tries to move the metering valve away from its seat as far as the rocker arm allows. When the cabin air pressure increases enough for the reference chamber air pressure to compress the evacuated capsule the rocker arm pivots around its fulcrum and allows the metering valve to move away from its seat an amount proportional to the compression of the capsule. When the metering valve opens reference pressure air flows form the regulator to atmosphere through the atmospheric chamber.



Isobaric Control Operation
Figure 26

When the regulator is operating in the isobaric range, cabin pressure is held constant by reducing the flow of reference chamber air through the metering valve. This prevents a further decrease in reference pressure.
The isobaric control responds to slight changes in reference pressure by modulating to maintain a constant pressure in the chamber throughout the isobaric range of operation. Whenever there is an increase in cabin pressure the isobaric metering valve opens which decreases the reference pressure and causes the outflow valve to open which then decreases the cabin pressure.

Differential Control System

The differential control system of the pressure regulator (Figure 27) incorporates a diaphragm a rocker arm, a valve spring and a ball type metering valve. One end of the rocker arm is attached to the head by the diaphragm which forma a
pressure sensitive face between the reference chamber and the atmospheric chamber.
Atmospheric pressure acts on one side of the diaphragm and reference chamber pressure acts on the other. The opposite end of the rocker arm holds the metering valve in a closed position. A valve spring located on the metering valve body tries to move the metering valve away from its seat as far as the rocker arm allows. When reference chamber pressure increases to the system differential pressure limit set above the decreasing atmospheric pressure it collapses the diaphragm which is set at differential pressure and opens the metering valve. Air flows from the

reference chamber to atmosphere through the atmospheric chamber, which causes a reduction in the reference pressure. This reduction in reference pressure causes the outflow valve to open to reduce the cabin pressure to maintain the system pressure differential.

Safety Valves

Cabin Air Pressure Safety Valve

The pressure relief valve prevents cabin pressure from exceeding the predetermined cabin to ambient pressure differential. A negative pressure relief valve and pressure dump valve may also be incorporated into this valve assembly.

Negative Pressure Relief Valve

A pressurised aircraft is designed to operate with the cabin pressure higher than the outside air pressure. If the cabin pressure were to become lower than the outside air pressure the cabin structure could fail. Outside air is allowed to enter the cabin to ensure that this does not happen. It is basically an inward pressure relief valve.

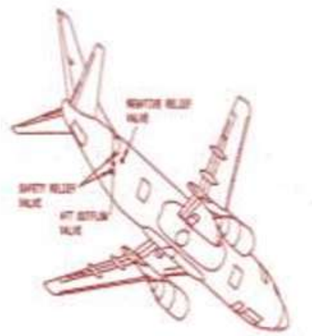
Dump Valve

This valve is normally solenoid actuated by a cockpit switch. When the solenoid is energised the valve opens dumping cabin air to atmosphere. Cabin pressure will decrease rapidly until it is the same as the outside air pressure and cabin altitude will increase until it is the same as the flight altitude.

Ditching valve

If any of the cabin control valves were situated below the water level and the aircraft ditch in the water, the cabin would quickly flood. To prevent this happening, either a mechanical or electrical ditching selection, can be made by the crew to seal off all pressurisation valves and inlets.

ELECTRONIC PRESSURISATION CONTROL



Pressurisation System Valves
Figure 31

Most modern airliners have the means to electronically control the cabin pressure automatically for the entire flight, from settings made by the flight crew before take off.

The pressure control system consists:

with pressure sensing inputs and outputs to monitoring indicators

-driven gate- outward safety relief valves.

Flight Deck Control Panel

This provides a means for the flight crew to control the cabin pressure by positioning the outflow valve. There are three mode selections available; _Auto‘, _Standby‘ or _Manual‘.

The desired mode will normally be _Auto‘, where all settings such as intended cruise (flight) altitude and destination airfield (landing) altitude are made before flight. This will allow automatic control of cabin pressure for the whole of that flight. This is called the fully automatic mode. Alternatively, _Standby‘ or back up mode can be selected, where a cabin altitude setting must be made for each desired cabin pressure change. The input setting is then controlled automatically as before. This is called the semi-automatic mode. If neither the fully or semi-automatic modes are available, (i.e.: the pressure controller fails), the outflow valve can be positioned directly from the flight deck by operating the electric torque motors to drive the valve. This is called the manual mode and a choice of an ac or dc electrical supply is available.

Automatic Pressure Controller

The pressure controller provides output control signals to the outflow valve‘s ac or dc torque motors. The motors position and modulate the valve to establish and control actual cabin pressure in accordance with the controller‘s pre-programmed climb, cruise or descent schedules. This will ensure that for every aircraft altitude there will be a particular cabin altitude. Input signals to the controller are from the flight deck control panel, cabin and ambient pressure sensors, barometric correction and air/ground sensing.

Outflow Valve

The valve has a moving gate designed to cover or uncover an aperture in the fuselage skin. An increase in the aperture size will
cause cabin pressure to fall (cabin altitude to ascend), whereas a decrease in the aperture size results in an increase in cabin pressure (cabin altitude to descend). The gate is driven by one of two electrically driven motors, the choice of ac or dc motor being

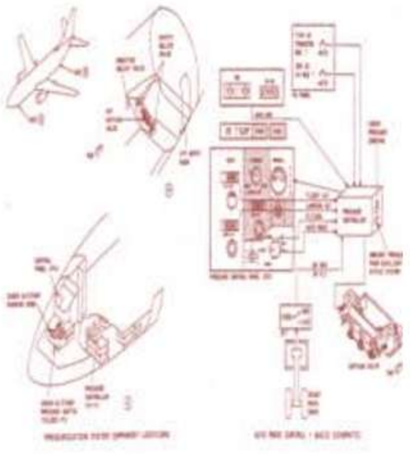


Figure 28

determined by flight crew input. Motor input signals come from the controller when in the auto or standby modes, or directly from a control panel switch when in the manual mode.

4 Inward and Outward Safety Relief Valves

Fuselage frames are designed to accept tensile loads associated with and outward force from within the pressure cell. Their ability to withstand compression loads that would occur if the pressure
outside the aircraft were higher than within the pressure cell is poor. Therefore an inward relief valve will open and equalise the pressure if the inward or negative differential exceeds about 0.5 psid. Two outward relief valves are fitted to prevent the maximum outward differential pressure from exceeding the structural limit. This will typically be around 8.5psid. Even though the main pressure control is electronic, the safety relief valves are mechanical operated and are completely independent of any automatic control system.

CABIN PRESSURE INDICATION

Most pressurisation systems have three basic cockpit indicators cabin altitude, cabin rate of climb and the pressure differential indicator. The cabin altitude gauge measures the actual cabin altitude.

The cabin rate of climb indicator tells the pilot the rate that the cabin is either climbing or descending. (I.e. the rate at which the cabin loses or gains pressure) A typical maximum climb rate is 500ft per minute and the maximum descent rate is 300ft per minute. The control can be automatic or manual depending on aircraft type.

The differential pressure gauge (Figure 34) reads the difference between the cabin and the outside air pressures. This differential pressure is normally controlled and maintained to a structural limitation around 7psid. This depends on the aircraft type and the operating ceiling of the aircraft. The differential pressure gauge may be combined with the cabin altitude (Figure 35).

SAFETY AND WARNING DEVICES

To ground test the pressurisation system with the engines running, at least three men are required inside the aircraft for safety reasons. Both air conditioning and pressurisation systems use safety and warning devices to protect the aircraft from possible catastrophic failures. Some of the protection devices may be inhibited in certain stages of flight; landing or take off where the extra distractions caused by such warnings may be too much for the crews to deal with safely.

With the air conditioning system the main concerns are with overheating of the air conditioning packs and extraction and ventilation fans, as well as hot air leaks from ducting which could damage surrounding structure or components.

Overheating

Most packs systems are protected from overheating by a thermal switch downstream of the pack outlet. If the outlet temperature reaches a pre determined figure the switch will operate causing the pack valves to shut, preventing air from getting to the packs, as well as sending a warning signal to the cockpit central warning panel with associated caution/warning lights and aural chimes and to illuminate a fault light on the pack selector switch.

Once the system has cooled down sufficiently the crew may have an option to reselect the overheated system. The overheat may have been caused by a fault in the automatic temperature control system in which case the pilot may be able to control the system manually via a manual selector switch on the cockpit controller.

Extraction or ventilation fans will be protected in much the same way. An overheat will signal the central warning panel with associated caution/warning lights and aural chimes. The fan may be isolated automatically or manually. Once the fan has cooled down it may be possible to re-select if required. Fans may also be protected from over or under speeding, which will also have an effect on the system temperatures. Speed sensors on the fan will indicate a fault when over or under speed limits are reached and a warning signal is sent to the cockpit central warning panel with associated caution/warning lights and aural chimes.

Duct Hot Air Leakage

Any ducting that includes joints is liable to leak under abnormal conditions. A duct protection system will include fire-wire elements around the hot zones such as engine air bleeds, air conditioning packs and auxiliary power units if fitted.

The sensing elements will be the thermistor type. As the temperature around the wire increases the resistance decreases until an electrical circuit is made. When the circuit is made a warning signal is sent to the cockpit central warning panel with associated caution/warning lights and aural chimes. The leaking duct may be isolated automatically or may require the pilot to take action to close off the air valves. The faulty system will then remain out of use.

Excess Cabin Altitude

If the cabin altitude was allowed to increase unchecked the crew and passengers could unknowingly suffer the effects of hypoxia. This dangerous condition is obviously undesirable especially for the aircrew. Most aircraft give a warning on the CWP with associated audio and visual warnings when the cabin altitude reaches 10000`.

Smoke Detection

Smoke detectors may be fitted within the cabin; avionics bay and cargo areas to monitor systems, which if become faulty may generate smoke on overheating, or are may be liable to catch fire.

These detectors will send a signal to the CWP with associated lights and audio warnings. They may also automatically switch on extractor fans, which will remove the smoke overboard and away from the cabin and cockpit areas. In this event, the pilot may have a switch or control lever to operate a valve to isolate the cockpit air conditioning ducting from the rest of the aircraft to prevent any smoke from getting to the cockpit.

13.12 FIRE PROTECTION (ATA 26)

(a) Fire and smoke detection and warning systems;

Fire Protection

Aircraft carry large volumes of highly flammable fuel in a lightweight, vibration-prone structure. This structure also carries engines that continually produce extremely hot exhaust gases. Add a complex electrical system with motors and relays that produce sparks, and radio and radar transmitters that emit electromagnetic radiation, and you have an ideal environment for fires.
Requirements for Fire
Fire is the result of a chemical reaction between some type of fuel and oxygen. When this reaction occurs, energy is released in the form of heat and light.
For a fire to start there must be fuel, oxygen, and a high enough temperature to start the reaction. Fires may be extinguished by removing the fuel or oxygen or by reducing the temperature to a level below that needed for the reaction.

The National Fire Protection Association has categorized fires and identified the types of extinguishing agents best used on each type. The four categories are Classes A, B, C, and D.
Class A fires are fueled by solid combustible materials such as wood, paper, and cloth. These fires typically occur in aircraft cabins and cockpits, so any extinguishing agent used for Class-A fires must be safe for the occupants.

Class B fires are fueled by combustible liquids such as gasoline, turbine- engine fuel, lubricating oil and hydraulic fluid. Class-B fires typically occur in engine compartments.

Class C fires involve energized electrical equipment. These fires can occur in almost any part of an aircraft and they demand special care because of the danger of electrical shock.

Class D fires arc those in which some metal such as magnesium burns. These fires typically occur in the brakes and wheels, and burn with a ferocious intensity. Never use water on a burning metal it only intensifies the fire.

Overview
The type of fire protection systems and equipment fitted to an aircraft can be sub-divided into specific areas of the aircraft:

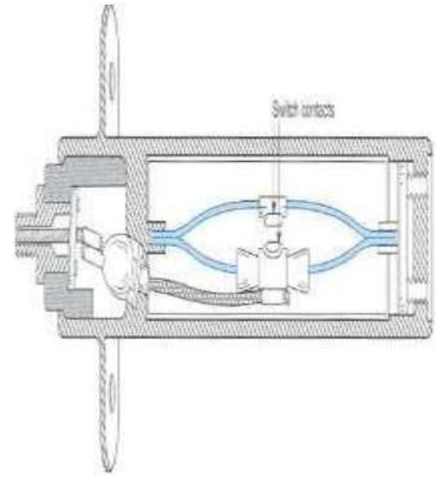
- engines/APU
- cargo bay
- passenger cabin.

The fire protection technologies used on aircraft depend on these areas and specified fire risk. The two agents used extensively in the aircraft industry were:

- bromochlorodifluoromethane, also known by the trade name Halon 1211, or BCF
- bromotrifluoromethane, also known by the trade name Halon 1301, or BTM.

These are both inert gases and are either applied locally (typically in hand-held extinguishers) or as total flooding applications (typically in cargo bays). Both these agents are very effective at extinguishing fires; however they are in a group of halogenated hydrocarbons that have been shown to contribute to depletion of the earth's ozone layer. Although halon fire extinguishers are still being specified on aircraft, both gases come under the terms of the

1989 Montreal Protocol (and subsequent revisions), that prohibits the new production of these agents. The industry is supporting the supply of halon 1211 and 1301 through recycling of existing halon stocks for in-service and new production aircraft. Alternative and replacement agents are being developed, and these are gradually being introduced throughout the fire protection industry. Examples of agents that do not deplete the ozone layer include:



- water
- dry powder
- vaporizing liquids (gaseous agents)

Combustion can be defined as a rapid and complex chemical reaction in which light and heat are evolved. All equipment in an aircraft is carefully designed and tested so that there is a low probability of starting and sustaining fire. Three factors are required to initiate and sustain combustion:

- fuel
- heat
- oxygen

Fuels include solids, liquids and gases; each type of fuel requires a minimum temperature to be reached and maintained. Oxygen is normally available from the ambient air. The type of fire detector and extinguishing agent deployed is largely determined by which fuel is likely to combust.

When a fire is detected, the warning system activates an alarm bell and illuminates red warning lights in the flight compartment. The fire warning also illuminates a master warning light and another light (typically in the fire handle) to identify the affected engine(s). When the crew confirms that they have an engine fire, they operate

the fire handle by first pulling and then twisting the handle. The action of pulling the fire handle activates micro-switches that shut down the engine by:

- closing the fuel shutoff valves
- opening the field circuit of that engine's generator(s)
- closing the bleed air supply from the engine into the pneumatic system

● closing the hydraulic systems engine driven pump shutoff valve.

Fires are detected in large reciprocating engine aircraft and small turboprop engine aircraft using one or more of the following:

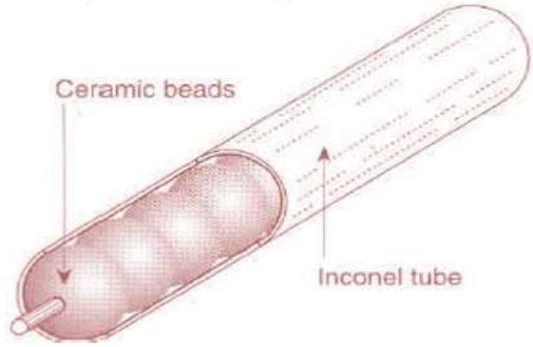
1. Overheat detectors/ thermal sensors- thermal switch, continuous loop, Rate of temperature rise detectors i.e.thermocouple
2. Flame detectors – optical sensors
3. Smoke detectors
4. Observation by crewmembers

The types of detectors most commonly used for fast detection of fires are the thermal sensors- thermal switch type, thermocouple, continuous loop and optical sensors and smoke detectors.

Thermal fire detection:

Thermo switch-type detector Unit fire detectors (Spot detectors)

These are effectively thermostatic overheat switches based on the bimetallic principle. The two contacts are attached to spring bows; these are compressed during



assembly and held apart by the outer barrel of the switch. As the detector is heated, it expands and the two contacts close, thus completing the warning circuit. An adjusting screw is found on the end-cap of some switches to allow for minor adjustments of operating temperature. This type of detector can be used in engine fire detection systems; alternatively it can be used in the bleed air overheat detection system. The detector can only sense fire/overheat in a localized volume of the installation; it is more likely that multiple detectors are used to provide increased detection probability.

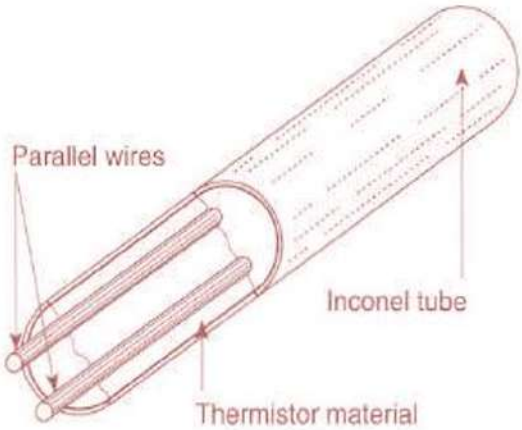
A small gas turbine engine would typically be fitted with six of these detectors. The wing leading of a large aircraft could be fitted with over 20 overheat switches. The switches are connected in parallel this forms a detection loop. If any one of the switches closes due to a fire or overheat being detected, the alarm circuit is activated thereby illuminating a system warning light. A simple test circuit allows some of the circuit to be tested from the flight compartment. Some aircraft are fitted with a dual loop of detectors to provide a back-up system in the event of loopv failure, e.g. open circuit wiring.

Rate-of-Temperature-Rise Detection System i.e. thermocouple A thermo switch-type detection system initiates a fire warning when any of the individual detectors reaches a predetermined temperature. But because a fire can have a good start before this temperature is reached, the thermocouple- type fire-warning system is used. This system initiates a fire warning when the temperature at any specific location in the monitored compartment rises a great deal faster than the temperature of the entire compartment. Thermocouple- type fire-warning systems are often installed in engine compartments where normal operating temperatures are quite high, but the rise to this temperature is gradual. A thermocouple is made of two different types of wire welded together, and the point at which the wires are joined is

called a junction. When several thermocouples are connected in series in a circuit, a voltage will exist within the circuit that is proportional to the difference in the temperatures of the various junctions.
The sensors are used with a thermocouple system . These sensors have a piece of each of the two thermocouple wires, typically iron and constantan, welded together and mounted in the housing that protects them from physical damage, yet allows free circulation of air around the wires. They form the measuring junctions of the thermocouple, and all of them are connected in series with the coil of a sensitive relay and a test thermocouple.

The sensors are mounted at strategic locations around the monitored compartment. One sensor is mounted inside a thermal insulating shield that protects it from direct air circulation, yet allows it to reach the temperature ofthe air within the compartment. This sensor is called the reference junction. When there is no fire, all of the junctions are the same temperature and no current flows in the thermocouple circuit. When the engine is started and the temperature of the engine compartment rises, the temperatures of all of the thermocouples rise together and there is still no current flow. But if there is a fire, the temperature of one or more of the thermocouples will rise immediately while the temperature of the insulated reference thermocouple rises much more slowly. As long as there is a difference in temperatures between any of the junctions, there is a difference in voltage between them. Not much current, but enough to energize the sealed sensitive relay, flows in the thermocouple circuit. The contacts of the sensitive relay close and carry enough current to the coil of the slave relay to close its contacts and allow current to flow to the fire-warning light and bell.

Continuous-Loop Detector Systems



Engine compartments, APU installations, and wheel wells are difficult locations to monitor for fire, and continuous-loop-type detectors are often used in these areas rather than individual detectors such as thermoswitches or thermocouples. There are two types of continuous-loop fire and overheat detection systems: thermistor and pneumatic.

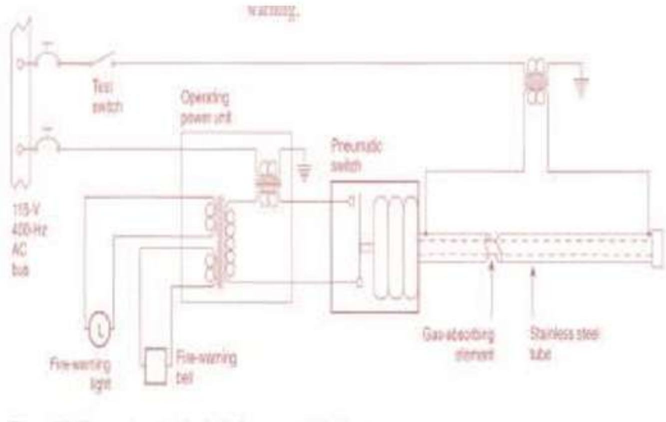
Thermistor-Type Continuous-Loop Systems
There are two configurations of thermistor-type continuous loop elements: single-conductor elements (fenwal sensing element) and two-conductor elements(kidde sensing element).
Fenwal sensing element:
The single-conductor element has a center conductor supported in a thin wall inconel tube by ceramic beads. An electrical connection is made to the conductor, and the outside tube is grounded to the airframe. The space between the beads is filled with a eutectic (low melting-point) salt whose resistance drops drastically when it melts. When any portion of the tube gets hot enough to melt the salt, the resistance between the centre conductor and the outside tube drops, and signal current flows to initiate a fire warning. When the fire is extinguished, the molten salt solidifies and its resistance increases enough that the fire-warning current no

longer flows.

Kidde sensing element:
The two conductor loop is also mounted in an inconel tube, and it has two parallel wires embedded in a thermistor material whose resistance decreases as its temperature increases. One of the wires is grounded to the outer tube, and the other terminates in a connector and is connected to a control unit thatcontinuously measures the total resistance of the sensing loop. By monitoring the resistance, this unit will detect a general overheat condition as well as a single hot spot Pneumatic-Type Continuous-Loop System
The pneumatic fire detection system also uses a continuous loop for the detection element, but this loop is made of a sealed stainless steel tube that contains an element which absorbs gas when it is cold, but releases this gas when it is heated.

One type of pneumatic fire
detection system is the Lindberg system. The stainless steel tube which makes up the loop contains the gas-absorbing element and the gas, and is connected to a pressure switch .When the loop, which is installed around the monitored area, is heated in a local area by a fire or by a general overheat condition, the gas is released and its pressure closes the pressure switch.

Closing this switch completes the circuit for one of the windings of a transformer and allows the 115-volt, 400-Hz power from the aircraft electrical system to illuminate the fire-warning light and sound the fire-warning bell.
This system is tested by closing the test switch. This allows low- voltage AC to flow through the tubing in the loop. This current heats the loop and causes the release of enough gas to close the pressure switch and initiate a fire warning.



Optical fire detection/ flame detector Optical sensors sense the light emitted from a flame in much the same way that a person recognizes fires.
Our brains can distinguish between the light energy from a fire and the light energy from another source, e.g. a light bulb. The optical fire detector (or sensor) also has to be able to make this distinction. Some aircraft use optical fire detectors in place of thermal detector elements to simplify the installation.
Depending on the size of engine, several optical sensors may be required. The sensors are easier to install and maintain than linear detector elements. Optical sensors offer the advantage of being able to monitor a specific volume of engine nacelle.
The spectral analysis of a hydrocarbon n fuel fire reveals peaks of energy in the infrared (IR) and ultraviolet (UV) frequencies bands. Optical fire detectors are designed to detect radiation one or both of these frequency bands. The type of detection technology used depends on a number of factors including the:

- speed of response to a fire
- ambient temperature for where the sensor is located

- likely source of potential false alarms.

Detecting a fire is one thing; being able to discriminate against other light sources is a vital part of the optical detector‘s performance.
An IR sensor designed for this specific frequency band provides a high level of reliable fire detection, while being relatively immune to nuisance alarm signals. The detection cell incorporates a pyro- electric cell and an optical filter; the latter only transmits radiation within the wavelength band of 4.2 to 4.7 microns.
This is packaged in a one-inch-diameter, three inches- long cylindrical housing with an optical window at one end and electrical connector at the other. The pyro-electric detection cell responds to a fire by generating a signal when 4.4 micron radiation energy is detected. The optical fire detector has a cone of vision; 100% represents the maximum detection distance for a given fire. The sensitivity of the detector increases as the angle of incidence decreases.

As with any fire detector (whether thermal or optical). The response time of the detector depends on the:

- size of the fire
- rate of propagation
- type of fuel burning
- distance from the detector.

Smoke Detectors

A smoke detection system monitors the lavatories and cargo baggage compartments for the presence of smoke, which is indicative of a fire condition. Smoke detection instruments that collect air for sampling are mounted in the compartments in strategic locations. A smoke detection system is used where the type of fire anticipated is expected to generate a substantial amount of smoke before temperature changes are sufficient to actuate a heat detection system. Two common types used are light refraction and ionization.

Light Refraction Type

The light refraction type of smoke detector contains a photoelectric cell that detects light refracted by smoke particles. Smoke particles refract the light to the photoelectric cell and, when it senses enough of this light, it creates an electrical current that sets off a light.

Ionization Type

Some aircraft use an ionization type smoke detector. The system generates an alarm signal (both horn and indicator) by detecting a change in ion density due to smoke in the cabin. The system is connected to the 28 volt DC electrical power supplied from the aircraft. Alarm output and sensor sensitive checks are performed simply with the test switch on the control panel.

Extinguishing a fire can be achieved by one or more of the following strategies:

- limiting or eliminating the fuel
- limiting or eliminating the oxygen
- reducing the temperature of the fire
- interfering with the chemical reaction.

Fires can be categorized by the types of fuel that are combusting; this in turn determines the detection and extinguishing strategy.
Primarily sources of risk/fuel in aircraft applications are:

- engines and APUs: class B
- cabin/flight compartment: class A
- cargo bay: class A.

Fire-Extinguishing Systems

Fire protection systems divide themselves logically into two categories:
Firedetection and fire extinguishing. The fire-extinguishing systems furthermore divide into hand-held and installed systems. Here we will consider the various types of fire-extinguishing agents, then the hand-held extinguishers, and finally, the installed systems.

Fire-Extinguishing Agents

Since fire is the chemical reaction between a fuel with oxygen, it can be controlled by interfering with this reaction. This can involve removing the fuel, smothering the fuel with a substance that excludes the oxygen, or lowering the temperature of the fuel. The most effective method for extinguishing aircraft fires involves using a chemical compound that combines with the oxygen to prevent it from combining with the fuel.

Water

Class A fires can be extinguished with an agent, such as water, that lowers the temperature of the fuel. Small hand-held fire extinguishers contain water that is adequately protected with an antifreeze agent. When the handle of these extinguishers is twisted, the seal in a carbon dioxide (C02) cartridge is broken, and the C02 pressurizes the water and discharges it in the form of a spray.

When the water changes from a liquid to a vapor, it absorbs heat from the air above the fire and drops its temperature enough to cool the fuel enough to cause the fire to go out. Never use water on Class B, C, or D fires. Most flammable liquids float on water, and the use of water on Class B fires will only spread the fire.

Water conducts electricity, and its use on a Class C fire constitutes a definite danger of electrocution. Water sprayed on the burning metal in a Class D fire will actually intensify the fire rather than extinguish it.

Inert Cold Gas Agents

Carbon dioxide (C02) and liquid nitrogen (N2) are both effective fire extinguishing agents. They both have very low toxicity.

Carbon Dioxide (C02)

C02 is heavier than air, and when it is sprayed on a fire it remains on the surface and excludes oxygen from the combustion process, and the fire goesout. C02 has been a favored extinguishing agent for many years. It is relatively inexpensive, nontoxic, safe to handle, and has a long life in storage.

C02 extinguishers are found in almost all maintenance shops, on most flight lines, and in most ground vehicles. Most of the older aircraft had handheld C02 extinguishers mounted in fixtures in the cabins and cockpits and fixed C02 extinguishing systems in the engine nacelles. These airborne extinguishers have been replaced in modern aircraft by more efficient types.

Liquid Nitrogen (N2)

N2 is more effective than C0 2, but because it is a cryogenic liquid, it must be kept in a Dewar bottle. Some military aircraft use N2 for

inerting fuel tanks and have it available for use in fire- extinguishing systems, primarily for use in extinguishing powerplant fires.

Halogenated Hydrocarbons

This classification of fire-extinguishing agents includes the most widely used agents today, as well as some of the agents used in the past that are no longer considered suitable.
These agents are hydrocarbon compounds in which one or more of the hydrogen atoms have been replaced with an atom of one of the halogen elements such as fluorine, chlorine, or bromine.

13.13 Fuel Systems (ATA 28)

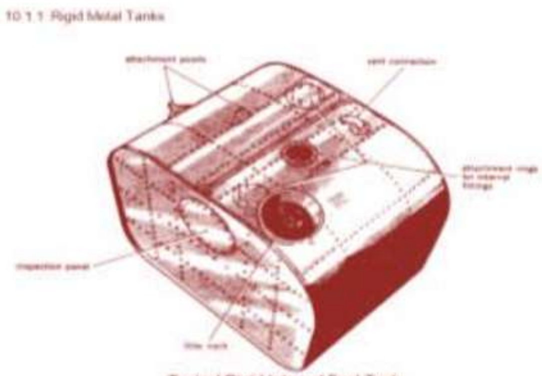
SYSTEM LAY-OUT

The purpose of the fuel system is to store and deliver fuel to the engines and the apu. An aircraft must be able to carry sufficient fuel to enable the engines to operate over long periods. To meet this requirement there must be some way of storing this fuel safely and supplying it to the engines in a suitable condition and at a controlled rate. A typical fuel system therefore will consist of a number of tanks, fuel lines, connections and fittings, which are compatible with all types of fuel meeting engine and apu specifications. Often, the fuel system is subdivided into storage, refuelling, distribution, transfer, venting and indicating subsystems. The following example of a system layout is for a typical large commercial twin aircraft. The number of tanks and system complexity will vary from aircraft to aircraft. Clearly a four-engine aircraft will have more components than a twin. The figure shows a typical fuel cell layout.

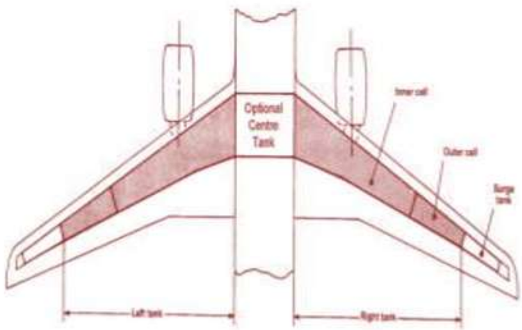
NOTE: For additional range, some operators will install centre tanks, these are offered as optional on most single isle and wide bodied aircraft. Fuel Tanks Fuel tanks normally fall into three categories of construction:

Whatever the construction method, fuel tanks should be shaped so that almost all the fuel is available to the engine. Awkward pockets which prevent fuel from leaving the tank are undesirable and are avoided if possible.

Fuel tanks are made in shapes and sizes to fit the spaces available in each particular airframe and therefore the size and shape of the fuel tanks will not be the same for all aircraft. Metal fuel tanks are constructed from aluminium alloy, stainless steel or tinned steel and they are riveted, welded, or soldered together. The tank is a light structure which is strengthened by the use of internal



Typical Rigid Internal Fuel Tank
Figure 2



Typical Fuel Cell Layout
Figure 1

stiffeners, angle pieces and by incorporating baffles to give strength and which are necessary, in large tanks, to reduce the effects of fuel surge caused when the aircraft manoeuvres. Secure attachment of a rigid tank within the airframe may be achieved by built-in padded cradles and padded metal straps. The cradle is shaped to match the contours of the tank and the straps secure the tank to its cradle. Each tank will have the brackets, strap guides and fittings to match the aircraft structure into which the tank is to be fitted. It must be stressed that very few aircraft over 5,700 kg would utilise metal rigid tanks, except when long range tanks are fitted in the cargo hold, i.e. commercial IATA LD6 containers, etc.

FLEXIBLE FUEL TANKS

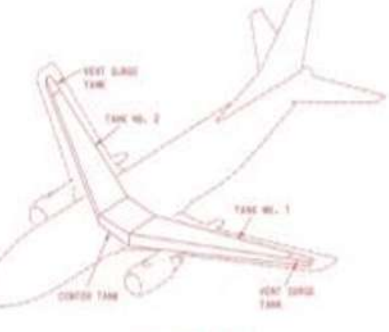
Flexible fuel tanks may be constructed with thin and very flexible walls (called bag tanks) or they may be made of thicker less flexible material. These tanks are made in shapes to fit particular spaces in the aircraft structure and their flexibility enables the tanks to be folded and inserted through a small aperture, which would not allow a rigid tank of similar capacity to be fitted. Because flexible tanks can be made in shapes to suit most of the space available, a greater fuel capacity is made available to a particular aircraft when flexible tanks are used. Some aircraft fuel systems are designed to include rigid, flexible and external fuel tanks so that the greatest possible fuel load is carried. The compartment for a flexible fuel tank is made as smooth as possible on the inside and projecting joints are covered to prevent chafing the tank material. Before a tank can be fitted, the compartment must be properly cleaned out and all swarf and loose items removed. After a flexible tank has been inserted into the tank compartment, the tank is carefully unfolded and the various external fittings are aligned. Usually the walls of the more flexible tanks are attached to the compartment walls by a type of press-stud fitting. When filled with fuel, the tank expands to contact the walls of the tank compartment so that the weight of the fuel is carried by the aircraft structure and not by the tank. Because the load is not carried by the tank, flexing of the aircraft structure does not impose harmful loads upon the tank material. Flexible fuel tanks are resilient, like an inner tube and because they are resilient, the tanks can withstand a considerable amount of distortion or shock loading. If a flexible tank is not completely full it is unlikely to burst on a crash impact.

Tank Coverings PROTECTIVE COVERING

A protective covering may be fixed to the outside of a flexible fuel tank. The covering is not special to type and similar covering materials are used to protect different types of tank. The protective covering usually consists of several layers of fabric, or fabric and rubber, which are cemented to the material of the tank with adhesives. When a tank is fitted with a protective cover it, in general, becomes stiff enough to support its own weight and retain its shape. However, when the various metal fittings are added, the tank will sag and it needs support when fitted. Some tanks, which do not have protective covers, are reinforced by nylon fabric or net. This type of reinforcement does not stiffen the tank, which remains very flexible and limp. This type of tank cannot support its own weight and is the type which is sometimes called a _bag tank".

Self-Sealing Coverings

These coverings have been developed to reduce the magnitude of a fuel leak if, for any reason, the fuel tank is pierced or ruptured. The self-sealing covering is usually made from layers of cellular rubber with an overall protective cover of glass fabric or nylon fabric on the outside. This type of rubber is a material that is immediately affected by contact with fuel. If a tank leaks, the cellular rubber swells on contact with the fuel and forces its way into the puncture to block the hole and reduce or stop the leak. Unfortunately, minor leaks may remain undiscovered for sometime until the self- sealing cover begins to swell and bulge on the outside.

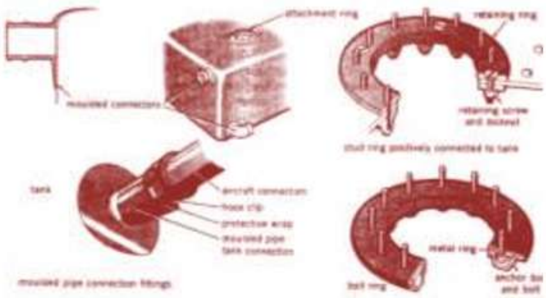


Attachments and Fittings

To complete a flexible fuel tank, provision must be made for attaching fuel system components and for joining each tank into the fuel system. The fuel tank is constructed with moulded connectors and apertures of an appropriate size and position but because of the flexible nature of the material, each aperture needs to be reinforced before a system component can be fitted. Each aperture is strengthened and stiffened by fitting a metal attachment ring. The attachment rings are sometimes called „stud rings" or „bolt rings".

INTEGRAL FUEL TANKS

Primary wing structure is used for aircraft integral tanks. They are normally located between the front and rear wing spars and between the upper and lower wing skin. Solid „tank end" ribs close the ends of each tank, while all the other ribs act as fuel baffles to minimise fuel slosh. Often a cent tank traverses the fuselage between the two inner wing root ribs. All fuel



Attachment Rings and Moulded Connections
Figure 3

tanks are fuel tight. Close metal-to-metal fit of all parts forms the basic seal, with sealing compounds and sealing fasteners on all joints to complete the fluid tight seal. The centre tank will have a secondary external barrier coating to prevent fuel vapour entering the pressurised section of the fuselage. Some of the wing ribs contain a series of free-swinging, fuel-actuated baffle check valves, to prevent fuel flow away from the electric boost pumps. Access panels, usually on the underside of the wing, provide access to each tank. The outer portion of the wing provides fuel overflow by means of a surge tank, which also affords venting into the system. The fuel tanks hold all the necessary equipment for refuelling/ de-fuelling and engine fuel feed. Equipment used for fuel quantity indicating is also contained within the fuel tank structure.

Water Draining

Water drain valves are provided at low points of each tank. All valves may be opened with standard tools and the outer seal of the valve is replaceable without emptying the tanks.

SUPPLY SYSTEMS ENGINE FUEL FEED

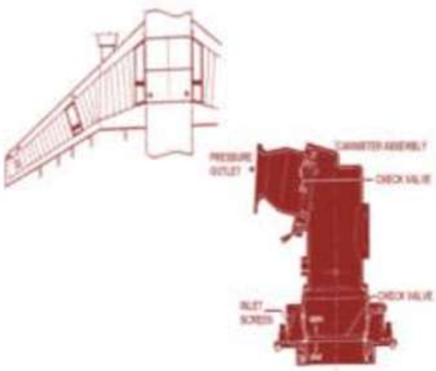
Design Requirements of an Aircraft Fuel Feed System On an aircraft, a fuel system should be designed to comply with many requirements as laid down in Joint Airworthiness Requirements. An example of these requirements is as follows:

1. Each fuel system should be constructed and arranged to ensure a flow of fuel at a rate and pressure to ensure proper functioning of the engine for each likely operating condition.
2. The fuel system must allow the supply of fuel to each engine through a system independent of the system supplying fuel to any other engine.
3. The system design should be such that it is not possible for any pump to draw fuel from two or more tank simultaneously unless means are provided to prevent the introduction of air into the system.

4. If fuel can be pumped from one tank to another in flight, the fuel tank vents and transfer system must be designed so that no structural failure can occur because of over-filling.
5. Integral tanks must have facilities for interior inspection and repair.
6. Fuel tanks must be designed, located and installed so that no fuel is released in or near the engines in sufficient quantities to start a fire in otherwise survivable crash conditions.
7. Pressure cross-feed lines passing through crew, passenger or cargo compartments shall either be enclosed in a fuel and vapour proof enclosure,
ventilated and drained to the outside, OR consist of a pipe without fittings and routed or protected against accidental damage.
8. The system shall incorporate means to prevent the collection of water and dirt or the deposition of ice or other substances from satisfactory functioning of the system.
9. Lines, which can be isolated from the system by means of valves or fuel cocks, shall incorporate provision for the relief of excess pressure due to expansion of the fuel.
10. Each fuel tank filler connection must be marked with type of fuel and be provided with a bonding point and drain discharging excess fuel.
11. There must be a fuel strainer at each fuel tank outlet or for the booster pump(s).
12. Each fuel line must be designed, installed and supported to prevent excessive vibration and allow a reasonable degree of deformation and stretching without leakage.

Engine Fuel Feed (Fuel Cells)
Another multi tank system is the use of fire cells. In normal
conditions, each engine is supplied from one pump in the optional centre tank or both pumps in the tank of its own wing. Any one pump can supply the maximum demand of one engine. A cross-feed pipe, controlled by a double motor actuated spherical plug valve, allows both engines to be fed from one side or all the fuel to be used by one engine. The valve is mounted on the rear spar in the centre section.

10.5.2 Engine Fuel Feed (Multi Tank and Booster Pumps)

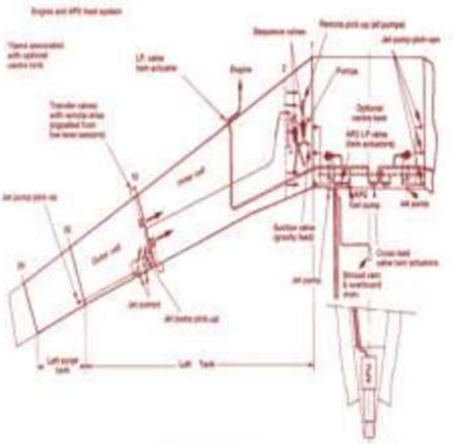


Two plug-ina.c. driven booster pumps supplied from different busbars are fitted in each tank. Each pump has a suction inlet. On each side, the two pumps in the wing tank and one pump in the centre tank (when fitted) deliver fuel via a built in non-return valve into a single pipe. The pumps in the wing tanks are fitted with pressure relief sequence valves that ensure that when all pumps are running, the centre tank pumps will deliver fuel preferentially. No sequence valves are provided on a two tank version aircraft.
In each wing tank the pumps are located in a collector box. The box is fed by gravity through flap non-return valves. This ensures that the system can continue to supply fuel under negative „g" or transient manoeuvres. A bypass is provided at the pumps to permit gravity feed. Air release valves are fitted to the feed lines.

The supply of fuel to each engine can be shut off by an engine LP valve mounted on the front spar. This is a spherical plug valve driven by a double motor actuator. To provide the maximum integrity, the two actuators are supplied from different busbars and the cables are routed separately. Controls and indications for pumps and crossfeed valves of the feed system are located on the overhead panel. In normal operation, all wing pumps will remain on throughout the flight. If a centre tank is fitted, switching of pumps is automatic. If there are no malfunctions, no action is required during flight. The engine LP valves are controlled by operating the engine fire handles.

FUEL FEED COMPONENTS

Fuel Pumps (Booster Pumps) Pumps employed in aircraft fuel systems differ in size, shape, output, etc. However regardless of type and any special features they may have, they all operate on the same principle and consist of very similar components. Each tank is normally provided with two fuel pumps. They are all identical and interchangeable. These pumps are installed in the canister assemblies to enable replacement without de-fuelling the tank. The fuel pumps are centrifugal pumps driven by 115 volts, three phase motors. The output of each pump is about 250-300 litres per minute. Maximum fuel pressure at zero flow is about 38



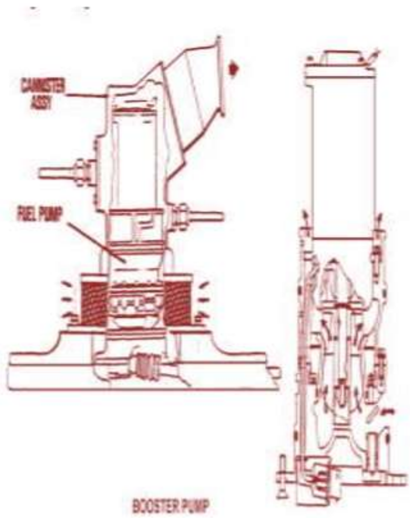
Engine Fuel Feed

p.s.i. Each pump includes a non-return and a by-pass valve. The by-pass valve is to reduce the pressure drop allowing an engine to be operated on suction feed up to about 6000 feet.
They are protected by a thermal fuse, which is activated at
approximately 175 degrees centigrade.
Some pumps have special features that are dictated by the aircraft role and any design requirements namely:

a. Pressure relief valve.
b. Non-return valve.
c. AC DC motor.
d. Thermal trip devices.
e. Cannister shut off valve to facilitate pump replacement with fuel in the tanks.

Jet Pumps

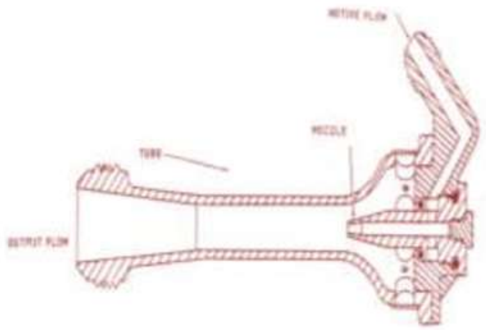
These are another method of transferring fuel around an aircraft fuel system. They use fuel bled from the booster pump which is continually fed through the central nozzle into a
venturi. The depression created in the venturi draws fuel from the surrounding tank, in through the filter then up through the venturi tube and either into
the next fuel tank or straight to the collector box.



BOOSTER PUMP

Sequence Valves

Sequence valves are fitted to give an automatic transfer from one tank to another,the following example is for an aircraft with pumps in the centre tank, inner tank and outer tank. The valve limits the fuel pressure of the outer tank pumps from 38 psi to 17.5 psi. This is to give priority to the inner tank fuel pumps for structural reasons. When the inner tanks are empty, the engines will be automatically supplied from the outer tanks So the outer fuel pumps run continuously.



JET PUMP
FIGURE 16

Transfer Valves The example at figure 13 shows the fuel tank split into two cells at rib 15. To enable transfer to take place, two transfer valves are fitted in this instance at rib 15. Operation of these valves is actuated by a signal from low level sensors shown just inboard of rib 2.

10.6.6 Cross Feed Valve

Cross-feed Valve

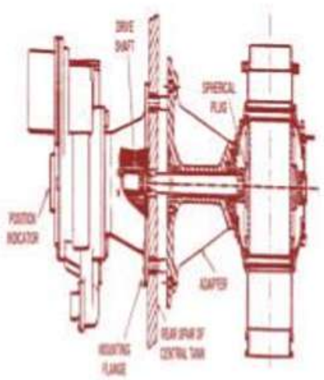


Figure 20

Cross Feed Valve

The cross feed valve enables fuel to be fed to any engine from any tank. Normally of a spherical type construction with two 28 VDC electric motors mounted on a differential gearbox. One motor only will drive the valve at any time, the other motor is a back up. The cross feed valve would normally be fitted on the rear spar as shown in the figure.

APU FUEL FEED

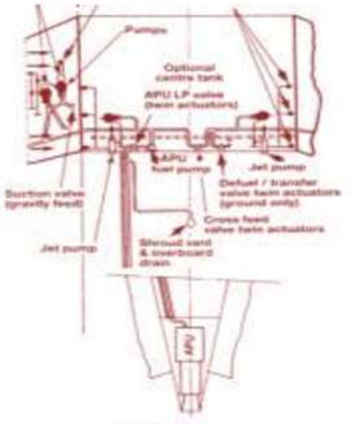
The feed to the APU is taken from the left engine feed but may be taken from the right engine feed when the cross feed valve is open. The tank booster pumps can supply fuel to the APU at the required pressure. For starting the APU without electrical power available for the tank pumps, a separate pump is provided that can be operated from the aircraft batteries and is mounted in the feed line on the rear spar of the centre section. The supply of fuel to the APU can be shut off by a valve mounted on the rear spar of the centre section. It is a spherical plug valve driven by a double

motor actuator. To provide the maximum integrity, the two actuators are supplied from different busbars and the cables are located in separated routes. The feed pipe emerges from the top of the tank and passes through the pressurised fuselage in a drained and vented shroud that extends to the APU fire wall.

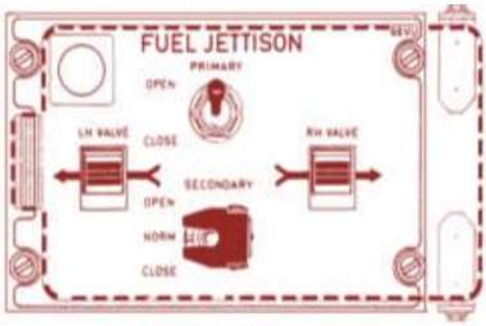
DUMPING, VENTING AND DRAINING DUMPING (JETTISON)

Fuel jettison systems are fitted to a number of large commercial aircraft to allow the jettisoning of fuel in an emergency thus reducing weight so as to prevent structural damage when landing. Fuel jettison systems are often fitted after the installation of a centre tank, because of the extra fuel weight. The system illustrated is from a wide-bodied twin fitted with multi tanks and booster pumps. The jettison pipe is branched off the feed pipe between the inner tank fuel pump and the inner tank shut off valve. A check valve is installed to separate the outer tanks during jettisoning. The function of this check valve is to prevent the dumping of the outer tanks fuel. The jettison pipe runs inside the wing tanks through the ribs into the outer tanks, where the jettison valves are installed. These valves are fitted to the bottom of the tank. Because of electrical emergency situations, the valve will be driven by two 28 VDC electric motors. The motors are mounted from the outside and are attached to the bottom of the tank through a gearbox and in many instances are interchangeable with the cross feed valves. The jettison operation is controlled from a jettison panel located either on a flight engineers station or from an overhead panel on a two crew configuration. Normally the panel is protected by a quick release cover. In the following example, two switches are provided to operate the jettison valve.

i. A primary switch for motor number one.
ii. A guarded secondary switch for motor number two.

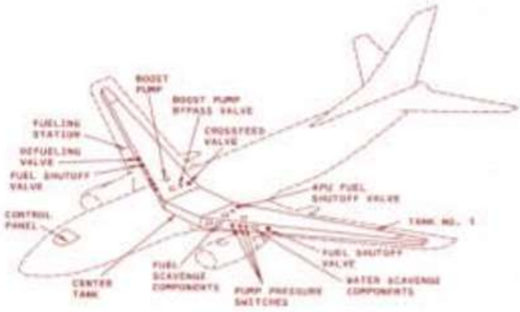


APU Fuel feed
Figure 21



Fuel Jettison Control
Figure 25

The position of the right and left-hand jettison valve is monitored by two magnetic indicators, showing green cross-line when the valve is closed and in-line when the valve is open. As is common with this type of indicator, it will show amber cross-line to indicate transit or malfunction.



Cross Feed Control Panel
Figure 31

CROSS-FEED AND TRANSFER

Cross-feed Valves permit the transfer of fuel from any tank to any engine, whereas Transfer Valves enable fuel to be transferred from tank to tank.

Auto – transfer When an aircraft has a wing with lateral dihedral the fuel pumps will normally be inboard and the fuel flow towards the wing root. When the wing contains more than one tank, the outboard tank will automatically transfer into the inboard tank and so be the first to empty. Since the inboard tanks will be feeding the engines, a transfer valve between the inboard and outboard tanks will be opened automatically, whenever a high level float switch in the inboard tank detects it being not full.

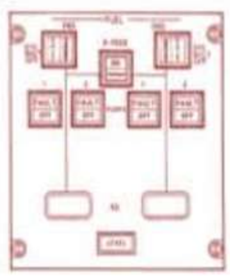
Manual – transfer No in-flight transfer of fuel between left and right mainplanes is possible for reasons of trim. However fuel can be fed from any tank to any engine by means of boost pump selection and the opening of a cross feed valve from the flight deck.

INDICATIONS AND WARNINGS

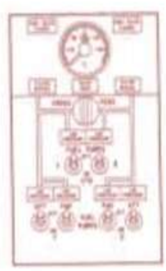
Provision is made to display fuel tank quantity, boost pump low pressure, crossfeed valve and fuel/fire shut off valve position, on the flight deck overhead panel. Though the layout will vary from aircraft type to type, generally it will be similar to the examples shown below.

FUEL LEVEL SENSING

A modern aircraft will use thermistors to send signals through amplifiers to actuate warnings, sequencing, etc. Older aircraft may use float switches as shown in the following diagram.



Overhead panel - Push switch type
Figure 33



Overhead panel - Toggle switch type
Figure 34

Float operated switches are of a magnetic type, similar to the one shown above and are designed to isolate the electrical mechanism from the fuel tank for safety reasons. Upward movement of the float brings the armature closer to the magnet and at a predetermined fuel level, it has sufficient influence to attract the magnet, which results in operation of the micro switch. As the fuel level and the float fall, the attraction of the armature is eventually overcome by the combined forces of the counterweight and the micro switch spring and the counterweight falls, changing the micro switch circuit.
Whether they are float switches or thermistors, their functions are as follows.
1 High level sensing. 2 Overflow sensing. 3 Low level sensing.
4 Under full level sensing.
5 Level sensing for calibration (Fuel Trim only).

High Level Sensing
High level sensing is installed to prevent an overfilling of the fuel tanks. When the fuel washes around the respective sensor, the:

The high level signal from the inner and outer tanks could be used for computation purposes in the fuel quantity computer, when refuelling in AUTO MODE.

Overflow Sensing
If during refuelling the high level shut off system fails, fuel enters the adjacent vent tank and washes around the overflow sensor. This is indicated by the amber FULL light on the refuel panel.

Low Level Sensing

Low level sensing is divided into:

If the outer tank LO LVL sensor is exposed to air, the associated amber LO LVL light comes on. The inner/centre tank low level sensing have only in the AUTO MODE a function (ref. fuel pump control).
Calibration Sensing (Fuel Trim only)
Calibration sensors are installed in centre tanks, inner tanks and trim tank. They give a signal at a predetermined filling level in the trim tank for accuracy test of the fuel quantity indication during refuelling. For the trim tank the calibration sensor switching level is corrected by the stabiliser position.

Under Full Level Sensing
When the fuel quantity drops in either outer tank below a certain level, the maximum flight speed (VMO) becomes reduced in order to protect the wing structure. The sensor signals are sent to the ADC (Air Data Computer).

FUEL QUANTITY SYSTEM MEASUREMENT AND INDICATION
The system has the following tasks:
1 Measuring of the fuel quantity in the tanks.
2 Indicating of the fuel quantity on:
3 Controlling of automatic refuelling.
4 Fuel quantity messaging to the flight management computer. The system comprises:
1 fuel quantity computer. 2 capacitance probes.
3 capacitance index compensator. 4 cadensicon sensor.
5 attitude sensor.
6 THS position detector.
7 associated indicator in the flight compartment.

PRINCIPLE OF CAPACITANCE GAUGING

A capacitor is an electrical device which stores electrical charge. The amount of charge it can hold depends upon three physical properties of the capacitor itself, namely: a. The surface area of the plates. b. The size of the gap between the plates. c. The insulating material (dielectric) between the plates. In a fuel tank ―capacitor stack‖ two of the above are fixed, ie. the area of the plates and the gap between them. The only variable is the dielectric which, in a fuel tank, is either fuel or air or both. The amount of charge held in the capacitor, when the tank is full, will be of a preset value. As the fuel level falls, the dielectric will gradually change to air and the amount of charge stored will reduce. This change in capacitance is sensed by a signal conditioner and the change in fuel level is thus sensed.

FUEL QUANTITY INDICATING SYSTEM

Each tank has installed a group of probes arranged so that a minimum of one probe is immersed at all times, the number of probes will vary from aircraft to aircraft. The following example is from a wide-bodied twin fitted with a fuel trim system. The

REFUELLING AND DE-FUELLING REFUELLING

As you will be aware, as any liquid flows through a pipeline, it will produce Static Electricity. If this static electricity were allowed to discharge in the presence of aviation fuel vapour, an explosion would result, with possible catastrophic results. To therefore minimise the explosion risks, the following guidelines must be followed. Safety Precautions:

correct grade of fuel (Av-gas, Av-tur, Av-tag).e activity kept to a minimum.

connected to the Bowser.

Refuelling a small aircraft is no more complex than filling the family car. One limitation is that on some aircraft it is not possible to fly the aircraft with all the seats occupied with full baggage allowance, when the tanks are full. This means that if the aircraft is to be flown fully loaded, it may be necessary to re-fuel to less than full, to keep the aircraft within its weight limits. As the aircraft become more complex, the refuelling exercise has to be carried out with more care. If the aircraft is small but has say, two tanks in each wing, and the fuel load is to be three quarters full; then it may be the rule for that aircraft that the inner tanks have to be filled to the top first and the remainder put into the outer tanks. This puts less bending load on to the wing spars.

When we get to larger aircraft, there are several further problems to consider. Not only must the aircraft be filled laterally in the correct order but, if the aircraft has the fin, tailplane and rear fuselage tanks mentioned earlier, it must be refuelled in the correct order longitudinally as well to ensure the aircraft stability is maintained.

DEFUELLING

Defuelling a pressure type fuel system is almost the reverse of the refuelling procedure. A de-fuel bowser would be connected to the single fuel point coupling, and using a combination of both the bowser"s suction pump and the aircraft"s own fuel supply booster pumps, selected tanks can have their contents returned to the bowser.

LONGITUDINAL BALANCE FUEL SYSTEMS

The weight of the fuel is a large percentage of an aircraft"s total weight, and the balance of the aircraft in flight changes as the fuel is used. These conditions add to the complexity of the design of an aircraft fuel system. In small aircraft the fuel tank or tanks are located near the centre of gravity so the balance changes very little as the fuel is used. In large aircraft, fuel tanks are installed in every available location and fuel valves allow the flight engineer to keep the aircraft balanced by scheduling the use of the fuel from the various tanks. High performance military jets and more modern civil aircraft will use a fully automatic fuel scheduling system to reduce the workload on the flight crew.

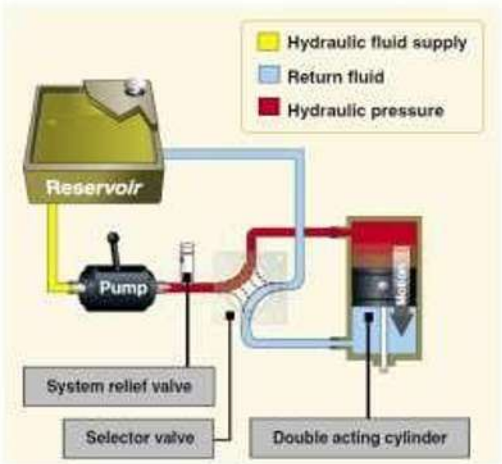
13.14 Hydraulic Power (ATA 29)

System layout:

There are multiple applications for hydraulic use in aircraft, depending on the complexity of the aircraft. For example, hydraulics is often used on small airplanes to operate wheel brakes, retractable landing gear, and some constant-speed propellers. On large airplanes, hydraulics is used for flight control surfaces, wing flaps, spoilers, and other systems. A basic hydraulic system consists of a reservoir, pump (either hand, electric, or engine driven), a filter to keep the fluid clean, selector valve to control the direction of flow, relief valve to relieve excess pressure, and an actuator.

Basic Hydraulic System

The hydraulic fluid is pumped through the system to an actuator or servo. A servo is a cylinder with a piston inside that turns fluid power into work and creates the power needed to move an aircraft system or flight control. Servos can be either single-acting or double- acting, based on the needs of the system. This means that the fluid can be applied to one or both sides of the servo, depending on the servo type. A single-acting servo provides power in one direction. The selector valve allows the fluid direction to be controlled. This is necessary for operations such as the extension and retraction of landing gear during which the fluid must work in two different directions. The



relief valve provides an outlet for the system in the event of excessive fluid pressure in the system. Each system incorporates different components to meet the individual needs of different aircraft.
A mineral-based hydraulic fluid is the most widely used type for small aircraft. This type of hydraulic fluid, a kerosene-like petroleum product, has good lubricating properties, as well as additives to inhibit foaming and prevent the formation of corrosion. It is chemically stable, has very little viscosity change with temperature, and is dyed for identification. Since several types of hydraulic fluids are commonly used, an aircraft must be serviced with the type specified by the manufacturer. Refer to the AFM/POH or the Maintenance Manual.
Hydraulic systems can be found today in a wide variety of applications, from small assembly processes to integrated steel and paper mill applications. Hydraulics enable the operator to accomplish significant work (lifting heavy loads, turning a shaft, drilling precision holes, etc.) with a minimum investment in mechanical linkage through the application of Pascal's law, which states:
—Pressure applied to a confined fluid at any point is transmitted undiminished throughout the fluid in all directions and acts upon every part of the confining vessel at right angles to its interior surfaces and equally

upon equal areas‖

Hydraulic System Components

The major components that make up a hydraulic system are the reservoir, pump, valve(s) and actuator(s) (motor, cylinder, etc.).

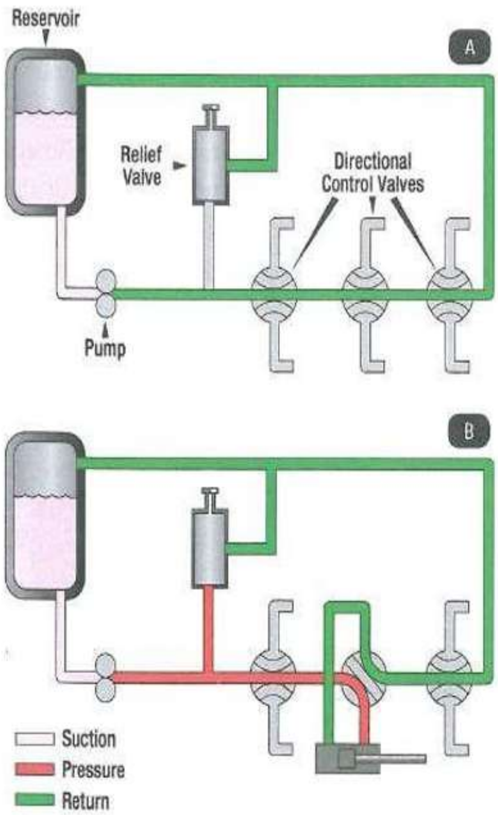
Reservoir

The purpose of the hydraulic reservoir is to hold a volume of fluid, transfer heat from the system, allow solid contaminants to settle and facilitate the release of air and moisture from the fluid.

Pump

The hydraulic pump transmits mechanical energy into hydraulic energy. This is done by the movement of fluid which is the transmission medium. There are several types of hydraulic pumps including gear, vane and piston. All of these pumps have different subtypes intended for specific applications such as a bent-axis piston pump or a variable displacement vane pump. All hydraulic pumps work on the same principle, which is to displace fluid volume against a resistant load or pressure.

Valves



Hydraulic valves are used in a system to start, stop and direct fluid flow. Hydraulic valves are made up of poppets or spools and can be actuated by means of pneumatic, hydraulic, electrical, manual or mechanical means.

Actuators

Hydraulic actuators are the end result of Pascal's law. This is where the hydraulic energy is converted back to mechanical energy. This can be done through use of a hydraulic cylinder which converts hydraulic energy into linear motion and work, or a hydraulic motor which converts hydraulic energy into rotary motion and work. As with hydraulic pumps, hydraulic cylinders and hydraulic motors have several different subtypes, each intended for specific design applications.

OPEN-CENTER HYDRAULIC SYSTEMS

An open-center system is one having fluid flow, but no pressure in the system when the actuating mechanisms are idle. The pump circulates the fluid from the reservoir, through the selector valves, and back to the reservoir. The open-center system may employ any number of subsystems, with a selector valve for each subsystem. Unlike the closed center system, the selector valves of the open center system are always connected in series with each other. In this arrangement, the system pressure line goes through each selector valve. Fluid is always allowed free passage through each selector valve and back to the reservoir until one of the selector valves is positioned to operate a mechanism. When one of the selector valves is positioned to operate an actuating device, fluid is directed from the pump through one of the working lines to the actuator. With the selector valve in this position, the flow of fluid through the valve to the reservoir is blocked. The pressure builds up in the system to overcome the resistance and moves the piston of the actuating cylinder; fluid from the opposite end of the actuator returns to the selector valve and flows back to the reservoir. Operation of the system following actuation of the component depends on the type of selector valve being used. Several types of selector valves are used in conjunction with the open center system.

Open-center hydraulic system

CLOSED-CENTER HYDRAULIC SYSTEMS

In the closed-center System, the fluid is under pressure whenever the power pump is operating. The three actuators are arranged in parallel and actuating units B and C are operating at the same time, while actuating unit A is not operating. This system differs from the open-center System
in that the selector or directional control valves are arranged in parallel and not in series. The means of controlling pump pressure varies in the closed center system. If a constant delivery pump is used, the system pressure is regulated by a pressure regulator. A relief valve acts as a backup safety device in case the regulator fails. If a variable displacement pump is used, system pressure is controlled by the pump's integral pressure mechanism compensator. The compensator automatically varies the volume output. An advantage of the open-center system over the closed- center system is that the continuous pressurization of the system is eliminated. Closed-center systems are the most widely used.

Closed-center Hydraulic system

HYDRAULIC POWER PACK SYSTEM

A hydraulic power pack is a compact unit that consists of an electric pump, a reservoir, valves, filters, and pressure relief valve all in one assembly. The advantage of the power pack is that there is no need for a centralized hydraulic power supply system and long stretches of hydraulic lines. This reduces weight. Power packs are driven by either an engine gearbox or electric motor. These systems are used to control the stabilizer trim, landing gear, or flight control surfaces directly, thus eliminating the need for a centralized hydraulic system.

Hydraulic power pack

MODERN HIGH-PERFORMANCE SYSTEMS

Many modern aircraft use a power supply system and fly-by-wire flight control. The pilot input is electronically sent to the flight control servos. The servos use hydraulic pressure to move the control surfaces. Cables or push rods are not used. Some manufacturers are reducing the use of hydraulic systems in their aircraft in favor of electrically controlled systems. The Boeing 787 is the first aircraft designed with more electrical systems than hydraulic systems. Large aircraft hydraulic systems are complexed close-center systems with a wide variety of components and typically

HYDRAULIC FLUID

Hydraulic system liquids are used primarily to transmit and distribute forces to various units to be actuated. Liquids are able to do this because they are almost incompressible. Pascal's Law States that pressure applied to any part of a confined liquid is transmitted with undiminished intensity to every other part. Thus, if a number of passages exist in a system, pressure can be distributed through all of them by means of the liquid. Manufacturers of hydraulic devices usually specify the type of liquid best suited for use with their equipment in view of the working conditions, the service required, temperatures expected inside and outside the systems, pressures the liquid must withstand, the possibilities of corrosion, and other conditions that must be considered. If incompressibility and fluidity were the only qualities required, any liquid that is not too thick could be used in a hydraulic system. But a satisfactory liquid for a particular installation must possess a number of other properties.

TYPES OF HYDRAULIC FLUIDS

To assure proper system operation and to avoid damage to nonmetallic components of the hydraulic system, the correct fluid must be used. When adding fluid to a system, use the type specified in the aircraft manufacturer's maintenance manual or on the instruction plate affixed to the reservoir or unit being serviced. The three principal categories of hydraulic fluids are:

l. Minerals: Mineral oil-based hydraulic fluid (MIL-H-5606) is the oldest, dating back to the 1940s. It is used in many systems, especially where the fire hazard is comparatively low.

2. Polyalphaolefins: MIL-H-83282 is a fire- resistant hydrogenated polyalphaolefin-based fluid developed in the 1960s to overcome the flammability characteristics of IMIL-H-5606. MIL-H-83282 is significantly more flame resistant than MIL-H- 5606, but a disadvantage is the high viscosity at low temperature. 3.Phosphate esters: These fluids are used in most commercial transport category aircraft and are extremely fire-resistant. However, they are not fireproof and under certain conditions, they burn. When servicing a hydraulic system, the technician must be certain to use the correct category of replacement fluid. Hydraulic fluids are not necessarily compatible. For example, contamination of the fire-resistant fluid MIL-H-83282 with MIL-H-5606 may render the MIL-H-83282 non fire-resistant.

HEALTH AND HANDLING

Skydrol fluids are phosphate ester-based fluids blended with performance additives. Phosphate esters are good solvents and dissolve away some of the fatty materials of the skin. Repeated or prolonged exposure may cause drying of the skin, which if unattended, could result in complications, such as dermatitis or even secondary infection from bacteria. Skydrol fluids could cause itching of the skin but have not been known to cause allergic type skin rashes. Always use the proper gloves and eye protection when handling any type of hydraulic fluid. When Skydrol/Hyjet mist or vapor exposure is possible, a respirator capable of removing organic vapors and mists must be worn. Ingestion of any hydraulic fluid should be avoided. Although small amounts do not appear to be highly hazardous, any significant amount should be tested in accordance with manufacturer's direction, followed with hospital supervised stomach treatment.

HYDRAULIC RESERVOIRS AND ACCUMULATORS RESERVOIRS

The reservoir is a tank in which an adequate supply of fluid for the system is stored. Fluid flows from the reservoir to the pump, where it is forced through the system and eventually returned to the reservoir. The reservoir not only supplies the operating needs of the system, but it also replenishes fluid lost through leakage. Furthermore, the reservoir serves as an overflow basin for excess fluid forced out of the system by thermal expansion (the increase of fluid volume caused by temperature changes), the accumulators, and by piston and rod displacement. The reservoir also furnishes a place for the fluid to purge itself of air bubbles that may enter the system. Foreign matter picked up in the system may also be separated from the fluid in the reservoir or as it flows through line filters. Reservoirs are either pressurized or non-pressurized. Baffles and/or fins are incorporated in most reservoirs to keep the fluid within the reservoir from having random movement, such as vortexing (swirling) and surging. These conditions can cause fluid to foam and air to enter the pump along with the fluid. Many reservoirs incorporate strainers in the filler neck to prevent the entry of foreign matter during servicing

NON-PRESSURIZED RESERVOIRS

Non-pressurized reservoirs are used in aircraft that are not designed for violent maneuvers, do not fly at high altitudes, or in which the reservoir is located in the pressurized area of the aircraft. High altitude in this situation means an altitude where atmospheric pressure is inadequate to maintain sufficient flow of fluid to the hydraulic pumps. Most non-pressurized reservoirs are constructed in a cylindrical shape.

PRESSURIZED RESERVOIRS

Reservoirs on aircraft designed for high-altitude flight are usually pressurized. Pressurizing assures a positive flow of fluid to the pump at high altitudes when low atmospheric pressures are encountered. On some aircraft, the reservoir is pressurized by bleed air taken from the compressor section of the engine. On others, the reservoir may be pressurized by
hydraulic system pressure.

ACCUMULATORS

The accumulator is a steel sphere divided into two chambers by a synthetic rubber diaphragm. The upper chamber contains fluid at system pressure, while the lower chamber is charged with nitrogen or air. Cylindrical types are also used in high-pressure hydraulic systems. Many aircraft have several accumulators in the hydraulic system. There may be a main system accumulator and an emergency system accumulator. There may also be auxiliary accumulators located in various sub-systems.

The functions of an accumulator are to:

l. Dampen pressure surges in the hydraulic system caused by actuation of a unit and the effort of the pump to maintain pressure at a preset level.

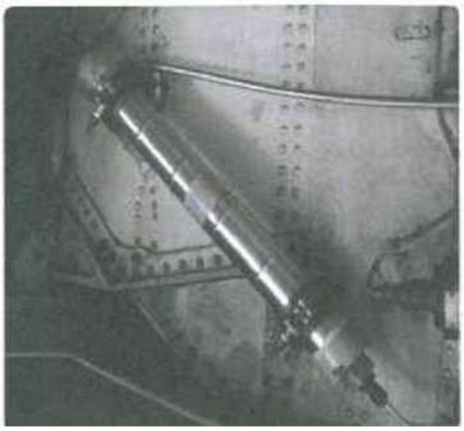
2. Aid or supplement the power pump when several units are operating at once by supplying extra power from its accumulated, or stored, power.

3. Store power for the limited operation of a hydraulic unit when the pump is not operating.

4. Supply fluid under pressure to compensate for small internal or external (not desired) leaks that would cause the system to cycle
continuously by action of the pressure switches continually kicking in.

There are two general types of accumulators used in aircraft hydraulic systems:

l. Spherical: It is constructed in two halves that are fastened and threaded, or welded, together. Two threaded openings exist. The top port accepts fittings to connect to the pressurized hydraulic system to the accumulator. bottom port is fitted with a gas servicing valve. A synthetic rubber diaphragm, or bladder, is



installed in the sphere to create two chambers. Pressurized hydraulic fluid occupies the upper chamber and nitrogen or air charges the lower chamber.

2. Cylindrical: consist of a cylinder and piston assembly. End caps are attached to both ends of the cylinder. The internal piston separates the fluid and air/nitrogen chambers. The end caps and piston are sealed with gaskets and packing to prevent external
leakage around the end caps and internal leakage between the chambers.

Maintenance consists of inspections, minor repairs, replacement of component parts, and testing. There is an element of danger in maintaining accumulators. Therefore, proper precautions must be strictly observed to prevent injury and damage

HYDRAULIC PRESSURE GENERATION

All aircraft hydraulic systems have at least one power driven pump and may include a hand pump as an additional unit when the power-driven pump is inoperative. The pump is the source of fluid flow, which when restricted, generates pressure in the hydraulic pneumatic air.

HYDRAULIC MOTORS

This pump-motor combination is used to provide a transfer of power between a driving element and a driven element. Some applications for which hydraulic transmissions may be used are speed reducers, variable speed drives, constant speed or constant torque drives, and torque converters.

Some advantages of hydraulic transmission of power over mechanical transmission of power are as follows:

1) Quick, easy speed adjustment over a wide range while the power source is operating at a constant (most efficient) speed
2) Rapid, smooth acceleration or deceleration
3) Control over maximum torque and power
4) Cushioning effect to reduce shock loads motor reversal of motion

Just as a rotating shaft drives a hydraulic pump workings to move fluid, fluid forced through the pump can rotate the attached shaft. This is the principle behind a hydraulic motor. Hydraulic fluid forced through the pump rotates the shaft Of the pump, which as a result, makes the pump a motor. Tie motion of the shaft is then used to drive something to which it is attached. Piston-type motors are the most commonly used in hydraulic systems. They are basically the same as hydraulic pumps except they are used to convert hydraulic energy into mechanical (rotary) energy. Hydraulic motors are either of the axial inline or bent axis type. The most commonly used hydraulic motor is the fixed displacement bent axis type. These types of motors are used for the activation Bent axis piston motor of trailing edge flaps, leading edge slats, and stabilizer trim. Some equipment uses a variable-displacement piston motor where very wide speed ranges are desired. Although some piston-type hydraulic motors are controlled by directional control valves, they

POWER TRANSFER UNITS (PTUs)

Hydraulic motors are also used in power transfer units (PTUs). In a PTU, two units, a hydraulic pump and hydraulic motor, are connected via a single drive shaft so that power can be transferred between two hydraulic systems. Depending on the direction of power transfer, each unit works as either a motor or a pump. The pressurized hydraulic system forces fluid through the motor which turns the shaft of the pump that moves fluid through the second hydraulic system. Thus, power is transferred from one system to the other while the PTU transfers power, it does not transfer any fluid from one system to the other. Different types of PTUs are in use; some can only transfer power in one direction while others can transfer power both ways. Some PTUs have a fixed displacement, while others use a variable displacement hydraulic pump. Regardless, the application of PTU's in aircraft allow component operation in a hydraulic system in which the pump has failed. The system with a working pump drives the motor of the PTU so that the pump shaft rotates in the system with the failed pump. Activation can be manual or automatic depending on the aircraft. In an automatically activated setup, a pressure switch is used to detect pump failure which open a valve to divert fluid from the healthy system to the PTU.

HYDRAULIC MOTOR DRIVEN GENERATORS (HMDGS)

In case of an electrical failure, a hydraulic motor driven generator can be employed. An HMDG provides an alternative source of electrical power. The servo controlled variable displacement HMDG is an AC generator driven by the hydraulic motor portion of the unit. Elbe generator part is typically designed to maintain the desired system output frequency of 400 Hz. Thus, an aircraft with an HMDG can maintain electrical power should a primary generator fail through the use of the hydraulic system. Conversely, a hydraulic pump failure is backed up by an electrically driven hydraulic pump.

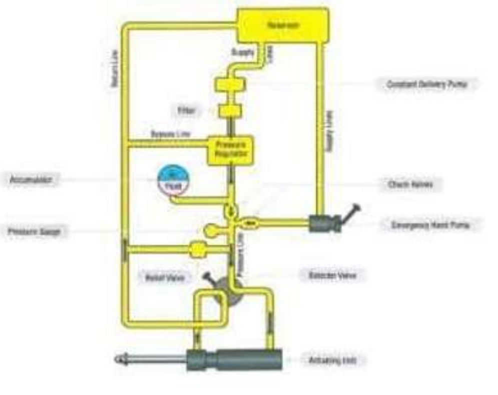
PRESSURE CONTROL

The safe and efficient operation of fluid power systems, system components, and related equipment requires a means of controlling pressure. There are many types of automatic pressure control valves designed for this purpose. Some of them are an escape for pressure that exceeds a set pressure; some only reduce the pressure to a lower pressure system or subsystem; and some keep the pressure in a system within a required range.

RELIEF VALVES

Hydraulic pressure must be regulated in order to use it to perform the desired tasks. A pressure relief valve is used to limit the amount of pressure being exerted on a confined liquid. This is necessary to prevent failure of components or rupture of hydraulic lines under excessive pressures. The pressure relief valve is, in effect, a system safety valve. The design of pressure relief valves incorporates adjustable spring loaded valves. They are installed in such a manner as to discharge fluid from the pressure line into

areservoir return line when the pressure exceeds the predetermined maximum for which the valve is adjusted. Various makes and designs of pressure relief valves are in use, but, in general, they all employ a spring-loaded valving device operated by hydraulic pressure and spring tension. Pressure relief valves are adjusted by increasing or decreasing the tension on the spring to determine the pressure
required to open the valve. They may be classified by type of construction or uses in the system.
The most common types of valve are:
1. Ball type - in pressure relief valves with a ball-type valving device, the ball rests on a contoured seat. Pressure acting on the bottom of the ball pushes it off its seat, allowing the fluid to bypass.
2. Sleeve type - in pressure relief valves with a sleeve type waiving device, the ball remains stationary and a sleeve-type seat is moved up by the fluid pressure. This allows the fluid to bypass between the ball and the sliding sleeve-type seat.
3. Poppet type - in pressure relief valves with a poppet type valving device, a cone-shaped poppet may have any of several design configurations.
however, it is basically a cone and seat machined at matched angles to prevent leakage. As the pressure rises to its predetermined setting, the poppet is lifted off its seat, as in the ball-type device. This allows the fluid to pass through the opening created and out the return port. Pressure relief valves cannot be used as pressure regulators in large hydraulic systems that depend on engine driven pumps for the primary source of pressure because the pump is
constantly under load and the energy expended in
holding the pressure relief valve off its seat is changed into heat. This heat is
transferred to the fluid and, in
turn, to the packing rings, causing them to deteriorate rapidly. Pressure relief valves, however, may be used as pressure regulators in small, low-pressure systems or when the pump is electrically driven and is used intermittently.



Pressure Relief Valve

Pressure relief valves may be used as:

1. System Relief Valve - the most common use of the pressure relief valve is as a safety device against the possible failure of a pump compensator or other pressure regulating device. All hydraulic systems that have hydraulic pumps incorporate pressure relief valves as safety devices.
2. Thermal Relief Valve - the pressure relief valve is used to relieve excessive pressures that may exist due to thermal expansion of the fluid. They are used where a check valve or selector valve prevents pressure from being relieved through the main system relief valve. Thermal relief valves are usually smaller than system relief valves. As pressurized fluid in the line in which it is installed builds to an excessive amount, the valve poppet is forced off its seat. This allows excessive pressurized fluid to flow through the

relief valve to the reservoir return line. When system pressure decreases to a predetermined pressure, spring tension overcomes system pressure and forces the valve poppet to the closed position.

PRESSURE REGULATORS

The term pressure regulator is applied to a device used in hydraulic systems that are pressurized by constant-delivery pumps. One purpose of the pressure regulator is to manage the Output of the pump to maintain system operating pressure within a predetermined range. The other purpose is to permit the pump to turn without resistance (termed unloading the pump) at times when pressure in the system is within normal operating range. The pressure regulator is located in the system so that pump output can get into the system pressure circuit only by passing through the regulator. The combination of a constant-delivery pump and the pressure regulator is virtually the equivalent of a compensator controlled, variable-delivery-type pump.

PRESSURE REDUCERS

Pressure reducing valves are used in hydraulic systems where it is necessary to lower the normal system operating pressure by a specified amount. Pressure reducing valves provide a steady pressure into a system that operates at a lower pressure than the supply system.

A reducing valve can normally be set for any desired downstream pressure within the design limits of the valve. Once the valve is set, the reduced pressure is maintained regardless Of changes in supply pressure (as long as the supply pressure is at least as high as the reduced pressure desired) and regardless of the system load, if the load does not exceed the designed capacity of the reducer.

Valve]

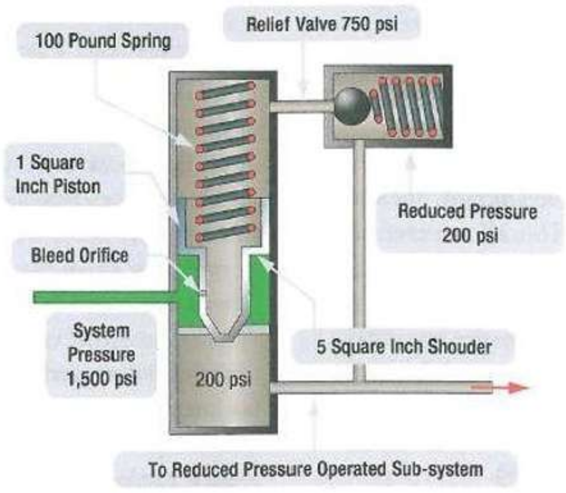
POWER DISTRIBUTION

The location of a pressure regulator in a basic hydraulic system. The regulator unloads the constant delivery pump by bypassing fluid to the return line when the predetermined system pressure is reached.

Operating mechanism of pressure reducing

fluid to a particular System, sub-system or component. In general, these types of valves are electrically powered. They are used to distribute hydraulic power to various components in the system. A shutoff valve may be used to create a priority in a hydraulic system.

Shut-off valve



SELECTOR VALVES

A selector valve is used to control the direction of movement of a hydraulic actuating cylinder or similar device. It provides for the

Power distribution in a hydraulic System is controlled through the use of variety of flow control valves. These valves control the speed and/or direction of fluid flow in the hydraulic system. They provide for the operation of various components when desired and the speed at which the component operates. Examples of flow control valves include:
selector valves, check valves, sequence valves, priority valves, shuttle valves, quick disconnect valves, hydraulic fuses and shutoff valves.

SHUTOFF VALVES

Shutoff valves are used to shutoff the flow of simultaneous flow of hydraulic fluid both into and out of the unit. Hydraulic system pressure can be routed
with the selector valve to operate the unit in either direction and a corresponding return path for the fluid to the reservoir is provided. There are two main types of selector valves: open-center and closed-center. Selector valves may be poppet-type, spool-type, piston-type, rotary type, or plug-type.

A poppet-type four way selector valve CHECK VALVES

Another common flow control valve in aircraft hydraulic systems is the check valve. A check valve allows fluid to flow unimpeded in one direction, but prevents or restricts fluid flow in the opposite direction. A check valve may be an independent component situated in-line somewhere in the hydraulic system or it may be built-in to a component. When part of a component, the check valve is said to be an integral check valve.

SEQUENCE VALVES

Sequence valves control the sequence of operation between two branches in a circuit; they enable one unit to automatically set another unit into motion. An example of the use of a sequence valve is in an aircraft landing gear actuating system. In a landing gear actuating system, the landing gear doors must open before the landing gear starts to extend. Conversely, the landing gear must be completely retracted before the doors close. A sequence valve installed in each landing gear actuating line performs this function. A sequence valve is somewhat similar to a relief valve except that, after the set pressure has been reached, the sequence valve diverts the fluid to a second actuator or motor to do work in another part of the system. There are various types of sequence valves. Some are controlled by pressure, some are controlled mechanically, and some are controlled by electric switches.

PRIORITY VALVES

A priority valve gives priority to the critical hydraulic subsystems over noncritical systems when system pressure is low. For instance, if the pressure of the priority valve is set for 2 200 psi, all systems receive pressure when the pressure is above 2 200 psi. If the pressure drops below 2200 psi, the priority valve closes and no fluid pressure flows to the noncritical systems. Some hydraulic designs use pressure switches and electric shutoff valves to assure that the critical systems have priority over noncritical Systems when system pressure is low.

SHUTTLE VALVES

In certain fluid power systems, the supply of fluid to a subsystem must be from more than one source to meet system requirements. In some systems, an emergency system is provided as a source of pressure in the event of normal system failure. emergency system usually actuates only essential components. The main purpose of the shuttle valve is to isolate the normal system from an alternate or emergency system. It is small and simple; yet, it is a very important component.

QUICK DISCONNECT VALVES

Quick disconnect valves are installed in hydraulic lines to prevent loss of fluid when units are removed. Such valves are installed in the pressure and suction lines of the system immediately upstream and downstream of the power pump. In addition to pump removal, a power pump Can be disconnected from the System and a hydraulic test stand connected in its place.

HYDRAULIC FUSES

A hydraulic fuse is a safety device. Fuses may be installed at strategic locations throughout a hydraulic system. They detect sudden increase in flow, such as a burst downstream, and shut off the fluid flow. By Closing, a fuse preserves hydraulic fluid for the rest of the system. Hydraulic fuses arc fitted to the brake System, leading edge flap and slat extend and retract lines, nose landing gear up and down lines, and the thrust reverser return lines.

OTHER HYDRAULIC SYSTEM COMPONENTS HYDRAULIC ACTUATORS

A hydraulic actuating cylinder transforms energy in the form of fluid pressure into mechanical force, or action, to perform work. It is used to impart powered linear motion to some movable object or
mechanism. A typical actuating cylinder consists of a cylinder housing, one or more pistons and piston rods, and some seals. Actuating cylinders are of two major types: single action and double action. Tie single-

action (single port) actuating cylinder is capable of producing powered movement in one direction only. The double action (two ports) actuating cylinder is capable of producing powered movement in two directions

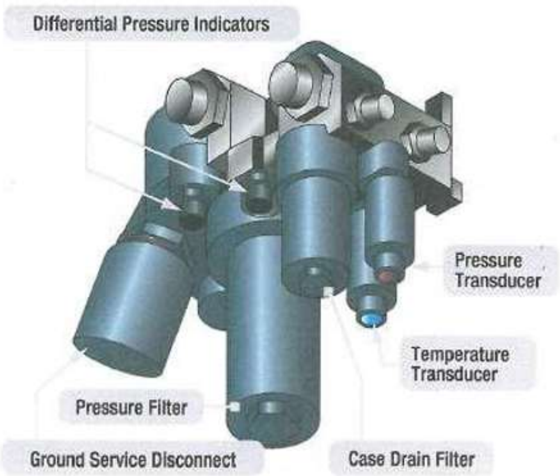
FILTERS

A filter is a screening or straining device used to clean the hydraulic fluid, preventing foreign particles and contaminating substances from remaining in the system. If such objectionable material were not removed, the entire hydraulic system of the aircraft could fail through the breakdown or malfunctioning of a single unit of the System. hydraulic holds in suspension tiny particles of metal that are deposited during the wear of selector valves, pumps, and other system components, Such minute particles of metal may damage the units and parts through which they pass if they are not removed by a filter. Since tolerances within the hydraulic system Components are quite small, it is apparent that the reliability and efficiency of the entire system depends upon adequate filtering. Filters may be located within the reservoir, in the pressure line, in the return line, or in any other location the designer of the system decides that they are needed to safeguard the hydraulic system against impurities. Modern design Often uses a filter that Contains several filters and other components. are many models and styles of filters. Their position in the aircraft and design requirements determine their shape and size.

HYDRAULIC INDICATING AND WARNING

There are just a few hydraulic system indications on the flight deck. Fluid pressure and temperature are the primary parameters monitored as well as fluid quantity. Reservoir pressurization air

pressure may also be monitored. Typically, electro-hydraulic transducers are mounted in the system in key locations so that hydraulic pressure and temperature can be displayed on a gauge or LCD screen. A separate transmitter and indication is used for brake pressure. For servicing and maintenance, direct reading indicators are installed so that maintenance technicians can observe system status while on the ramp. System pressure sensors are often located at the ic system pressure filter modules. Low pressure warning switches arc located of the pump and may also be at the module. A hydraulic panel on



the flight deck incorporates pump switches and temperature and pressure indications in older aircraft. Warnings and system indications are displayed on system status screens away from the switches on class cockpit aircraft. Hydraulic fluid quantity is monitored at the reservoir through the use of a float gauge, sight glass, or other sensing mechanism which sends a signal to the flight deck for gauge or LCD display. A quantity warning switch may be included in the system. Hydraulic system fluid temperature indication is usually limited to an OVERHEAT annunciation for each Temperature switches, often located in the return line the fluid enters the reset trip when a temperature is reached. A signal is sent to the flight deck for annunciation. Temperature sensors for hydraulic systems with electrically driven pumps serve as motor temperature monitoring devices as well. Motor driven pumps are more likely to get hot than engine driven pumps. hot motor transfers some of its heat to the fluid as it circulates.

Filter module
component

Hydraulic system warnings include low pressure annunciations for each hydraulic system. Typically a lamp will illuminate, flash or change color on the flight deck when a pressure sensor sends an electric or electronic signal that a low pressure condition exists. Many indicators display a low pressure warning when the hydraulic pumps are OFF which goes away when the pumps are switched ON and operate normally.

INTERFACE WITH OTHER SYSTEMS

Many aircraft systems use hydraulic power such as landing gear extension and retraction, flight controls, and auto pilot. In most cases, the operational logic for these advanced systems are controlled by computer. to integrate the mechanical power of the hydraulic system, hydraulic system parameters and status condition must be input into the controlling computer. In the absence of any malfunction, the computer controller activates the correct hydraulic system components when needed. Vital systems control logic can also dictate operation in alternate modes should the hydraulic system parameters be out of the normal operating range. For example, if the hydraulic pump used for normal operations is not maintaining acceptable system pressure, logic circuits reconfigure the operational mode from 'normal' to an alternate mode that utilizes the back-up hydraulic pump. Hydraulic system parameters that are captured in analog format are converted to digital format for use in the control system logic.

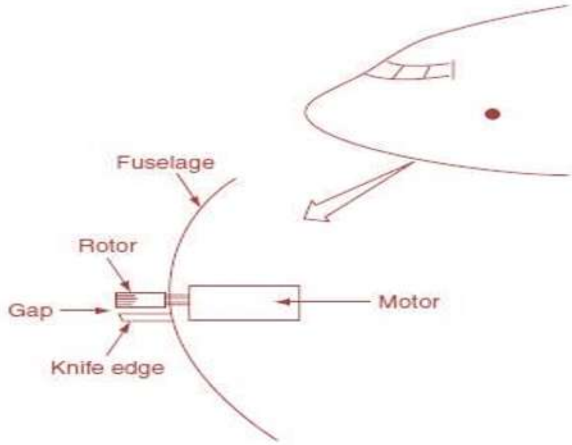
13.15 Ice and Rain Protection (ATA 30)

Airframe ice and rain protection

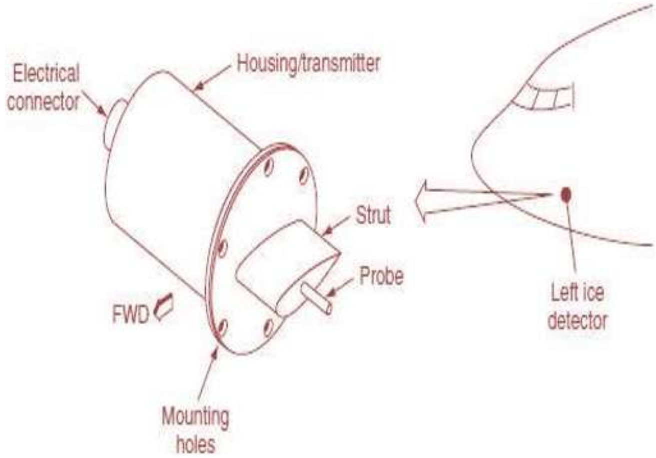
Flying in ice and/or rain conditions poses a number of threats to the safe operation of the aircraft. Ice formation can affect the aerodynamics and/or trim of the aircraft. Icing occurs when super- cooled water makes contact with the airframe and/or ice particles. Super-cooled water exists at temperatures below freezing point; this is because water needs nuclei to form ice crystals. The water freezes on the surface of the nuclei, and then grows in size by forming layers.
Water freezes when energy is given up to its surroundings. Ice accumulation on an aircraft can causes secondary damage by breaking off and being ingested into the engine. Ice and rain also reduces visibility through the windscreen. External sensors and equipment can also be affected by the build-up of ice, e.g. pitot tubes that sense airspeed and angle of attack sensors, and these must not be allowed to freeze.

Ice detection

Various technologies are available to provide a warning to the crew of icing conditions. The simple method is to monitor the
outside air



Temperature and atmospheric conditions.
When flying at night, an ice inspection light can be used to illuminate critical areas, e.g. wing leading edges.
Automatic ice detectors are normally located at the front of the aircraft. Various technologies are employed; the function of the sensor is to detect ice accretion and provide a warning to the crew and/or turn on the ice protection systems.
One type of automatic ice detector consists of a motor-driven sensor that is located against a knife edge cutter, at the front of the fuselage.
As ice builds on the rotor, the gap closes with the knife-edge and torque is applied to the rotor. The body of the motor is held in position by springs, and as the torque increases, the motor starts to rotate in its mounting until a pre-determined point is reached and a micro-switch is closed, thereby operating a warning light.



Analternative technology uses an ultrasonic ice detector. The sensing probe is exposed to the airflow and is vibrated by an electromagnetic coil at a natural frequency of 40 kHz. The probevibrates at a lower frequency when it accumulates ice due to its increased mass.
A logic unit within the detector housing determines when the probe vibration is less than 39,867 Hz; this occurs when a known mass of ice has been accumulated, and the cylinder is heated to melt the ice.
A timing schedule is programmed into the control unit and monitors the time taken for the frequency to change. The heater remains on until the probe vibration returns to its nominal 40 kHz value. Nominal heating time is six seconds; if the heater is on for more than 25 seconds, power is removed, and a fault condition is notified to the crew. The detector provides control functions for selection of both engine and wing ice protection.

Ice Control Systems
Ice affects both engines and airframes and accounts for a large number of aircraft accidents.
Reciprocating-engine-powered aircraft are susceptible to carburetor ice, which shuts off the airflow to the engine. Structural ice forms on the airfoil surfaces and adds weight, as well as disturbing the smooth flow of air needed to produce lift. There are two types of ice control systems: anti-icing systems, which prevent the formation of ice, and deicing systems, which remove ice after it has formed. Both of these systems are discussed here.
A complete ice control system consists of:
- Surface deicers
- Windshield ice control
- Powerplant ice control
- Brake deicers
- Heated pitot heads

Three types of structural ice affect aircraft in flight: rime ice, glaze ice, and frost. Rime ice is a rough, opaque ice that forms when small droplets of water freeze immediately upon striking the aircraft. It builds up slowly, causes a great deal of drag, and deforms the airfoil, increasing the stall speed of the aircraft. Rime ice is relatively easy to break loose with deicer boots.

Glaze ice is the most dangerous ice. It forms on aircraft flying through supercooled water or freezing rain. Glaze ice adds a great amount of weight and is difficult for the boots to break loose.

Three factors must be present for rime or glaze ice to form on an aircraft in night. There must be visible moisture in the air, which can be in the form of rain, drizzle, or clouds. The surface of the aircraft must be below the freezing temperature of water, and the drops of water must be of the appropriate size for the formation of ice.

Ice protection
Two strategies are used for ice protection: de-icing and anti-icing. De-icing allows ice to form and then be removed on a periodic basis. The build-up of ice will have been investigated during type testing of the aircraft and the build-up removed before it poses a hazard. Anti-icing is when ice is not permitted to form at all.
Three primary methods are used for both de-icing and anti-icing:
- fluid
- pneumatic
- thermal.

All three are controlled and operated by electrical systems; an aircraft can be fitted with any one method, or a combination of
all three.
Specific areas to be protected from ice formation are as follows:
Airframe
- leading edges
- control surfaces
- lift augmentation devices
- windscreen (or windshield)

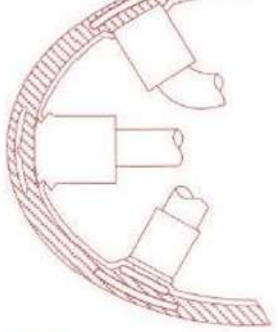
Propulsion
- air intakes
- propellers

External components
- pitot tubes
- temperature sensors
- angle of attack sensors
- water drains.

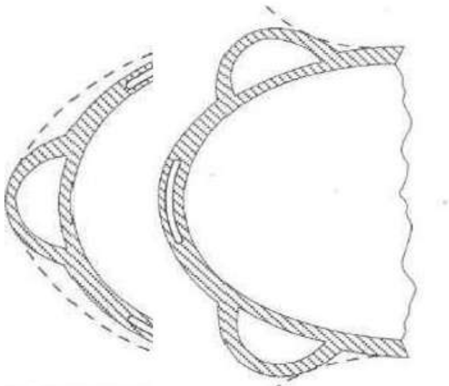
De-icing fluids
Fluids are typically used on wings, vertical stabilizer, horizontal stabilizer, propellers and windscreens. Onboard fluid protection is applied through pipes and/or small holes in the airframe. The fluid is transferred from a storage tank by electrical pumps and directed to the required zones by electrically operated valves.
The fluid can also be applied on the ground by specially equipped vehicles with booms to allow easy access to the entire aircraft. De- icing fluids are typically composed of ethylene glycol or propylene glycol, together with thickening agents, corrosion inhibitors, and colored dye. The aircraft is sprayed with a fluid that melts any existing ice on the aircraft and also prevents ice formation prior to takeoff.
The timing of when the aircraft is sprayed has to take into account the weather conditions (in particular ambient temperature, wind speed, precipitation, and humidity) aircraft skin temperature, and the scheduled departure time.
Fluid performance is characterized by its holdover time: this is the period of time that an aircraft remains protected against ice prior to takeoff, and can be up to 80 minutes depending on conditions and

fluid additives. The colored dye is used so that treated areas can be readily identified.

Pneumatic ice protection
Pneumatic ice protection is used on the leading edge of wings and the vertical/horizontal stabilizers. Fabric reinforced rubber tubes (known as boots) are attached to the surfaces of leading edges of wings, tail planes and fins.



A When system is not operating, suction holds all three tubes deflated and tight against leading edge.



B When systen
tube inflates an
remains inflated
seconds, then deflates.

C Outer two tubes inflate and raise cracked ice from surface so wind can blow it away.

These tubes are inflated and deflated with air on a cyclic basis to cause a slight deformation of the boot; this causes the ice to break off. The air supply is from an electrically driven pump, or bled from the engine compressor via a regulator.
Electrical solenoid valves direct the air into the boots either in sequence or simultaneously. The tubes are deflated and kept flush with the airframe when not in use; this is achieved by connecting the boots to a vacuum source. The system can be operated automatically or manually.

Thermal ice protection

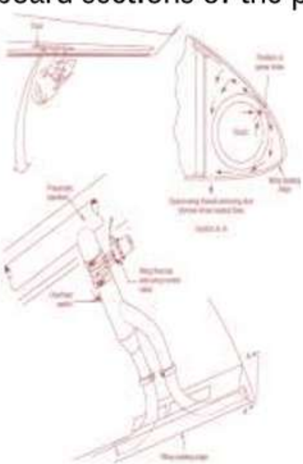
Thermal ice protection can be achieved by one of two methods: hot air bled from the engine compressor,
or electrical heaters. Hot air is directed to the required zone(s) by electrically operated valves controlled from the flight compartment or by a control unit.
Temperature sensors are used as part of a control and feedback system to provide overheats protection.
Applications for bleed air ice protection include the leading edge of wings, vertical/horizontal stabilizers and engine air intakes.

Electrical heating systems are used for both anti-ice and de-icing; heater mats are bonded to the:
- airframe
- engine intakes
- propellers
- rotor blades
- windscreens.

The heater mats are formed from fine-wire elements inside layers of insulation and protective materials. Typical heating elements are nickel, nickel-chrome or copper-nickel. Larger aircraft use 115 V AC variable frequency power supply. Smaller aircraft use a 28 V DC power supply.

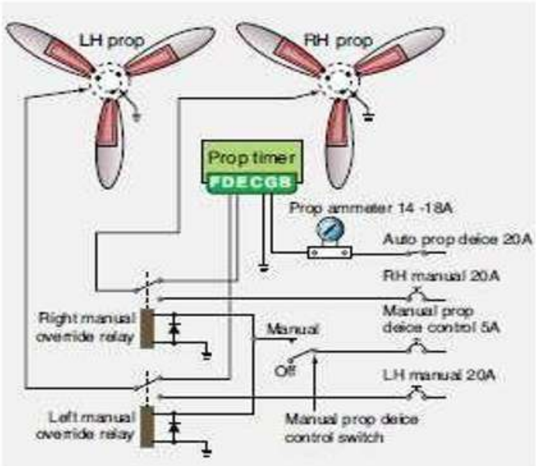
Propeller de-icing is achieved by electrical heating elements that are bonded to the leading edge of the blades. Some aircraft have

heating mats on the inboard and outboard sections of the propeller blade. The elements are connected to



the 28 V DC power supply via slip rings and brushes inside the hub. Typical current
requirements for propeller heating are 15–20 A. Power is applied from a timer control circuit that alternates the heating of outer and inner mats on each of the propeller blades over a 30 second cycle.

Engine intake leading edges are continuously heated when the system



is turned on (anti- icing). This applies to gas turbine engines and turbo-props). The inner and outer surfaces are heated via a cyclic timer to provide de-icing.
Engine intakes are installed with mats that are shaped to provide breaker strips; this ensures that the ice breaks off in pre-determined sections in a controlled sequence.
Larger aircraft use engine bleed air for thermal anti-icing along the leading edge of the wing. Spray holes direct the air inside the leading edge before it is ejected overboard. Similar installations are used for tail plane and engine nacelles

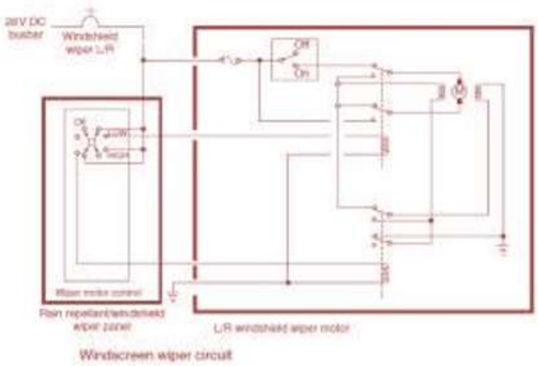
Windscreen ice and rain protection
Various methods of windscreen ice and rain protection are used on a range of aircraft types. One method uses a metallic fi lm deposited onto the surface of the screen; this is connected to an electrical supply on either side of the panel. An alternative technology uses fi ne wire elements sandwiched within the laminated glass panel. Individual windscreen heaters use 4 kW of power, to keep window temperature at approximately 30 ° C. Automatic cycling of power regulates the amount of heat being absorbed by the windscreen. Power is isolated
from the heaters in the event of fault conditions; temperature sensors detect overheat and current transformers detect electrical overload conditions. Temperature sensors are monitored by a control unit; these can be simple thermostats or thermistors.

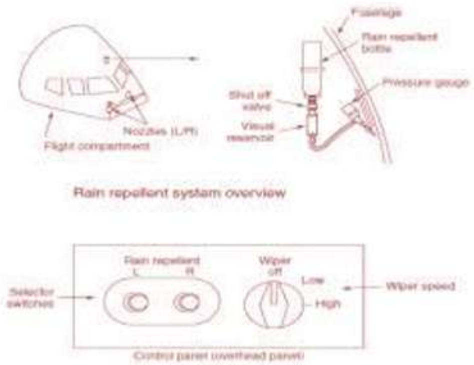
Windscreen wiper
The windscreen wiper system is based on 28 V DC variable speed motors; the rotary motion of the motor is changed by a gear mechanism in the
converter to produce the sweeping motion of the wiper arm over the windscreen. The normal arrangement is to have one wiper assembly per screen to ensure that at least on pilot can keep a clear screen in the event of failure. The motors are set by the control switch; typical wiper speeds are:
- low, 160 cycles per minute
- High, 250 cycles per minute

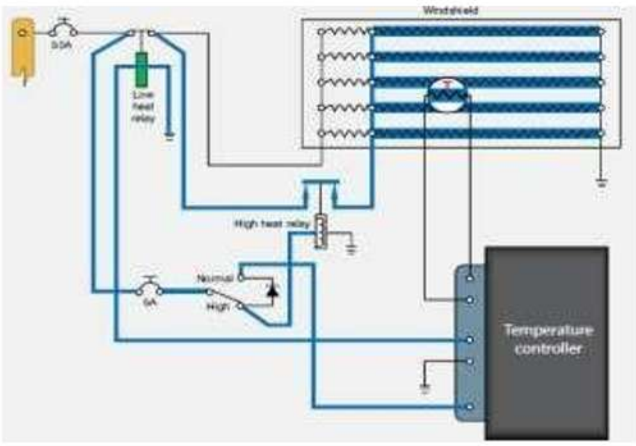
A parking switch in the motor/converter sets the wiper blade to the park position when the wiper is selected off. In the off position, the park switch in each motor/ converter closes; this causes the blades to position them at the bottom of each windscreen.

Rain repellent

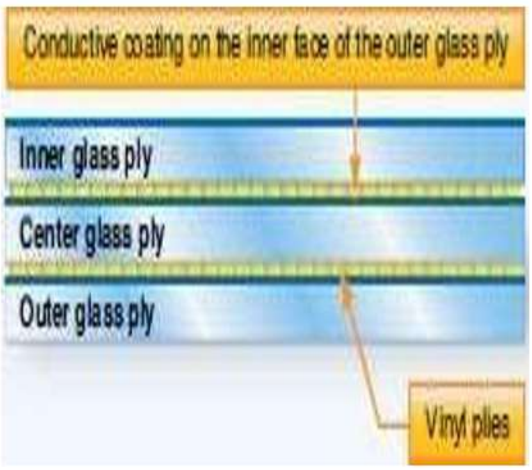
This system is used to maintain a clear area on the windscreen during take-off and landing. The rain repellent bottle is located inside the fuselage roof; this contains a pressure gauge, visual contents reservoir and a shut-off valve. The control panel (normally combined with the wiper system) contains two switches to control the rain repellent system.



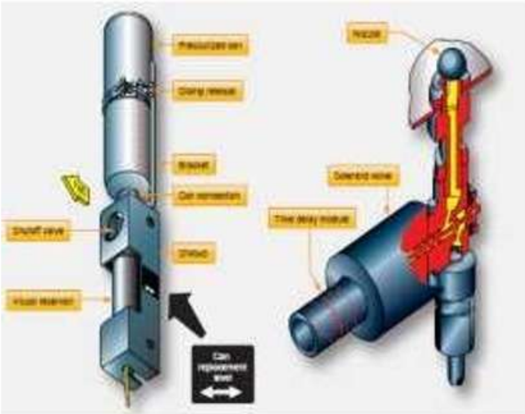




The repellent fluid is stored in a container; this is pressurized from an external air supply. Two electrically operated solenoid valves
control the flow of repellent to the windscreens. Nozzles on the fuselage (forward of the windscreen) direct the repellent spray onto the windscreens.



reduce the wiper blade loading pressure on the window, causing ineffective wiping or streaking. The other is in achieving fast enough wiper oscillation to keep up with high rain impingement rates during heavy rain falls. As a result, most aircraft wiper systems fail to provide satisfactory vision in heavy rain.
The rain removal system shown in Figure
controls windshield icing and removes rain by directing a flow of heated air over the windshield. This heated air serves two purposes. First, the air breaks the rain drops into small particles that are then blown away. Secondly, the air heats the windshield to prevent the moisture from freezing. The air can be supplied by an electric blower or by bleed air.



Windshield Frost, Fog, and Ice Control Systems In order to keep windshield areas free of ice, frost, and fog, window anti-icing, de icing, and defogging systems are used .These can be electric, pneumatic, or chemical depending on the type and complexity of the aircraft. A few of these systems are discussed in this section.

Electric High performance and transport category aircraft windshields are typically made of laminated glass, polycarbonate, or similar ply material. Typically clear vinyl plies are also included to improve performance characteristics. The laminations create the strength and impact resistance of the windshield assembly. These are critical feature for windshields as they are subject to a wide range of temperatures and pressures. They must also withstand the force of a 4 pound bird strike at cruising speed to be
certified. The laminated construction facilitates the inclusion of electric heating elements into the glass layers, which are used to keep the windshield clear of ice, frost, and fog. The elements can be in the form of resistance wires or a transparent conductive material may be used as one of the window plies. To ensure enough heating is applied to the outside of the windshield, heating elements are placed on the inside of the outer glass ply. Windshields are typically bonded together by the application of pressure and heat without the use of cement. Figure illustrates the plies in one transport category aircraft windshield.

Whether resistance wires or a laminated conductive film is used, aircraft window heat systems have transformers to supply power

and feedback mechanisms, such as thermistors, to provide a window heat control unit with information used to keep operating temperature within acceptable limits. Some systems are automatic while others are controlled by cockpit switches. Separate circuits for pilot and co-pilot are common to ensure visibility in case of a malfunction.

Chemical

Chemical anti-ice systems exist generally for small aircraft. This type of anti-ice is also used on windshields. Whether alone or part of a TKSTM system or similar, the liquid chemical is sprayed through a nozzle onto the outside of the windshield which prevents ice from forming. The chemical can also deice the wind shield of ice that may have already formed. Systems such as these have a fluid reservoir, pump, control valve,

filter, and relief valve. Other components may exist. Figure shows a set of spray tubes for application of chemical anti-ice on an aircraft windshield.

Landing Gear Retraction Systems

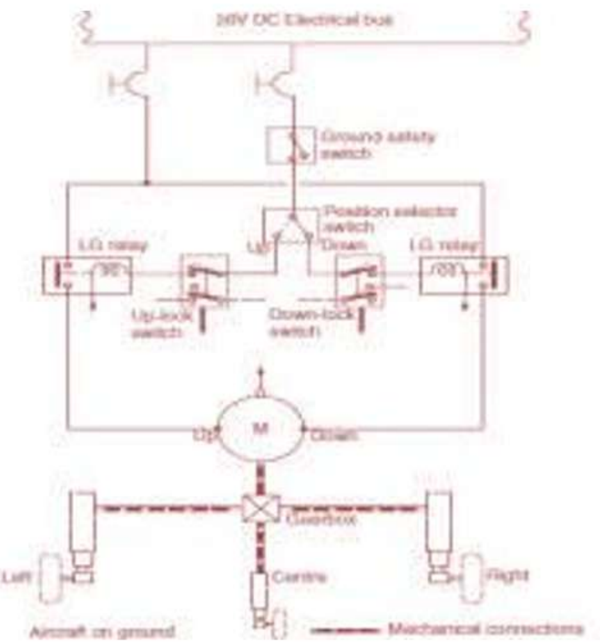
As the speed of aircraft becomes high enough that the parasite drag of the landing gear is greater than the induced drag caused by the added weight of a retracting system, it becomes economically practical to retract the landing gear into the aircraft structure. Small aircraft use simple mechanical retraction systems. Some use a hand crank to drive the retracting mechanism through a roller chain, and the most simple system of all uses a direct hand lever mechanism to raise and lower the wheels.

Many aircraft use electric motors to drive the gear-retracting mechanism, and some European-built aircraft use pneumatic systems. Most American built aircraft use hydraulically retracted landing gears.

Landing gear Indication

Three control and indicating system configurations are found on aircraft with retractable landing gear (or undercarriage):

- hydraulic control and operation, electrical indication of position
- electrical control and operation, electrical indication of position
- electro-hydraulic control and operation, electrical indication of position



The landing gear comprises two or more wheels and shock absorbers, or oleo legs; these can be fixed in position or retractable. Large aircraft have hydraulic systems with electrical position sensors for gear up/down indications. General aviation (GA) aircraft normally have electrical control and operation, and electrical indication of position.

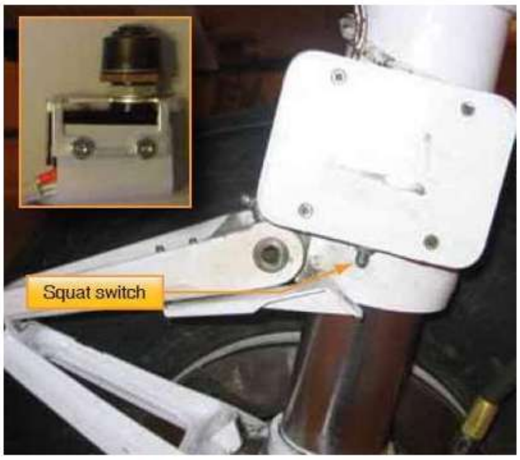
Operation is via a series-wound split field electrical motor driving through to each undercarriage leg via a gearbox, torque tubes, cables and pulleys.

The 28 V DC power supply is connected to the reversible motor via a safety switch and the position selector switch. This switch operates the landing gear relay via up-lock and down-lock micro switches.

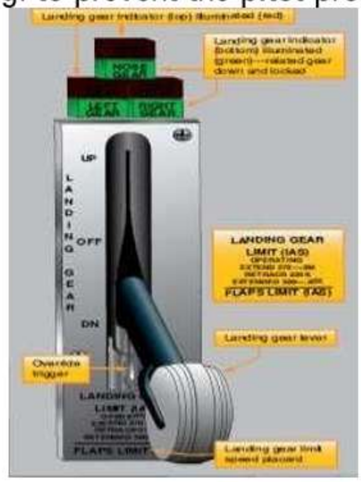
When the landing gear starts to retract, the down-lock switch contacts close. When the landing gear is fully retracted the up-lock switch contacts open, and this removes power from the motor.

When the landing gear is selected down, the up-lock contacts close and the gear can be extended.

The landing gear safety switch senses when the aircraft is on the ground to prevent the landing gear from being retracted. Some aircraft have a micro-switch (or proximity-



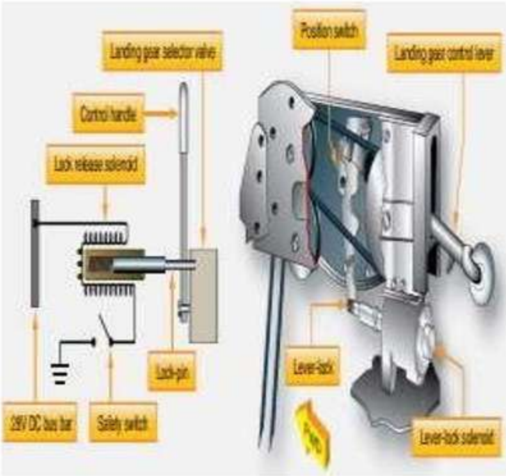
sensor) attached to the oleo leg; when the aircraft is on the ground, the oleo leg is compressed from the weight of the aircraft; this closes the switch contacts. The switch (or sensor) is now being used to detect the weight on wheels (WoW); this is sometimes referred to as a squat switch. When the aircraft takes off, the oleo leg extends when the aircraft weight is transferred to the wings and the WoW, or squat switch, opens its contacts. (Note that the switch can actually be configured as normally open or normally closed on the ground depending on the aircraft design.) WoW or squat switch contacts are also used by other systems that need to be set into air or ground modes, e.g. to prevent the pitot probes and ice detectors from being heated on the ground.



Ground Locks

Ground locks are commonly used on aircraft landing gear as extra insurance that the landing gear will remain down and locked while the aircraft is on the ground. They are external devices that are placed in the retraction mechanism to prevent its movement. A ground lock can be as simple as a pin placed into the pre-drilled holes of gear components that keep the gear from collapsing. Another commonly used ground lock clamps onto the exposed piston of the gear retraction cylinder that prevents it from retracting. All ground locks should have a red streamers attached
to them so they are visible and removed before flight. Ground locks are typically carried in the aircraft and put into place by the flight crew during the post landing walk-around.

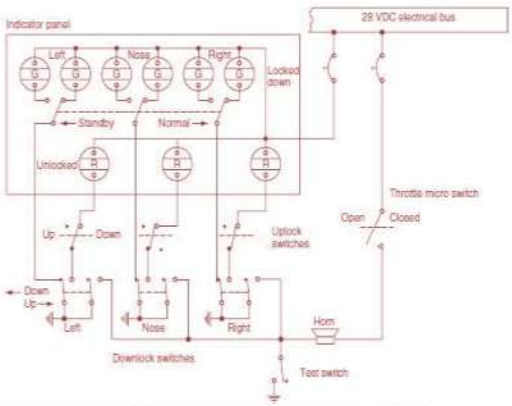
Landing gear position indication is derived from simple micro-switches in two locations: gear up and gear down. Switches are operated by a cam or lever and this completes the circuit.
The quantity of lights depends on the aircraft



manufacturer and certification requirements of the aircraft type. The simple system shows if the nose and main gear are up or down with a single indication; it is more usual to have an indication of each gear leg position. Landing gear indications sometimes include an audible warning (klaxon or horn) when the throttles are retarded and the gear is not locked down.
The system configuration in widespread use has the following indications:

- gear down and locked(three green lights)
- gear up and locked (all three lights out)
- gear in transit (three red lights).

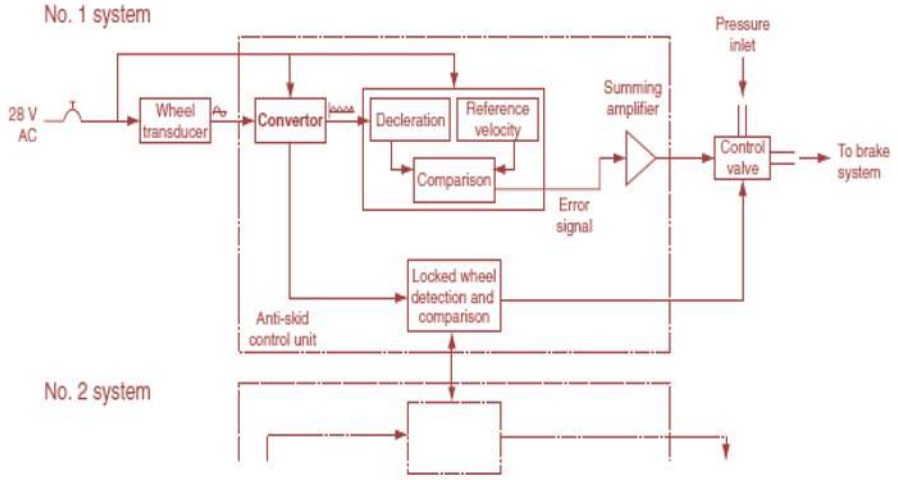
With power applied, the three green



lights are illuminated via the down-lock switch. When the gear is selected up, the down-lock switch opens, the green lights extinguish and the three red lights are turned on. When the gear is fully retracted and locked, the red lights are extinguished. When the gear is selected down, the up-lock switch closes, and three red lights come on. With the gear fully down and locked, the down- lock switch closes, the red lights are extinguished and the green lights are switched on. In the event that the throttles are closed (during an approach) and the gear is up, a warning horn will sound.

Emergency Extension of the Landing Gear

All aircraft with retractable landing gear are required to have some acceptable method of lowering the gear in flight if the normal actuating systems fail. The landing gear has as a free-fall valve between the gear-up and the gear-down lines of the power pack. If the power pack fails, the pilot can move the free-fall handle to the EMERGENCY EXTEND position, which opens the free-fall valve and allows fluid from the gear-up side of the actuating cylinders to flow directly to the gear-down side. The gear will then fall from the wheel wellswith little opposition, and its weight combined with the air moving past it will force it to mechanically lock in place. More complex landing gear systems use compressed air or nitrogen to provide the pressure for emergency extension of the gear. In such systems, ashuttle valve is installed at the actuator where the main hydraulic pressure and the emergency air pressure meet. During normal operation, fluid enters the actuator through oneside of the shuttle valve. In the event of failure of the hydraulic system, the pilot can place the gear handle in the GEAR-DOWN position, which releases the emergency air supply into the system .The piston in the shuttlevalve moves over to seal off the normal hydraulic system and direct compressed air into the actuator.



Anti-skid

The anti-skid system (also called an anti-lock braking system:
ABS) is designed to prevent the main landing gear wheels from locking up during landing, particularly on wet or icy runway surfaces. Traditional method used to operate brakes is from hydraulic pressure controlled from the brake pedals. This pressure can be varied by the pilot by increasing or decreasing the amount of force being applied to the pedals. If too much pressure is applied by the pilot, the wheels will lock-up and the aircraft will skid on the runway. The anti-skid system ensures optimum braking under all conditions by modifying the pressure being exerted by the pilot on the brake pedals The applied pressure is reduced before the wheels lock up, and then reapplied to continue the braking action; this occurs as a repeated on/off cycle. Modifying the applied pressure is achieved by modulation; this can vary the applied pressure in both time and pressure.

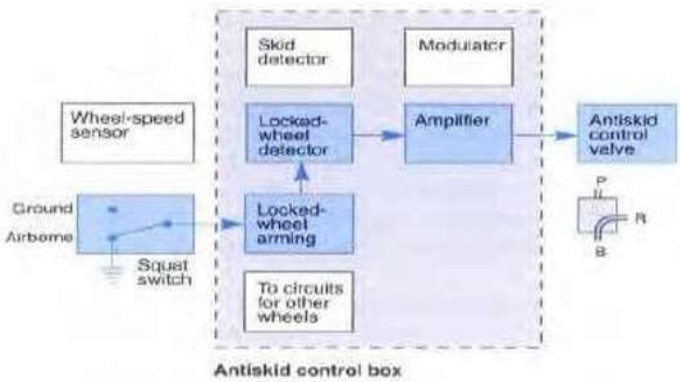
The system comprises speed transducers (or sensors) on each wheel, a control unit and electro-hydraulic control valves. During a skid condition, the wheel(s) experience a rapid deceleration. Typical speed sensors are based on the tachometer; this is a small AC generator with the stator formed with a permanent magnet. The rotor coil is built into the axle and turns inside the stator field. Referring to fundamental principles, when the coil is rotated inside the field, currents are induced and a voltage is generated; the tachometer‘s output is proportional to
wheel speed. The tachometer AC output is rectified in the control unit; the DC output is monitored for rate of change to determine if the wheel is accelerating or decelerating. A comparator circuit generates an error signal which is amplified and used to operate a control valve to modulate the applied brake pressure. Control laws are used to determine when the wheel is approaching

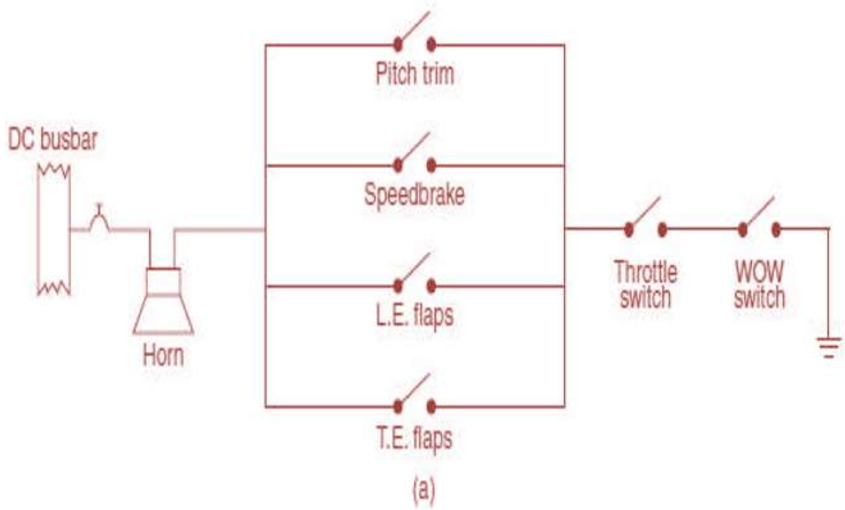
a skid condition, this occurs when the wheel speed is decreasing at a given rate. Hydraulic pressure is decreased in the braking system and the wheel accelerates. Once the skid has been averted, hydraulic pressure can be reapplied.

Configuration warning
The system (also known as a take- off warning system) provides a warning if the pilot attempts to take- off with specific controls not selected in the correct position, i.e. an unsafe configuration. A simple take-off warning system is illustrated in Fig.. The type of system
fitted depends on the aircraft type and size; typical parameters being monitored include (but are not limited to):

- pitch trim
- speed brake
- leading edge (LE) flaps
- leading edge (LE) slats
- trailing edge (TE) flaps

The position of each of these controls is monitored together with the squat switch (weight on wheels) and throttle position. If an unsafe take-off configuration is detected, a warning horn is sounded. This system can be viewed in combination logic terms as illustrated in Fig.). On larger aircraft, more parameters are monitored, and more logic functions are required for
configuration warning.





OXYGEN SYSTEMS GENERAL
If an aircraft is designed to fly at heights above, say, 8,000 feet, there must be some way in which we can maintain a comfortable environment for the crew and passengers to breathe normally. This is normally done

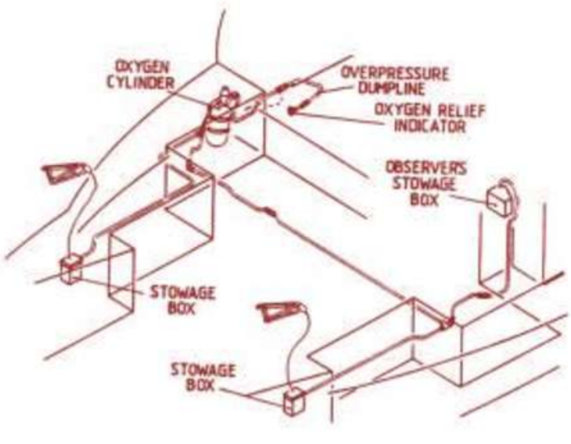
by cabin pressurisation. If for whatever reason the pressurisation failed above this altitude an alternate but emergency source of breathable air must be supplied. This is normally by individual oxygen supplies from gaseous, liquid and chemical sources. Civil aircraft use the gaseous and chemical type, with the military using liquid. Some small, unpressurised aircraft only require oxygen occasionally and use a system that meters a continuous flow of oxygen; the amount based on the altitude flown. Aircraft that fly at altitudes above 18000 feet have a diluter demand system that also meters oxygen based on the altitude flown but directs it to the mask only when the user inhales. Aircraft flying at very high altitude where the outside air pressure oxygen charging in progress

When charging a gaseous system ensure:
1. No refuelling operations are being carried out.
2. No switching on or off electrical supplies.
3. Adequate warning notices are in place i.e.
4. That there is no smoking or naked flames.
5. That the aircraft is earthed

is too low to force the oxygen into the lungs use pressure demand systems. These systems send oxygen to the mask under a slight positive pressure that forces the oxygen into the lungs.

OXYGEN SAFETY PRECAUTIONS
The safety precautions associated with the use of oxygen are laid down in the aircraft maintenance manuals. Although oxygen is none flammable it will support combustion. If oil grease dust or metal particles are present a spontaneous explosion may occur. The following safety precautions must be adhered to:
1. Keep oil and grease away. Oxygen equipment, hoses and fittings must not be handled with greasy hands or wearing greasy overalls.
2. Keep oxygen away from fire. A small fire or spark will rapidly grow in an oxygen-enriched atmosphere.
3. No smoking.
4. Handle oxygen components carefully.
5. Don‘t mix oxygen
6. Always follow any instructions given in manuals and/or on charging panels.

6. That adequate fire fighting equipment is available.



SYSTEM LAYOUT
The crew and passenger gaseous oxygen systems and their oxygen cylinders are usually independent of each other except for a common charging point and an over pressure relief facility. Both these systems provide for storage of the oxygen at high pressures and its delivery to the crew and passenger manifolds and outlets

under low pressure. In general gaseous oxygen systems are used for the cockpit and chemically generated oxygen is used for the cabin. Some aircraft use gaseous systems for both the cockpit and cabin.

15.3.1 Cockpit System Layout

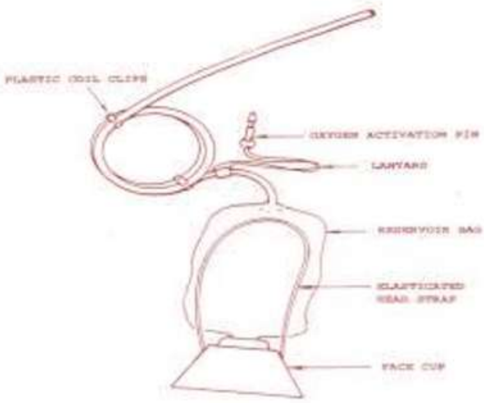
The aircrew will have a mask for each occupant. These will be quick fitting and will be located inside boxes that are within easy reach.

Crew Mask

Crew oxygen masks contain a microphone implanted in the mask that is permanently connected; to allow communications to be maintained at all times. The end of the mask hose is connected to the supply regulator that regulates the oxygen flow to the mask. On some aircraft an inflatable harness is used to allow one handed fitting of the mask. The mask and harness is contained in a storage box shown in figure 2. When the mask is required the storage box release levers are squeezed together. The box doors are unlocked and the mask is withdrawn. A green oxygen on flag will appear. On withdrawal of the mask, the harness is automatically inflated. Once the mask and harness has been fitted the release levers are released and the oxygen that has inflated the harness is exhausted to atmosphere. The harness deflates and tightens on the crew members head.
The storage box contains a test lever that can be operated to test the oxygen flow to the mask. When the system is operating correctly a blinker indication on the flow indicator turns green. There will also be a 100% selector button which when depressed will allow pure undiluted oxygen to be delivered to the mask.

Passenger Mask

The passenger masks will be found within the Passenger Service Unit (PSU) and will be deployed by gravity on actuation of the
drop out mechanism. Each seating position both in the cabin must have an easily fitted mask, which will be used by each occupant. Some aircraft do not have a mask for each person but have strategically placed masks in the PSU for the passengers to share. Some aircraft do not have —drop out‖ systems and the masks may have to be deployed manually by the cabin crew.
These are normally simple cup shaped mouldings with an elasticated strap. The cup is designed to fit all sizes
from babies to adults. A reservoir bag is fitted to the mask to store an immediate supply of oxygen.



Continuous Flow Oxygen System

Continuous flow systems are usually used in passenger oxygen systems where oxygen is needed only occasionally. These systems are wasteful of oxygen but due to their simplicity are installed on most aircraft. The oxygen is carried in a high-pressure bottle. The pressure is regulated down to around 400 psi (depends on aircraft type) by a pressure reducing valve and the oxygen is metered by a pressure regulator to around 70 psi before it is delivered to the masks.

A pressure relief valve is incorporated into the system to prevent damage in the event of a failure of the pressure-reducing valve. If the pressure is relieved through this valve a green blow out disc on the outside skin of the aircraft will blow giving a visual indication. As well as a visible blow out disc some aircraft also deploy a red streamer in an over pressure condition.

Continuous Flow Masks

Continuous flow oxygen systems use re-breather type masks. These masks may be a simple transparent plastic re-breather bag.

The mask is held loosely over the mouth and nose with an elastic band and oxygen continually flows into the bag through a plastic tube that is plugged into the mask outlet.

When the user exhales the air that was in the lungs for the shortest period is the first out and fills the re-breather bag. The remaining air in the lungs is exhausted from the mast. Inhaling again the exhaled air in the bag is enriched with the oxygen supply and is re-breathed.

More sophisticated continuous flow masks are used in pressurised aircraft. In the event of the loss of cabin pressure

an automatic valve is turned on to send oxygen to into the passenger oxygen system. The oxygen pressure actuates the door actuator valve, which opens the overhead mask compartments. A mask drops down. When the passenger pulls the mask tube a lanyard operated rotary valve opens and starts the oxygen flow. The passenger places the mask over his mouth and nose and breathes normally.

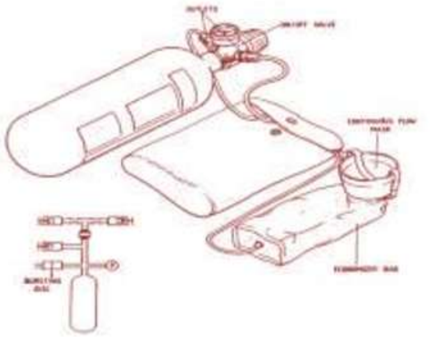
Valves mounted in the base plate of the mask allow some cabin air to enter the mask and allow exhaled air to leave. During inhalation the pure oxygen in the bag is taken into the lungs. When the bag is empty cabin air is taken in through the mask and mixes with the oxygen flowing through the tube. During exhale the air from the lungs leaves the mask through one of the valves while pure oxygen is flowing from the regulator into the bag ready for the next inhale.

Demand Type Oxygen System

The cockpit crews of most commercial aircraft are supplied with oxygen through a diluter demand system. The system meters oxygen only when the user inhales and the amount of oxygen metered depends on the altitude of the aircraft. Almost all pressurised aircraft have a diluter demand type system for the aircrew and a continuous flow type system for the passengers.

Pressure Demand Oxygen Regulator.

At altitudes above 40000 feet the oxygen in the air has such a low pressure that even the pure oxygen supply must be forced into the lungs. The low pressure from the users lungs are insufficient to draw in the oxygen. This is done under a slight positive pressure from the regulator.

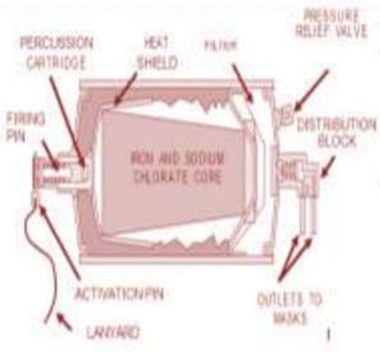


Portable Oxygen Systems

At various positions in the cabin, there are located, portable oxygen sets for use by the cabin crew to allow them to check that the passengers have all got their masks on. These masks can also be used to help breathing in the case of fumes or smoke in the cabin.

A slightly different type of oxygen set can also be found in the cabins of most passenger aircraft. These are called 'therapeutic' sets and are used for medical purposes when, for example, a passenger is having difficulty breathing. The set, which is illustrated overleaf, allows the passenger to receive an enriched or 100% oxygen supply, until they are feeling better or medical assistance is obtained after landing.
A typical portable system is shown in Figure 5. It has two outlets, which might be therapeutic. A therapeutic outlet delivers more volume of oxygen than normal and is used to aid those passengers that may have breathing difficulties or heart conditions. The types of mask used with portable equipment, depends upon the designer's or the company's requirements.

SOURCES OF OXYGEN



Chemical Oxygen Generator

Most aircraft use gaseous oxygen as the primary source for the aircrew and a chemically generated source for the passengers. Some aircraft with oxygen generators fitted are having them replaced with a gaseous oxygen system due to the
associated fire hazards. The main reason for using gaseous oxygen is the ease of handling and its availability at most airports, even though oxygen generated systems are more lightweight.
The main disadvantage of a gaseous oxygen system is that the oxygen is stored at a high pressure, it reacts explosively with greases and oils and the storage bottles are very heavy. Bottles are made from high tensile steel, but on the more modern aircraft, Kevlar wrapped aluminium alloy, carbon fibre or plastics are used. They are painted either black with a white dome top or green (USA), and stencilled with Aviation Oxygen in white letters.
Chemical Oxygen Generator
Oxygen generators or oxygen candles as they are sometimes known, is a convenient way to carry oxygen in an aircraft, when it may only be required in emergencies. They have a long shelf life and they are lightweight. The storage capacity is about three times that of a gaseous oxygen system. Once used they are easily replaced. A typical candle is shown in Figure 6.
Sodium Chlorate and iron is mixed with a binding material and is then moulded into a solid block. The block is installed in an insulated stainless steel case. When oxygen is needed, pulling the

oxygen mask withdraws a safety pin from the firing mechanism, and a spring-loaded percussion cap or an electrical squib igniter starts the sodium chlorate decomposing by chemical reaction. Enough heat is generated to start the reaction and then the heat of the reaction sustains itself. (It does not burn).
As it decomposes it releases the oxygen at a pre-determined rate. The block will continue to react until the sodium chlorate is consumed. There is no way to cut off the process once it has started. The by product of the reaction, apart from the oxygen, is sodium chloride (salt) and ferrous oxide (rust)
The oxygen that is produced is proportional to the cross sectional area of the core and the rate of reaction. The rate of decomposition is determined by the concentration of iron in the core. The oxygen production is

greater at initial reaction (larger cross sectional area) to provide high oxygen output during the initial few minutes of the emergency decent. Once generation has started core temperature is approximately 450 degrees F.
The distributing and regulating system is self-contained. It consists of a manifold attached to one end of the stainless steel cylinder. The oxygen is filtered to remove any salt particles before it is supplied to the manifold. The manifold contains calibrated connections for a number of oxygen masks and they ensure an equal flow to each mask. Normal output from the generator is 10 psi and it is therefore not regulated prior to breathing. A pressure relief valve is also located on the casing to relive pressures in the generator above 50 psi.
The disadvantage of the system is mainly the large amount of heat generated, which means that the generator must be well insulated from the airframe structure. Some aircraft that use oxygen generators are replacing them with gaseous oxygen systems due to associated the fire hazards.

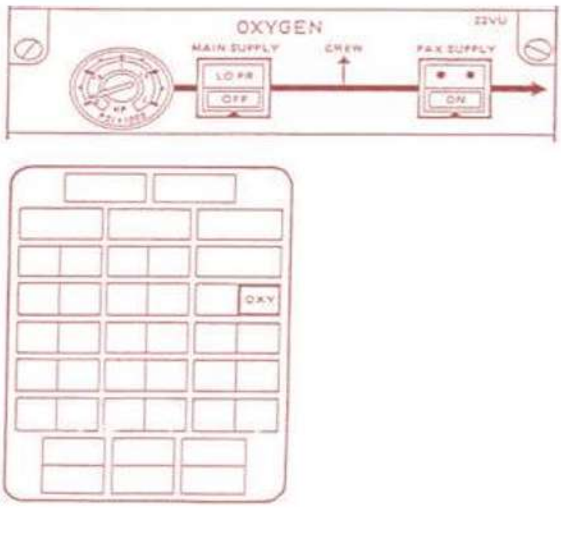
Gaseous Oxygen Systems
Oxygen in a gaseous state is contained in storage cylinders the number and capacity of the cylinders depending upon the number of passengers and crew. The normal charge of the cylinders is usually 1800 psi and a capacity of 30 to 120 cubic feet. Cylinders normally have a manually operated shut off valve in the neck of the bottle to facilitate bottle removal. A direct reading pressure gauge is also fitted, as is an electrical transducer that sends pressure indication signals to the cockpit instrumentation.
Charging Of Systems
Gaseous systems can be re-charged either at the aircraft, from portable, large capacity, bottle sets (oxygen trolley), or by removing the bottle itself, via quick release clamps and connections, and re-charging in a dedicated oxygen charging room or bay.
Which system is used is dictated by the Airworthiness Authority of the country of registration, some of which allow 'on aircraft' charging, whilst most insist that all bottles are removed for re- charging remotely from the aircraft.
With ―on aircraft‖ charging, a regulated oxygen trolley supply is attached to the aircraft charging point. The hose is usually purged before connection to clear the hose of impurities and moisture. During the charging process temperatures are generated in the pipelines to the storage bottles. To dissipate this temperature thermal compensators are installed.
These compensators are sintered bronze elements soldered inside the pipelines. They act as a heat sink to dissipate the heat whilst allowing the oxygen to flow to the storage bottles.
With chemical oxygen systems, when the units themselves become life expired and due for return to their manufacturers, they are simply removed as a unit from the PSU. When they have been made safe, (usually by the fitting of a safety pin in the firing sear), they are returned to the factory.
Oxygen Distribution
The supply pipes, in the high-pressure side of the system, from the storage bottle to the pressure-regulating valve, are made from stainless steel or copper based alloys. They are colour coded at each end with the words ―breathing oxygen‖ and a black rectangular symbol on a white background.
From the storage bottle the pressure is reduced to an acceptable level before being distributed to the passenger and crew compartments. As the maximum pressure to the masks will be 70psi, the distribution pipelines, from the pressure regulator valve are made from aluminium alloy or plastic.
The distribution lines for the aircrew go from the storage bottle to the cockpit pressure regulator and the passenger lines go from the storage bottle, up the side walls and then along the roof. Each passenger service unit (PSU) where the masks are stowed are connected to the roof piping. Test connections are installed in the system to allow pressure gauges to be fitted during system testing.

SUPPLY REGULATION
Diluter Demand Type Regulator

Oxygen flows into the regulator through the supply valve and when the user inhales the pressure inside the regulator decreases and the demand valve opens under action from the demand valve diaphragm allowing oxygen to flow to the mask.
The aneroid capsule operated metering valve mixes cabin air with the oxygen. When the aircraft is flying at low altitudes the user gets mostly cabin air and a small amount of oxygen. As the altitude
increases the aneroid capsule metering valve progressively reduces the amount of cabin air and increases the amount of oxygen supplied. At about 34000 feet the cabin air is shut off completely and pure oxygen is supplied.
If there is smoke in the cockpit or if the pilot feels the need for pure oxygen the oxygen lever can be moved to the 100% position. The cabin air is shut off and the aneroid metering valve fully opens and only pure oxygen is supplied to the mask when the user inhales.
If the regulator malfunctions the emergency lever can be operated. This opens the demand valve allowing a continuous flow of pure oxygen to the mask.

Continuous Flow Regulators.
There are automatic and manual continuous flow regulators. The automatic regulator contains and aneroid capsule that senses the aircraft altitude and meters the correct amount of oxygen for that altitude. The manual regulator has a control valve that allows the pilot to adjust the flow rate based on the altitude. A calibrated orifice in the mask outlet determines the amount of oxygen delivered to the mask.
INDICATIONS AND WARNINGS
The systems are provided with an overpressure relief facility. This is normally a green coloured rupture disc. The disc will be located at the overboard discharge fitting which is flush with the aircraft‘s skin. When the maximum cylinder pressure is exceeded the cylinder safety valve operates discharging the excess pressure into the overboard discharge line. The green disc ruptures, as the excess pressure escapes to atmosphere and a red (or yellow) indicator

becomes visible. Some aircraft also deploy a red streamer from the fitting to make it instantly visible.
On aircraft with oxygen generators fitted, once the generator has been activated, dolls eye indicators on the end casing turn from orange (or purple) to black. Some have heat sensitive tape wrapped around the outer casing. The tape changing colour when the generator has been activated.
There will be various indications given on the oxygen panel and centralised warning panel (CWP) indicating faults within the system.
A pressure gauge will be fitted which shows the pressures in the storage bottles. The gauge will have a green segment and a red segment. The green segment indicates the actual pressure in the system. The red segment will indicate that the bottle is empty or maybe that the shut off valve is closed.A low pressure switch is fitted in the system downstream of the storage bottle and will give an indication (LO PR) on the local panel if the pressure reduces below a pre-set
figure. An associated (OXY) caution light will illuminate on the CWP and a single chime warning will also sound.
Indications and Warning Panels



13.18 Pneumatic/Vacuum (ATA 36)

GENERAL

Pneumatic systems are fluid power systems that use a compressible fluid, air. These systems are dependable and lightweight and because the fluid is air there is no need for a return system

Some aircraft have only a low pressure pneumatic system to operate the gyro instruments, others use compressed air as an emergency backup for lowering the landing gear and operating the brakes in the case of hydraulic failure. Other aircraft have a complete pneumatic system that's actuates the landing gear retraction, nose wheel steering, passenger doors and propeller brakes.

SAFETY PRECAUTIONS

When working on bleed air systems, it is important to follow the precautions below:

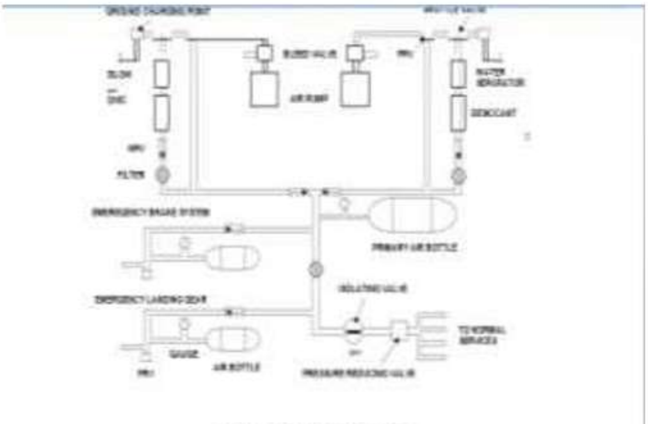
Bleed air is hot! Do not touch pipes and ducts.

Always replace seals, (normally crush seals), when replacing joints.

Tighten clamps to the torque figure quoted in the Maintenance Manual.

Never lever against ducts, as dents cause hot spots.

All duct supports and struts must not put any strain on to the duct.



A Typical Pneumatic System

Figure 1

FULL PNEUMATIC SYSTEMS

The majority of aircraft use hydraulic or electrical power to operate landing gear systems, but some aircraft use air systems. Some advantages of using compressed air are:

Air is universally available and in unlimited supplies.

Pneumatic system components are reasonably simple and lightweight.

No return lines are fitted resulting in a weight saving.

There is no fire hazard and the danger of explosion is slight. Contamination is minimised by the use of filters.

Figure 1 shows a typical high pressure pneumatic system, that uses air compressors driven from the engines accessory drive. The compressed air is discharged through a bleed valve to a pressure relief (unloading) valve. The bleed valve is held closed by oil pressure. In the event of oil pressure failure the bleed valve opens to offload the compressor. The pressure relief valve maintains system pressure at around 3000 psi.

A shuttle valve in the line between the compressor and the main system makes it possible to charge the system from a ground source. When the engine is not running the shuttle valve slides over to isolate the compressor.
Moisture in a compressed air system will freeze as the air pressure drops when a component is actuated. To prevent this from happening, the water must be completely extracted from the air. A
water separator is fitted which collects the moisture from the air onto a baffle and it is allowed to drain overboard. An electric heater prevents the water in the separator from freezing.
After the air leaves the water separator any remaining moisture is removed as the air flows through a desiccant or chemical dryer. The air is then filtered before it enters main system.
The air is then fed to each of the storage bottles, which provide the emergency air for several systems. A manually operated isolation valve allows the air supply to be shut off to so that maintenance can be carried out on the systems without having to discharge the storage bottles.
The air is stored at maximum system pressure around 3000
psi to supply the landing gear and brakes in an emergency. A pressure reducing valve is fitted to reduce the air pressure down to the operating pressure that the majority of the components work at 9around 1000psi) ie landing gear normal operation, the passenger door, the propeller brake and the nose wheel steering.

VACUUM SYSTEMS
A supply of air at a negative pressure can be required for a number of purposes. The supply of vacuum to instruments for example, usually comes from either a small vacuum pump attached to the (piston) engine of the aircraft or from a venturi jet pump, which obtains its power via a tapping from the (jet) engine. The low pressure caused by the venturi draws in air to supply the system.
Other requirements for a source of vacuum might be in a pneumatic de-icing system. This type of de-icing uses the inflation of flexible leading edge mats to break-off the ice, which has formed. To keep the de-icer boots, as they are called, in place, they are fed a negative pressure from a venturi, which ensures that the

boots are sucked flat onto the wing leading edge, ensuring a smooth, aerodynamic surface.

Vane Type Air Pump
Figure 2

LOW PRESSURE PNEUMATIC SYSTEMS LAYOUT
These systems provide air for gyroscopic altitude and direction indicators and air to inflate the pneumatic de-icing boots. This compressed air is usually provided by a vane type engine driven air pump (Figure 2).
Engine Driven Air Pump
On early aircraft engine driven air pumps were used primarily to evacuate the casings of air-driven gyroscopic instruments so they were more commonly known as vacuum pumps. On later aircraft the discharge air was used to inflate de-icing boots on control surfaces and are now more correctly called air pumps.
There are two types of air pumps that are used, these are wet air pumps and dry air pumps.
Wet Air Pumps
Wet pumps have steel vanes that are lubricated and sealed with engine oil which is drawn in through the

pump mounting pad and exhausted with the discharge air. This oil is removed from the discharge air with an oil separator before it is used for de-icing or driving the instruments.

Dry Air Pumps

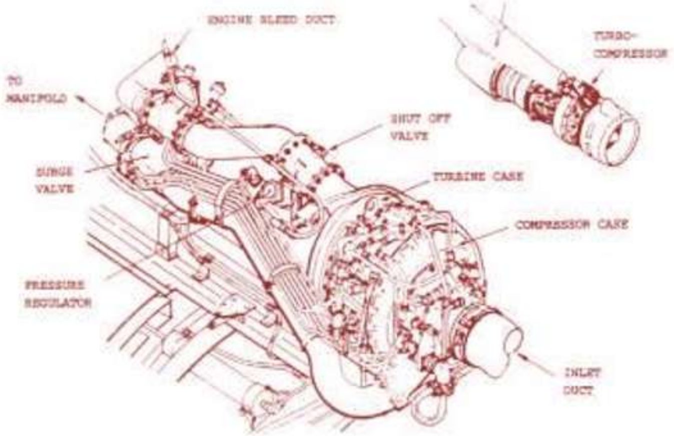
Dry air pumps were developed so that there was no oil in the discharge air and therefore there were no requirements for an oil separator. The pump vanes are made from carbon and are self lubricating. The main problem with this kind of pump is that the vanes are easily breakable by any contaminants that enters the pump. To prevent this form occurring the inlet air is filtered.

AIR SUPPLY SOURCES

The source of air supply and arrangement of the system components depend on the aircraft type and system employed but in general one of the following methods may be used:

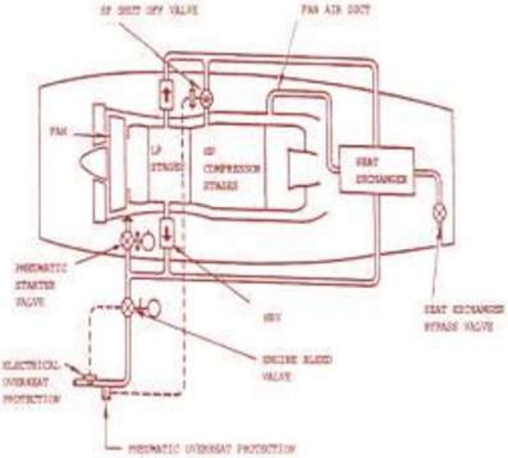
Engine Bleed Air

This is used in turbo jet aircraft in which hot air is bled of from the engine compressors to the cabin. Before the air enters the cabin it is passed through a pressure and temperature control system which Reduces its pressure and temperatu re and is then mixed with ram air.



Because of the great variation of air output available from ground to maximum flight rpm there is a need to maintain a reasonable supply of air during low rpm operation as well as restricting excessive pressures when operating at full speed. Two tappings are taken from the engine, one form the LP stages and one form the HP stages to maintain a reasonable pressure band at all engine speeds. Figure 3 shows a typical 2 stage bleed air system.

At low engine rpm the LP air is of insufficient pressure for use in the pneumatic systems, so air will be tapped from the HP stages. When engine speed increases the LP air pressure will also increase and at a pre-determined pressure the HP air will be shut off and when operating at maximum engine speeds the air will be taken purely from the LP stages. In all normal stages of flight therefore the bleed air will come form the LP stages.



Compressors or Blowers

This is used by some turbo jet, turbo prop or piston engine aircraft, the compressors or blowers being either engine driven via an accessory drive, by bleed air or electric or hydraulic motors. The compressor inlet duct is connected to an air scoop and its outlet is connected to the pneumatic manifold. The unit is controlled by a shut off valve which is operated from the cockpit.

When insufficient LP air pressure is available for the pneumatic systems at low engine speeds the aircrew will select the shut off valve to open. This will direct the LP air to drive the turbo compressor. A pressure regulator is incorporated to ensure a constant output at the required pressure.

On large multi-engine aircraft only some of the engines will have a turbo compressor (Figure 4) which is normally mounted with its associated controls in an engine bay.

Auxiliary Power Unit (APU)

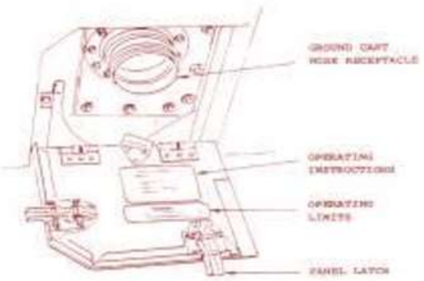
This provides an independent source of pressurised air. It is basically a small gas turbine engine that provides air and other service whilst the aircraft is on the ground with its main engines stopped. It is usually a self contained unit located in the tail section of the aircraft where it can be run safely (Figure 5). On some aircraft the APU can be started in flight and act as a back up source of air, hydraulics services in the event of a loss of an engine. Typical APU Setup Ground Supply For use on the ground when the engines are not running. This unit will run until the aircraft is independent of the trolley. The ground cart is basically a compressor driven by an engine, usually a diesel. The compressor output pressure is regulated to match the aircrafts system pressure. A quick release hose is connected from the cart to the aircraft service panel. The maximum aircraft systems pressure and operating instructions including safety precautions are detailed on the inside of the service access panel.

Ground Cart Control Panel

Instructions for operating the ground cart will be found on a panel on the carts control panel. Figure 6 shows a typical ground cart control panel.

PRESSURE CONTROL

In many bleed air systems the pressure is regulated only by the operation of the high pressure shut off valve. The range of pressure may be from 10psi at ground idle to 65 psi at take off power. Many modern aircraft use bleed air for many systems



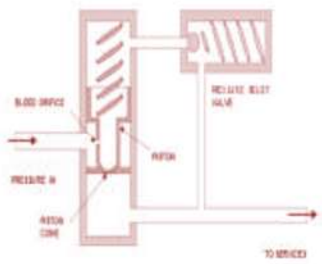
that are sensitive to pressure variations and therefore some form of regulation is required.

The pressure regulator is a pneumatically operated valve which will give a pre- determined output pressure form the engine bleed air system. The regulator may also perform as the shut off valve. This is then called a pressure regulating and shut off valve.

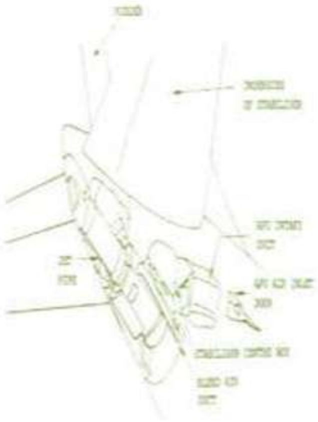
Pressure Regulator

This valve operates on the principle of a balance between air pressure and spring pressures. Referring to Figure 7. Assuming the piston has an area of 1

square inch and is held in its seat by a spring that pushes with a 100 pounds force. The piston has a shoulder of 0.5 square inches and this area is acted on by a system air pressure of 1500psi. The cone shaped seat of the valve has an area of 0.5 square inches and is acted on by a reduced pressure of 200psi. A bleed orifice in the piston allows air pressure into the



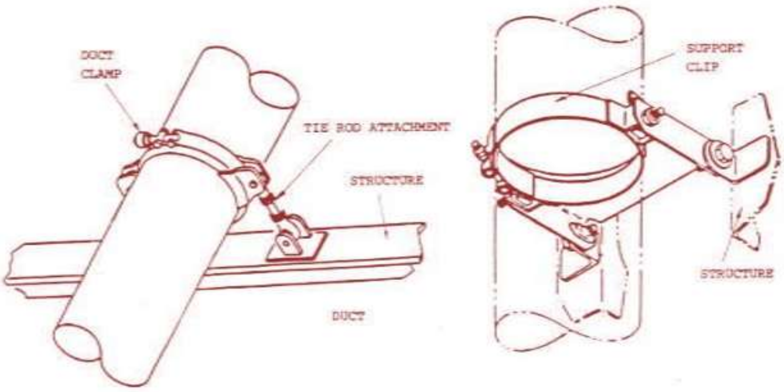
Pressure Regulator
Figure 7



piston chamber. A relief valve being acted on by the reduced 200psi pressure and relief valve spring pressure, maintains the air pressure in the piston chamber at 750psi.

When the air supply is used by a pneumatic service, the reduced downline pressure of 200psi reduces further. This reduced pressure is now insufficient to keep the relief valve closed. The 750psi piston chamber pressure unseats the relief valve and reduces the piston chamber pressure.

The reduced piston chamber pressure unseats the piston cone piston which allows the system pressure



to bleed into the down lines. Once the downline pressure rises to 200psi, the piston cone and the relief valve re-seat and the system is once again in balance.

DISTRIBUTION

Distribution is achieved by ducting and pipelines that carry the charge air from the engine compressors to the various services that require air for their operation. Due to the heat of the bleed air any leakage of the ducts will cause an extreme temperature rise in the area of the leak with the possibility of fire or damage to the surrounding structure and equipment. Leak detection systems are therefore incorporated. Figure 9 shows a typical distribution layout.

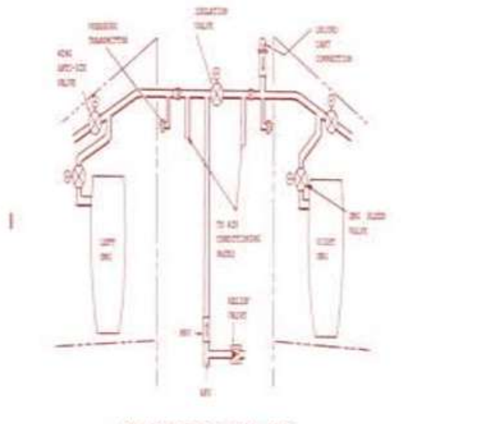
The ducting is made up of many sections for ease of maintenance and cheapness of replacement. They are constructed of thin wall material and clamped together with joints that allow for thermal expansion.

Engine bleed air system ducts are manufactured from stainless steel and the ducts and pipelines are usually manufactured from titanium as they are able to withstand higher temperatures and are

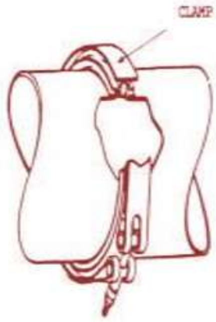
lighter in weight. The duct sections are supported throughout their length by clamps and tie rod attachments to the aircraft structure as shown in Figure 8.

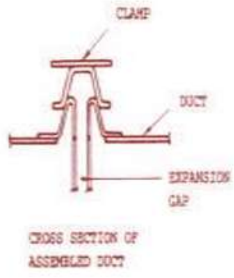
Expansion Joints

Joints are assembled cold and when in use the temperatures ducting can reach up tom 350 degrees F. Expansion devices must be incorporated into the systems to prevent any



Bleed Air Distribution Manifold
Figure 9





distortion or buckling of the ducts. This expansion can be allowed for in several ways.

Pre-Stressed Joint

One method is to have the duct sections installed slightly shorter in length and allow them to expand with the heat to fit correctly. The ducts will be pre-stressed by the clamps when cold (Figure 10).

Flexible Ball Joint

Another method is with a flexible ball joint fitting at the duct ends. The joint is designed to allow for slight flexing and misalignment as well as expansion. A flange on one end of the duct is connected to a bearing nut on the other and screwed together to form the joint (Figure 11). Shims are used to ensure adequate clearance is maintained for the expansion and flexing and a crush type metal seal is used to prevent air leakage at the joints.

Cable Attachment Joint

The cable attachment type joint is used where large temperature changes exist, ie from cold soak at high altitudes to maximum working temperatures when the pneumatic system is selected on. This joint has bosses attached at each end of the duct.

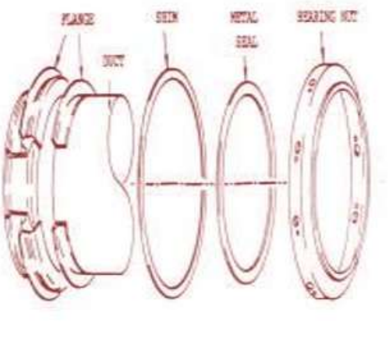
There are usually 3 short cables equally spaced around the duct (Figure 12). The cables have a swaged ball end fittings at one end and a swaged threaded fitting at the other. Each end is located in a bracket on the ducting. A seal is fitted around the duct before the ducts are connected. A nut is fitted on the threaded end and tightened. This pulls tightens the cables and seals the duct. A small gap is left at the seal ends to allow for expansion.

INDICATIONS AND WARNINGS

Safety devices are fitted into pneumatic systems to prevent a possible overheat or overpressure which could cause severe damage to the air ducting or systems.

Overpressure

Overpressure is usually caused by a malfunction of the high pressure shut off valve that remains open when the engine is operating at its maximum rpm. In most systems a pressure relief valve is fitted in the engine bleed air ducting which relieves excess pressures. The pressure relief valve may also work in



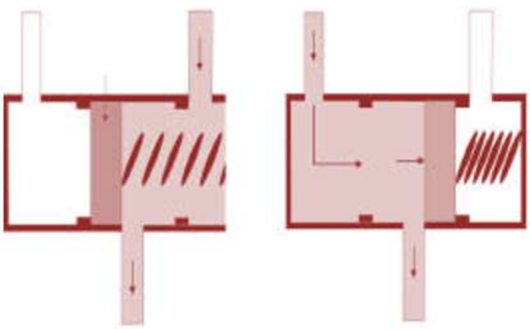
conjunction with a pressure switch will close the high pressure shut off valve at a pre determined pressure.

Overheat
Over temperature of the bleed air is prevented, by an electrical temperature sensor, downstream of the engine bleed air valve. When a pre determined temperature is reached the electrical sensor will signal the high pressure shut off valve to close. An overheat will be indicated to the aircrew on the CWP and associated control panel.

Duct Hot Air Leakage
Any ducting that includes joints is liable to leak under abnormal conditions. A duct protection system will include fire-wire elements around the hot zones such as engine air bleeds, air conditioning packs and auxillary power units if fitted.
The sensing elements will be the thermistor type. As the temperature around the wire increases the resistance decreases until an electrical circuit is made. When the circuit is made a
warning signal is sent to the cockpit central warning panel with associated caution/warning lights and aural chimes. The leaking duct may be isolated automatically or may require the pilot to take action to close off the air valves. The faulty system will then remain out of use.

SYSTEM INTERFACES
The pneumatic system interfaces with various other aircraft systems. Once the bleed air has been reduced in pressure to around
40 to 50 psi, most services have their own pressure and temperature controls, as well as generating their own warnings and indications to the CWP or system control panels in the cockpit.
Pneumatic Gyro Power systems
The gyroscopes in pneumatic gyro instruments are driven by air impinging on cups cut in the periphery of the wheel. There are two methods of obtaining air to drive the instruments:
Air Pump Suction
The air pump suction evacuates the instrument case and draws air in through a filter. The filtered air id directed through a nozzle and it strikes the driving cups to drive the gyro
instrument. A suction relief valve regulates the suction to the correct value to drive the instrument and a suction gauge reads the pressure drop across the instrument.
Dry Air Pump Pressure.
Since many aircraft fly at high altitudes where there is insufficient air pressure to drive the instruments another method must be used. The gyro instruments are driven by the air from the

pressure side of a dry air pump. The air is filtered before it is taken into the air pump and is regulated before it flows through an in line filter to the instruments. After driving the instruments it is evacuated overboard.
Backup High Pressure Pneumatic Systems
On some aircraft, in case the hydraulic systems fail there must be provision for an emergency extension of the landing gear and application of the brakes. The system comprises of a pressurised cylinder which contains approximately 3000psi of compressed air or nitrogen. A shuttle valve (Figure 13) in the actuator line directs hydraulic fluid to the actuator for normal operation or compressed air/nitrogen for emergency operation.



Pneumatic De-Icing systems

The compressed air system used for inflating de-icing boots uses wet air pumps. The oily air leaves the pump and passes through baffle plates in an oil separator. The oil collects on the baffles and drains down to a collector at the separator base and returned to the engine oil sump.

Clean air leaves the separator and flows through the de-icing selector valve to a pressure regulating valve, where its pressure is reduced to the value needed for the boots. It then flows to the distribution sequencing valve. When the system is switched off the air is directed overboard. De-icing systems are dealt with in more detail in Module 11.12 Ice And Rain Protection.

Air Conditioning And Pressurisation.

Bleed air supplies provide hot air to the air conditioning packs. The hot air passes through primary and secondary heat exchangers before it is mixed with cold air to provide conditioned air into the aircraft. As the hot air passes through the system it flows across a turbine which drives the system compressor.

Bleed air is also used for cabin pressurisation. The air drives a compressor which pressurises the air before it is fed to the cabin. Some aircraft use a jet pump to pressurise the air. Th air passes through an inter cooler to reduce its temperature before entering the cabin.

Air-conditioning systems are often protected by flow control valves, which double as shut-off valves in the case of a fault.

Air Driven Hydraulic Pumps.

Some aircraft use hydraulic pumps operated by air turbines. These are driven by bleed air from the engines and the flow is controlled and modulated by a solenoid operated pressure regulator and shut off valve to maintain the turbine speed within set limits. The turbine is connected to the pump via a shaft and the air is exhausted to atmosphere from the turbine outlet.

Pressurising Of Hydraulic Reservoirs.

Aircraft flying at altitudes in excess of 20000 feet require the hydraulic reservoir to be pressurised to prevent foaming of the fluid, due to the low ambient air pressure and to prevent pump cavitation. The bleed air is fed to a regulator/reducing valve which regulates the pressure supplied to the reservoir. A pressure relief valve is fitted to the system which vents any excess air pressure to atmosphere.

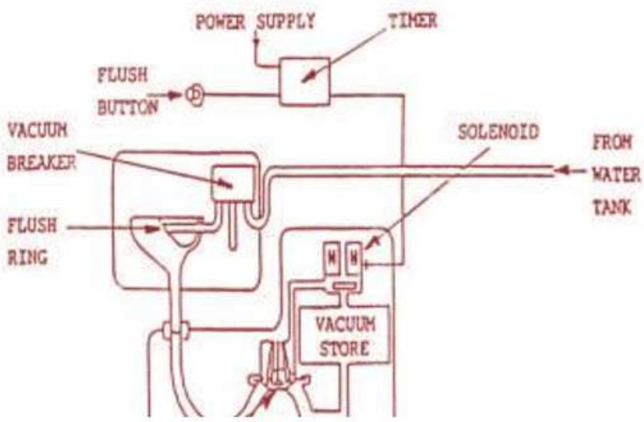
Waste And Water Systems

The toilet systems fitted to larger aircraft use a vacuum to empty a number of toilets into a single collector tank. This saves having a self-contained tank, full of de-odorising fluid and the associated pumping mechanisms attached to each toilet assembly.

The flush operation consists of fresh water from the potable supply and, most importantly, the vacuum, which draws the waste into the collector tank. This is obtained by having the tank connected to the outside of the aircraft. Only at low levels, when the outside air pressure is insufficient, is a small vacuum pump called into operation. Figure 14 shows a typical vacuum toilet system.

Pneumatic Stall Warning

These systems are common on light aircraft. A slotted plate is mounted on the wing leading edge and its slot coincide with the stagnation point of the wing during normal flight. The slot is connected to a horn via a tube. When the angle of attack is sufficient to induce a stall the low air pressure is drawn into the tube and sounds the horn giving the pilot warning of an impending stall.



13.19 Water/Waste (ATA 38)

WATER SYSTEMS

The term —Potable water‖ refers to drinking water. On aircraft it is used not only to supply water for drinking, but also the galleys and to provide hot and cold water to wash basins throughout the aircraft. A centralised water tank can feed any number of galleys and toilets through a gallery of pipes. This will speed servicing turnaround times when there need only be one main replenishment point. Potable water is Hyper-chlorinated to control bacteria and is carried out at set intervals. The major components in a potable water system are:

A storage tank.

Air pressure system to force water from the storage tank to the services.

Distribution lines Filling system

Quantity indication system Valves to drain the system

The tank is usually stored under the cabin floor in a cradle structure and is constructed from either fibreglass with metal bonded bands or stainless steel. The quantity and volume will be dictated by the number of passengers carried and the length of the time the aircraft is airborne. Aircraft that are expected to operate in cold climates may have heater blankets built in to the design to keep the tank and the replenishing panel free of ice.

The tank assembly will incorporate a drain, filler connection, overflow connection an air pressure connection and outlet pipelines to the galley and toilets.

Pressure Control

The supply of air for the movement of water is, tapped from the bleed air supply of the engine compressor or the APU. Some aircraft, which require the ability to draw water when there is no air pressure (on the ramp), have an electrically powered air compressor that will provide a head of pressure to enable water to be drawn off at any time. The compressor may automatically start when the bleed air pressure drops below a pre-determined value.

On aircraft using a compressor, a riser loop is incorporated to prevent water entering the compressor, the top of the loop being higher than the distribution ducting ensuring that the water goes to the distribution lines first. A pressure switch will control the compressor starting and stopping as the bleed air pressure varies.

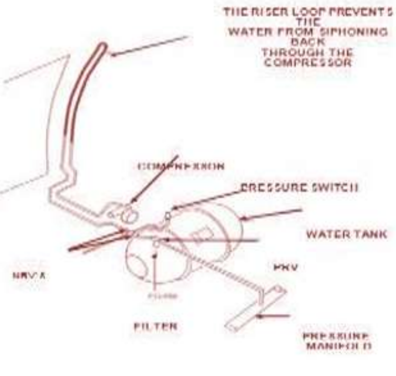
The distribution lines are connected to the tank drain, fill connection, overflow connection, air pressure connection and the supply lines to all of the galleys and lavatories.

Water Distribution System

A main water distribution line is taken from the water tank and is routed up into the cabin ceiling. Individual pipelines are routed from this pipeline to the toilets and galleys. The distribution lines are usually flexible hoses enclosed in an aluminium sheath.

The flexible hose is normally insulated to prevent it from freezing. The outer sheath prevents any leakage from entering the cabin. Any leaking water will be directed to the lower fuselage through drain tubes where it can then be drained overboard. A quick release connection is located above each toilet and galley to enable the supply line to be disconnected for removal of the toilet or galley.

On smaller aircraft the water tank may be located above the wash basin and galley areas and provides



water to the systems under gravity.

Water Heating

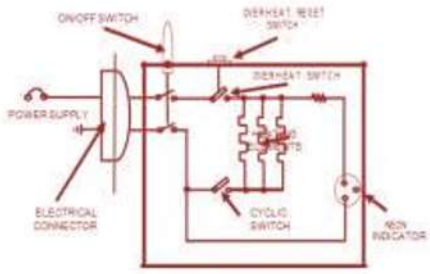
A water heater with a small capacity is installed in the supply piping under each lavatory sink and provides hot water to the hot water tap. The heater contains electrical elements in the base of the heater unit. On the side of the tank is a warning light, a control switch, an overheat re-set switch and a pressure relief valve.

Normally the heater switch will be on and the light will be illuminated. A switch controller will regulate the water temperature to around 125 degrees F. If a malfunction occurs and the temperature increases to 190 degrees F the overheat switch will operate and switch off power to the heater unit. The power light will go out. After a cooling down period the heater will have to be manually reset by pressing the re-set button on the heater unit.

A pressure relief valve will relieve pressures in excess of around
140 psi. the primary function of the relief valve is to relieve pressures caused by the water overheating. A typical water heating system is shown in Figures 2.

Waste Water Collection And Drainage

Waste water
collection and drainage depends on the aircraft. On some aircraft the water from the washbasins drains directly overboard while

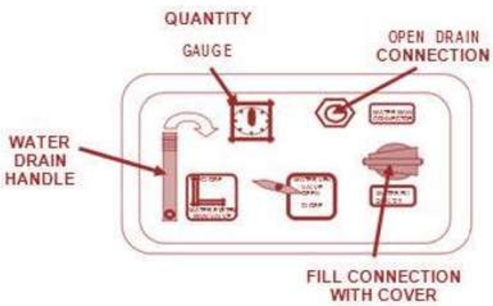


Heating System – Schematic
Figure 2

on others it drains into a soil tank and is used to flush the toilet system. Water drained overboard are drained through drain masts under the fuselage. These masts are normally electrically heated to prevent freezing and the forward motion of the aircraft ensures that the water is finely atomised as it leaves the aircraft. To test the drain mast heaters on the ground the hand is carefully used feeling for warmth.

Quantity Indication

Some aircraft use a simple sight gauge by the side of the tank to indicate the level of the waste tank contents. On larger aircraft the tank will be
fitted with a sensor to
remotely signal the tank levels to the cabin crew. One method of indication is to use a gauge on the attendants
panel and a
corresponding gauge which is fed from the same float and electrical transmitter on the water service panel.



Servicing Panel
Figure 4

Another method of indicating tank contents is to use a series of lights controlled by magnetic floats installed inside the tank. When the waste water level operates one of the magnetic floats a circuit is made and a corresponding light on the panel is illuminated.

Water Service Panel

A water service panel will normally be found on the lower part of the fuselage, where it can be easily reached by the maintenance crew who have the job of replenishing the tank during the

turnaround maintenance. The panel will probably contain most of the following items. A filling point, a drain/overflow point, a quantity indication, either in the form of an array of lights or a gauge unit and an external air connection. A typical servicing panel is shown in Figure 4

The filling point on the panel will allow the replenishing rig/ truck to fill the tank during a turnaround servicing, whilst the drain/overflow will show when the tank is full. When full any excess water overflows out of the overflow line. Once the water is seen from the overflow valve the fill/vent valve is closed to the vent position.

The quantity indicator will allow the tank to be filled to a 'less- than-full' quantity, where the aircraft is, perhaps, on very short flight legs and the excess weight of the water that will not be used, is traded-off against fuel. The external air connection allows a ground air air source to be connected to allow the water to be moved, within the system, whenever there is no internal air pressure available.

The water drain valve is manually operated and allows the tank contents to drain under gravity. When the tank is emptied the drain valve is manually re-set. The fill/vent valve can be manually or electrically operated and rotates the valve to the fill or vent position. Its operation may also electrically isolate the air compressor, if fitted during filling.

The purpose of the vent valve is to prevent an air lock occurring in the wash basin taps by opening the tap lines to atmosphere. Modern aircraft have self venting taps which automatically relieve any air locks.

WASTE SYSTEMS

The provision of aircraft toilets is an essential requirement for any aircraft carrying passengers over long distances. These toilets must be maintained and serviced with care, as the comfort and health of the passenger must be protected. They should be clean and odour free at all times.

There are three main types of toilet fitted to aircraft. The type used will depend upon the number of passengers the aircraft can carry, and also the age of the aircraft. In all cases it is essential that all the relevant health precautions are observed during all forms of servicing carried out on these units.

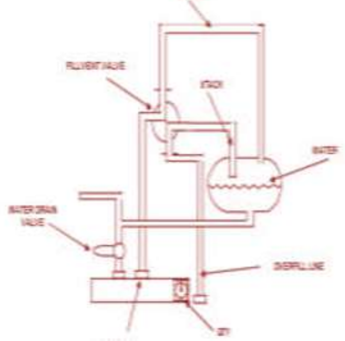
Due to the nature of the fluids carried in many toilets, protection must also be given to the structure of the aircraft to protect it from corrosion caused by these fluids.

The three types of toilet are: Removable toilet assembly.

Liquid flush type. Vacuum toilet assembly.

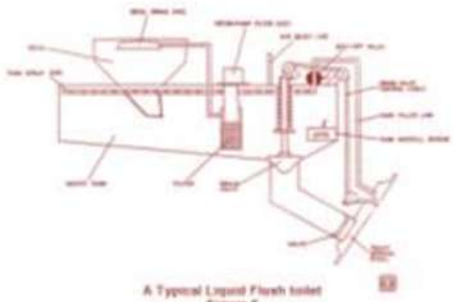
Removable Toilet Assemblies

The removable, or 'carry out' toilet is of the simplest type of aircraft toilet. This unit is often referred to as an „Elsan", named after the original company which manufactured this type of toilet. It is simply a storage bin with a toilet seat fitted to the top and partially filled with a strong chemical deodorant.



This type of toilet is removed from the aircraft and emptied into an approved disposal site. Once washed out, it is replenished with deodorant and re-fitted into the aircraft, using some form of quick release attachment such as 'pip' pins.
You will only find this type of toilet fitted to short range small light aircraft.

Liquid Flush Toilets
These are the most common type of toilet found in passenger aircraft, each toilet being a completely self contained quick release unit, having its waste collection tank mounted directly beneath the toilet bowl. The tank is normally made out of composites or plastics. Directly below the waste tank is a service panel. An illustration of a typical liquid flush toilet assembly is shown overleaf.
The assembly shown in Figure 6, contains the following components, which will be found in most liquid flush toilet installations:
Motor and Pump. Filter.
Drain Valve Rinse Ring. Flush Line. Air vent.



A Typical Liquid Flush toilet
Figure 6

System Operation
When the flush button is pressed, the motor runs for a fixed time, usually around 10 seconds, which pumps the fluid through the bowl spray pipe in a swirling action. This action flushes the bowl contents into the tank, via a lightly sprung (loaded into the closed position) hinged separator. At the end of the 10-second cycle, the motor re-arms to run again, in the reverse direction, to ensure the filter does not become blocked with solid waste.
Vacuum Toilets
On an aircraft fitted with a number of liquid flush toilets, there were two major problems, the corrosion risk and the time taken to drain and replenish each individual toilet. Both of these problems are overcome by installing vacuum toilets.

There are dry toilet modules installed at convenient locations, to suit the seating layout around the passenger cabin and connected to a central storage tank by pipelines.
The vacuum toilet uses a waste container that has a negative pressure inside, (vacuum). This vacuum draws the waste from the bowl together with the clean flushing water and deposits it into the tank. On very large aircraft, more than one waste tank is used to overcome the problem of one tank filling up during the flight.
The toilet systems fitted to larger aircraft use a vacuum to empty a number of toilets into a single collector tank. This saves having a self-contained tank, full of de-odorising fluid and the associated pumping mechanisms attached to each toilet assembly.
The flush operation consists of fresh water from the potable supply and, most importantly, the vacuum, which draws the waste into the collector tank. This is obtained by having the tank connected to the outside of the aircraft. As the aircraft"s speed increases the pressure at the connection drops which causes the waste to be drawn to the storage tank.
At low speeds or low altitudes, when the pressure differential is insufficient to draw the waste to the storage

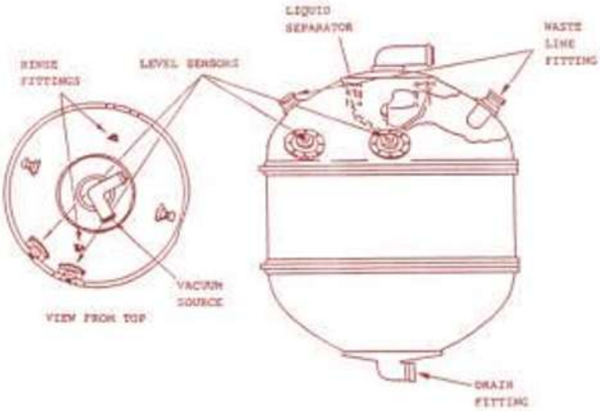
tank, a small vacuum pump called a „vacuum generator", is operated by a pressure switch to provide the required pressure drop. Its normal range of operation is between sea level up to 16,000 ft.
The illustration below shows a typical vacuum waste storage tank installation:

Emptying

Large aircraft usually hold waste in a storage tank that is emptied after the aircraft has landed. The task of emptying the tanks at the end of the flight usually rests with the specialist companies, sub-

contracted to the airlines, they empty all waste tanks at particular airports.
The tanks are emptied in one of two methods, gravity or suction. The gravity method empties the tank, after the hose of the toilet emptying vehicle has been connected, by simply operating the shut-off valve. Once the tank is emptied, it is flushed out and, depending on the type of tank, replenished with deodorising fluid.

Suction requires both that the emptying vehicle has the correct equipment, (set at the correct suction value), and that the aircraft has ducting that is cleared for use with suction equipment. If the aircraft only has 'gravity' emptying ducting and piping, severe damage will be caused to much of the toilet equipment, if used with vacuum emptying equipment.



Corrosion Control

All the areas where toilet equipment is fitted must be protected against corrosion. The effect of many toilet chemicals on aluminium alloy aircraft structure is severe. All spillages must be neutralised and cleaned off as soon as possible, whilst thorough checks of all the areas of the aircraft that could be affected, must be inspected at regular intervals. Such areas could be the toilet floor itself and beneath that floor; the vicinity of the collector tank(s), around the draining/filling panel and anywhere else the corrosive fumes could affect the structure.
Some toilet units are enclosed in an anti corrosion tank. Any leaks would be self contained within this tank. The tank would be connected to the drain lines. The toilet floors may be made from composite materials to reduces the likelihood of corrosion damage. All connections in the service panel are sealed off when the service panel is closed.

13.20 Integrated Modular Avionics (ATA42)

Introduction

This section describes integrated modular avionics (IMA). As the name indicates, an aircraft with IMA has avionics systems that are integrated and modular. Integrated means, that multiple functions are combined into a single piece of equipment. Modular refers to a design method that allows the system to be divided into separate, replaceable modules. The term Avionics itself derives (torn aviation electronics, and it refers to electronic systems used in aircraft. Avionics encompasses a wide range of systems. Avionics is used for navigation, for communication, for control of the aircraft, and for other purposes. On a modern aircraft, there are dozens of systems that can be considered avionics systems.
Integrated modular avionics is a design methodology, not an avionics system per se. in other words, it describes the way the avionics systems are put together, not the specific functions of the individual avionics systems. IMA represents and advance in avionics technology. Aircraft with IMA can realize reductions in the bulk, apse, and weight of their avionics systems. Also, the overall reliability of the avionics can be improved, as less components are required.
Ease of fault isolation, and additional facilities such as BITE (built—in-test-equipment), is able to be achieved with the inherent architecture design. With aircraft systems having more software based functions, and computers becoming more powerful, computer requirements for equipment can be shared.
The following is just a partial list of functions that may be Integrated into an IMA System:

Bleed Management Air Pressure Control
Air Ventilation and Control
Avionics and Cockpit Ventilation Control Air Traffic Communication
Avionics Communication Router Electrical Load Management Circuit Breaker Monitoring
Electrical System Built In Test Equipment (BITE) Fuel Management
Braking Control
Steering Control
Landing Gear Extension and Retraction Tue Pressure Indication
Brake Temperature Monitoring Referring to the above list, IMA can be considered divided into three functions:
Cockpit (fly-by-wire electrical flight controls, communication and warning systems)
Utilities (incorporating the numerous systems above) Cabin (air-conditioning and other systems)

Design methods for avionics systems have evolved over time. From the late 1980's Business Jets have incorporated concepts of IMA, but on a smaller scale found in modern airliners. Military design developments for the F-22 and F-35 have also migrated into the commercial aviation field. Initially, avionics systems were discrete, stand-alone systems. This means that each system was separate.

For example, an aircraft's compass system might have consisted of a flux detector in the wing, a gyroscope in the avionics compartment, and a heading indicator on the instrument panel. These components were used only by the compass system. They were not shared by other systems on the aircraft. Although the compass system's components were connected to each other with wiring, the compass system itself was not normally connected to any of the other systems on the aircraft.
(An exception to this might be where an aircraft's auto-pilot has a Heading mode facility where compass heading data is fed as an outer loop source.)
Traditionally, on non-IMA aircraft, each avionics system had its own separate indicator and its own separate controls. As more and more avionics systems were developed and installed in aircraft, more indicators and controls had to be installed. Instrument panels became more complex and crowded. Figure 19 - 1 shows such an instrument panel. In addition, as more avionics systems were developed and installed, more LRUs (line replaceable units) or 'black boxes" were installed in avionics compartments. More wiring was needed to interconnect these LRUs with

their associated cockpit controls and indicators. More electrical power was needed to operate the systems.
Each additional indicator, LRU, and wire that is installed on an aircraft takes up space and adds weight. Because both space and weight-carrying capability are at a premium, it is desirable to keep the number of indicators, LRUs, and wires to a minimum. In the case of indicators, engineers began to develop designs that used the same indicator to display information from more than one system. For example, older designs had separate indicators for the compass system, the radio navigation system, and the weather radar system.
In newer designs, these systems are all connected to single,
"integrated' indicator such as a navigation display (ND). The use of integrated indicators saves space and weight, and it streamlines pilot workload by reducing the number of indicators that must be scanned during flight. Much more information can be displayed on one instrument, or in a smaller area, which ergonomically creates an easier workload for the pilot.
Figure 20-2 shows a modern, integrated instrument panel. The design concept that was first used to combine indicator functions in the

Figure 20-2. An instrument panel with fewer, integrated instruments.

cockpit has been carried further with integrated modular avionics. In an aircraft with IMA, the same concept - integration - is applied to many of the LRUs as well. Instead of individual, independent LRUs, an aircraft with IMA uses modules which are integrated into a single system. The modules perform the functions formerly performed by the independent LRUs, but they are not completely independent. They share circuitry.
Traditional (non-IMA) avionics suites have many separate LRUs (*black boxes) located in an avionics compartment. In such systems, there is a considerable amount of duplication among the black boxes. For

example, each black box in the avionics compartment typically contains its own power supply. These power supplies receive aircraft power and use it to provide the various voltages needed by the circuits within that box. Also, each power supply is connected to the aircraft's electrical power system by a separate wire.
These power supplies might be functionally identical to each other. In a non-IMA aircraft, there could be twenty identical power supplies in twenty separate boxes in the avionics compartment. However, if the power supply in one of the black boxes fails, that system fails because that box cannot use power from another black box. In an aircraft with integrated modular avionics, some of the self-contained black boxes arc replaced by modules. The modules form part of an integrated system because they are plugged into a mainframe or rack which is a single piece of equipment. This results in a bulk and weight reduction because some circuitry is now shared among the various modules.

For example, instead of having twenty duplicate, non-redundant power supplies, an IMA aircraft might have three redundant power supplies that are each capable of supplying all twenty modules. If one of the IMA power supplies fails, there are still two redundant power supplies that can power the modules. A failure of one or even two of the power supplies does not result in the failure of any of the avionics modules. Thus, the IMA aircraft is more fault-tolerant, resulting in better reliability.

The integration concept reduces the total number of LRUs needed. Often, the same kind of data processing circuitry is required for different avionics functions. In previous (non-IMA) designs, this data processing circuitry had to be duplicated in separate LRUs for each separate system. With 1MA, the data processing circuitry can be contained in fewer LRUs (modules), and it is shared among the systems that require it. This combining of functions results in an overall reduction in the number of separate LRUs. Less wiring is needed in an IMA aircraft. This is partially due to the integration of functions that eliminates unnecessary duplication of circuitry.
Using the example of the power supplies, with IMA there is no need to provide twenty separate wires from the aircraft electrical system far power supplies because there are only three power supplies for the IMA modules. In addition to this, the fact that IMA aircraft make extensive use of digital data buses also reduces the amount of wiring needed. (Figure 20-3)

DIGITAL DATA BUS USE

REDUCES WIRING

The use of digital data buses can result in a tremendous reduction in the amount of wiring used in an aircraft. Digital data buses are used to transfer information from one piece of equipment to another, using far fewer wires than were previously required. A number of different digital data bus systems are used in various aircraft. Some of the more important ones are ARINC 429, ARINC 629, and AFDX (Avionics Full Duplex). An example that illustrates this reduction in wiring is radio tuning. When a radio is tuned, frequency information must be transferred from a radio tuning unit to the radio receiver or transceiver that is being tuned. This frequency information might consist of Sam or more

digits.
Say a pilot wishes to tune a VHF communications transceiver to the frequency 128.35 MHz. The pilot enters this frequency into the tuning unit in the cockpit.
From there, it must be carried to the VHF transceiver located in the avionics compartment. (Figure 20-4)
With traditional avionics systems, those not using digital data buses, each piece of information to be transferred from one location to another requires at least one separate wire. Often, a far larger number of wires is needed. Using our example of a pilot tuning the VHF communications transceiver to 128.35 MHz, each digit of the selected frequency must be transferred from the tuning unit to the transceiver. Because there are five digits in this frequency, it might seem that five wires would be needed to transfer this data. However, because the information being transferred is complex (each digit might be anything from a zero through a nine), even more wires are needed. A typical pre-data-bus method of accomplishing this transfer was the ARINC "two out of five" or 2x5 tuning scheme.

Figure 20-5 shows the 2x5 tuning scheme. 2.5 tuning requires the use of five wires for each digit of information being transferred. Of these five wires, two are be connected to ground and three are not. The particular wires which are grounded determine whether the digit
transferred is a zero, a one, a two, etc. Since all VI-IF communication frequencies begin with the number "1," it is not necessary to transfer that digit from the tuning unit to the transceiver. However, each of the other four digits in the selected frequency must be transferred. Four digits at five wires per digit results in 20 wires. A ground wire is also needed, bringing the total number of tuning wires to 21.
Using a digital data bus like ARINC 429, this same tuning information can be transferred using only two wires. With ARINC 429, the two information-carrying wires are twisted together, and they are covered with a braided shield. The shield is usually kept grounded, and protects the two inner wires from electromagnetic interference. The two inner wires are referred to as a "twisted pair". The reason it is possible to transfer so much information on only two wires is that the information is sent serially. This means that the same two wires carry one bit of information at a time. One bit of information is sent, then another, and then another. Soon, all the required information has been transferred from the tuning unit to the radio. The information transfer is done at a rapid rate: ARINC 429 can transfer up to 100 000 bits of information each second. ARINC 629 can transfer up to two million bits per second. AFDX is a newer digital data bus system used that can transfer data at rates up to 100 Mb/sec. Fewer wires are needed for each successive developed data bus system and so a substantial weight savings is realized.
AFDX is the latest development for avionics data buses. It has been successfully developed for the A380 and B787 and will be used for the A350 and A400M aircraft. AFDX uses a star topology switching network and it's frame format is fully compliant with IEEE STD 802.3 (Ethernet). Relative to other formats, AFDX has characteristics of: good integration and ease of interfacing; reduction in cable interconnects and wiring; high bandwidth; high reliability and fault tolerances.

COMPUTER ARCHITECTURE
Examples of practical computer assemblies utilized by leading IMA design companies are listed below:
• Cabinet Modules, with each module connected to a BACKPLANE for intermodule communication, and gateways. (Honeywell AIMS)
• Cabinet of Cards; using a propriety DEOS operating system — commonly found in Business and Regional jets; (Honeywell Primus EPRIC)
• Independent Modules as LRU's; an AIMS derivative, (Honeywell VIA)

Using latest technology, design for IMA systems has been approached by designing the hardware around commercially available computer operating systems. This has created economic momentum in the building of IMA systems with less cost and complexity of applications. Much initial design of software and applications has been averted by employing the existing commercial systems that have been available. Ease of software upgrades has thus been also been ensured. Modern airliners invariably use either of two IMA concepts:

COMMON CORE SYSTEM (CCS) CONCEPT

This IMA system consists of a common core System with network components. The core system contains a central computer (dual computers for redundancy purposes) with data processing circuitry that processes many different kinds of information. This processing circuitry, is shared by the various avionics systems that have been integrated. Information from various sensors, controls, and LRU's is brought into the core system for processing, usually in a concentrated form, from data concentration modules. Data is then sent out from the core system to displays, actuators, and other places in the aircraft where that information is used.

The core system uses the same computer processors, or modules, for many different purposes. For example, a core system can use the same processor for such tasks.

As calculating throttle settings for best fuel economy, calculating the amount of rudder deflection needed to coordinate a turn, monitoring an instrument landing system receiver for malfunctions, determining if the

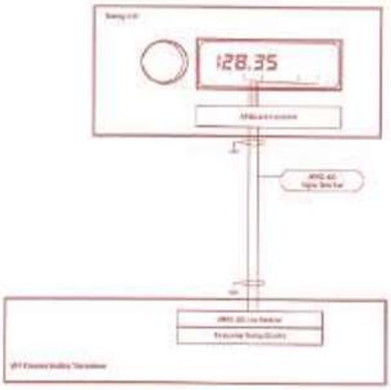


Figure 20-7. VHF Comm system tuning wires using ARINC 429 digital data bus.

stall warning system needs to be activated, and many others. The core system's processing power is shared among these various functions. This sharing eliminates the need to have a processor in each and every system.

The core system uses the same computer processors, or modules, for many different purposes. For example, a core system can use the same processor for such tasks as calculating throttle settings for best fuel economy, calculating the amount of rudder deflection needed to coordinate a turn, monitoring an instrument landing system receiver for malfunctions, determining if the stall warning system needs to be activated, and many others.

The core systems processing power is shared among these various functions. This sharing eliminates the need to have a processor in each and every system.

NETWORK COMPONENTS

Network components are the parts of the IMA system that allow data to be transferred into and out of the core system. These components include input and output devices and data bus wiring. Input devices receive data from a digital data bus and couple it to the core system processing circuits. Output devices prepare core system output data for transmission along a digital data bus. The exact properties of the input and output devices vary with the particular kind of data transfer system that is being used. As mentioned above, there are several different data transfer systems used in aircraft, some of the more common one being ARINC 429, ARINC 629, and AFDX. Figure 20-8 contains various data transfer formats that have become standard with specific systems.

BOEING 777 AIMS

Figure 20-8. Boeing 777 Airplane Information Management System (AIMS) cabinet.

An example of an aircraft that uses integrated modular avionics is the Boeing 777. In this aircraft, Boeing calls its IMA system the airplane information management system (AIMS). The core system of the 777 AIMS system is contained in two cabinets which are located in the aircraft's main equipment center.

Each of the AIMS cabinets has ten line replaceable modules (LRMs) which plug into it. These modules are of four different types. The cabinets themselves contain network components and circuitry that is shared by all the LRMs installed in each cabinet. This common circuitry is contained in a "backplane bus." The cabinets are also networked with other LRUs in the aircraft via digital data buses.

In the 777, the Line Replaceable Modules in the AIMS process information for the following avionics systems:

- Primary Display System (PDS),
- Flight Management Computer System (FMCS),
- Thrust Management Computer System (TMCS),
- Central Maintenance Computer System (CMCS),
- Airplane Condition Monitoring System (ACMS),
- Data Communication Management System (DCMS)
- Flight Data Recorder System (FDRS).

The AIMS is interconnected with many other units in the aircraft. It receives input data from these units, and it provides output data to them. (Figure 20-9) The AIMS uses several methods for sending and receiving information.

These include six different digital data transfer formats, analog data transfer, and wireless RF links. The primary data transfer format used by the AIMS is ARINC 629. The AIMS is connected to 66 LRUs located throughout the aircraft via ARINC 629. ARINC 629 uses a bi- directional data bus. This means that data can be sent both ways on the same data bus.

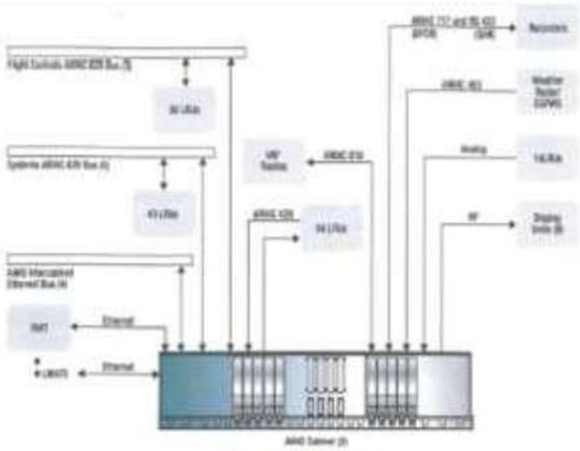
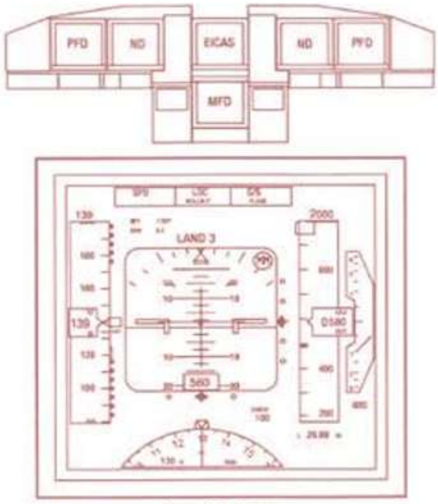


Figure 20-9. Boeing 777 AIMS interfaces.

The AIMS can both send information to an LRU, and receive information from that LRU, using a single data bus connected between the two. ARINC 629 uses a more rapid data transfer rate (up to 2 Mbps) than some other digital data transfer systems.

The AIMS also uses the ARINC 429 data format to communicate with 56 LRUs throughout the aircraft. Data transfer using ARINC 429 is unidirectional. This means that information can only be transferred in one direction on a single ARINC 429 bus. If an LRU needs to send information to the AIMS, one bus is required.

If the AIMS needs to send information back to that LRU, a second bus is required. ARINC 429 data transfer rates are somewhat slower than those used with ARINC 629. The transfer rates range from 12 kbps to 100kbps. ARINC 429 is a widely used data transfer system that was first developed in the 1970s. It is used in many makes and models of aircraft. The later developed Boeing 787 IMA system uses fibre-optic Ethernet that connects all systems with the CCS using AFDX/ARINC 664 protocol. Computer processors use ARINC 653 operating systems.



Primary Flight Display

Figure 20-11. Display Unit operating as a Primary Flight Display (PFD).

The later developed Boeing 787 IMA system uses fibre- optic Ethernet that connects all systems with the CCS using AFDX/ARINC 664 protocol. Computer processors use ARINC 653 operating systems.

The AIMS displays information on six flat-panel display units (DUs). (Figure 20-10) These units are located on the instrument panel in the cockpit. They show the information that has been gathered from the various systems connected to the AIMS, and which has been processed by the AIMS. These DUs are the main displays used by the flight crew.

The DUs give the following types of displays:

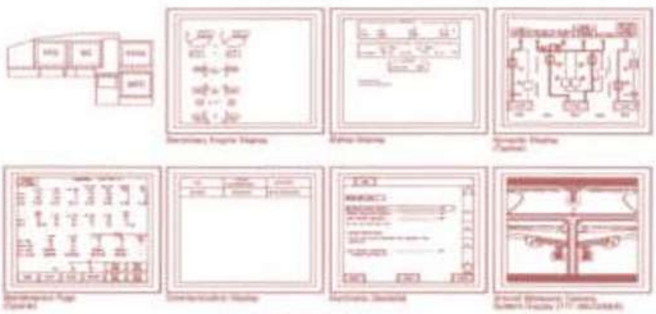


Figure 20-14. Multi Function Displays (MFDs)

Primary Flight Display (PFD) Navigation Display (ND)

Engine Indicating/Crew Alerting System (EICAS) Multifunction Display (MFD)
Switches are provided in the cockpit, that allows the flight crew to control the DUs. These switches allow a particular
type of display to be shown in different positions on the instrument panel. If one of the DUs fails, its function can be taken over by another functional DU. The system thus provides a great deal of flexibility.

Figure 20-11 shows a DU operating as a PFD. It shows aircraft pitch and roll attitude, airspeed, altitude, vertical speed, heading, and other information.

Figure 20-12 shows a DU operating as an ND. The ND can display data in four different modes. These modes are

VOR, Plan, Map, and Approach. The displays are laid out differently for each mode, and different information is displayed in different modes. The ND shows heading, VOR, DME, localizer, glideslope, TCAS, along with other information.

Figure 20-13 shows a DU operating as an EICAS display. This display shows engine operating parameters such as EPR, N1, LOT, etc. It shows fuel quantity,
landing gear position, flap/slat position, and other aircraft system information. It also provides warnings, cautions and advisories to the flight crew.
Figure 20-14 shows a DU operating as an MFD. The MFD can display many different kinds of information. Different screens (formats) can be selected. These formats are Secondary Engine Display, Status Display, Synoptic Display, Maintenance Page,
Communication Display, Electronic Checklist, and Ground Maneuver Camera Display. The Boeing 777 AIMS is one implementation of an integrated modular avionics system.
OPEN IMA CONCEPT
IMA concept was developed for the Airbus 380. It incorporates numerous computers with individual functions, connected in a network, hence it's not reliant on a single CCS to operate the

aircraft's equipment. Eight computer processing modules, having similar architecture, memory and power supply characteristics, are utilized in a network connection using AFDX (Avionics Full Duplex Switched Ethernet) - ARINC 664. AFDX is capable of 100 M bits/sec. and ifs operation is based on a star topology using common switched modules to transfer data between systems.
CPIOM (core processor input/output modules) using ARINC 653 for the internal data transfer format, are interfaced with each computer to host various applications, and provide signal acquisition and transmission. Each CPIOM is different, depending on the aircraft system ifs specific computer interfaces with. A single computing module requires levels of partitioning so that the performan
ce of each application system is not affected by any other. (Figure 20-
15)

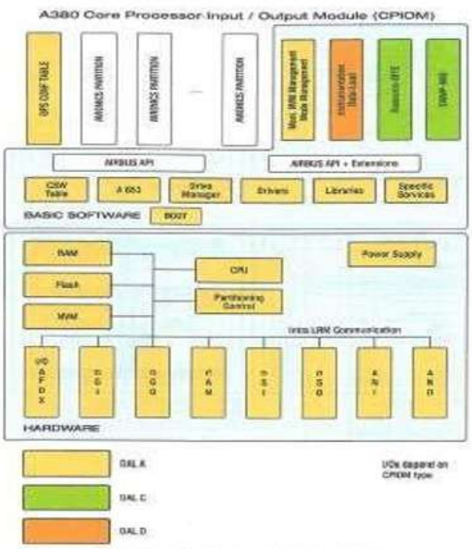


Figure 20-15. A380 basic netwwork system.

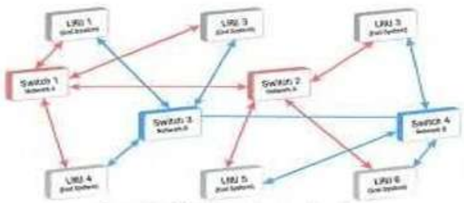


Figure 20-16. A380 basic network system and AFDX star topology system.

These partition boundaries are achieved by:

•Timing; each application has strict periodic allocation timing dependent on its importance
•Memory, a specific application can only access parts of memory that have been configured for it. Internal monitoring defect and prevent violations.
• I/O segregation provides allocation of channels and communication paths for each application.
Advantages of partitioning include:
System independence; thus ease of modifications and testing
Ease of configuration
Other aircraft's IMA systems vary somewhat in their specifications. The exact system architecture and the terminology used varies from one manufacturer to another, and from one model to another.
However, the basic features of IMA found in any IMA system are:

•Modular design
•Integration of modules within the system
•Use of digital data buses for interconnection
•Fault tolerance
•Flexibility

13.21 Cabin Systems (ATA44)

INTRODUCTION

Cabin Systems include those used to communicate both within the aircraft cabin and between the cabin and the cockpit. They also include systems for in-flight entertainment (IFE) of passengers, and systems used by passengers to communicate with the ground.
Like most aircraft systems, cabin systems have evolved and upgraded over the years. Older aircraft used analog electronics, while newer designs use digital electronics. In the use of IFE video, early systems used video projectors and large screens visible to all passengers. Video was stored on reels, and later on videocassettes. There was only one choice of video to watch. Newer aircraft are equipped with individual flat-panel video displays at each seat, and video is stored digitally. Passengers can select from many different video sources. Early systems used pneumatic headsets for passenger IFE audio; modern aircraft use electronic headsets.
As with all complex systems, there are variations in cabin systems. Different models of aircraft have different systems and different features. Some of the features that can be included in cabin systems are:

- Passenger Address - for addressing passengers through overhead cabin speakers.
- Cabin Interphone - for communication among the cabin crew members and also between the cabin and the flight deck.

- Flight Attendant Calling - allows passengers to call for assistance from the cabin crew, Audio and Video Entertainment.
- Moving Map Displays - showing the aircraft's current location, altitude, and airspeed.
- Telephone, Fax service, and Internet Service.

CABIN INTERCOMMUNICATION DATA SYSTEM

A cabin intercommunication data system is used by flight attendants, pilots, and passengers. Typical user interfaces for the system are flight attendant panels, cabin handsets, cockpit handsets, and flight attendant call buttons at passenger seats, here are also speakers and passenger information lights/signs.
Flight attendants can use the system to call each other in various pain of the cabin. For example, a flight attendant located at the galley at the front of the cabin can call another flight attendant who is at the rear of the cabin. Also, the pilots on the flight deck can call the flight attendants at their stations in the cabin, and vice versa. Flight attendants can make general announcements to the passengers. Passengers can activate flight attendant call lights. An example of a cabin intercommunication system is described below under 777 Cabin Services System.
PA announcements, made through Cabin speakers and passengers headsets, over-tide all other passengers audio and

entertainment. Pilot's PA announcements have priority over Cabin Crew PA announcements for obvious safety reasons. "Chimes‖ are a sound similar to a low frequency bell and are used to gain the attention of Cabin Crew though the cabin speakers.

CABIN NETWORK SERVICE

A cabin network service is a digital system that is typically hosted on a server within the aircraft. It provides services, such as access to email accounts and the internet. It may also provide access to data stored in databases. A cabin network service is essentially a computer Local Area Network (LAN) within the aircraft, one that can interface with external networks.

The interconnections within the LAN can be wired, or they can be wireless. The design trend is toward wireless connections. Because of the potential for interference with other systems, the use of wireless is restricted to certain phases of flight. Operation of wireless systems is not permitted during takeoffs and landings (refer to Wi-Fi section in this sub-module).
Although systems vary, and new features are added to each new version, typical uses of the cabin network service are for passengers to connect to the internet while in flight, and to access In Flight Entertainment (IFE). Infact, on some aircraft the cabin network service is integrated with the IFE into a

single system. Some systems also permit interconnection to satellite communication system for in-flight telephone calls.
Cabin Networks can transfer large volumes of data to many different locations by using multiplexing and de multiplexing techniques, which are described below under In Flight Entertainment. Multiplexing reduces the quantity of wiring needed, and allows advancedIFE generation systems to be easily developed and installed in aircraft.
All IFE systems fitted to modern airliners must be designed in accordance with relevant safety regulations. In the advent of failure, IFE systems must not interfere with the function of any other aircraft systems; they are isolated from other electrical systems and it is common for an IFE system master switch to be located in the cockpit.
Installation of IFE systems must therefore be carried out to ensure that the protection and integrity of all other aircraft systems is maintained and that failure of IFE system will not affect any safety aspect or function of the aircraft. Future concepts for IFE include live television reception with on demand capabilities.

CABIN CORE SYSTEM

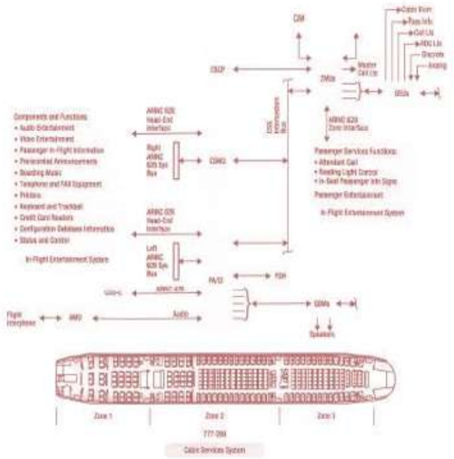
777 CABIN SERVICES SYSTEM

A specific example of an aircraft with a multi-featured cabin system is the Boeing 777. In the 777, the system is called the cabin services system (CSS). It includes both the cabin intercommunication data system and the cabin network service. Figure 21-1 shows a block diagram of the 777 Cabin Services Systems.
As can be seen in Figure 21-1, the system is digital and makes extensive use of digital data buses. The system uses three different data bus system: ARINC 629, the CSS Intersystem Bus, and ARINC 628. The ARINC 629 bus connects to the aircraft information management system (AIMS), which is the primary information management system for the aircraft. The CSS intersystem bus is a data Transfer bus internal to the cabin system. ARINC 628 is a data interface specifically designed for use with in-flight entertainment (IFE) systems. There are several available IFE systems foe an operator (Airline) to choose from, and the 777 can accommodate any IFE system that uses the ARINC 628 interface.
The central processor of the system is the cabin systems management unit (CSMU). This unit is connected to the aircraft's main ARINC 629 data bus. The CSMU connects to the IFE system through a different data bus - the ARINC 628 bus. There is also a passenger address /cabin interphone (PA/CI) controller, which the cabin crew uses to control the passenger address (PA) and cabin interphone (CI) functions. The cabin systems control panel (CSCP) is an interface used to load data into the system. The aircraft's cabin is divided into zones. Each zone has a zone management unit (ZMU), which is a processor connected to the rest of the system through a data bus called the 'CSS Intersystem Bus, Each ZMU is controlled by a cab in area control panel (CACP).

Cabin attendant handsets (CAHs) are connected to the ZMUs. These handsets are similar to telephones. They are used by the flight attendants to communicate with each other, and with the flight crew. They are also used for making announcements to the passengers. (Figure 21-2)
Each ZM is connected to a series of overhead electronic units (OEUs) within its zone. At each passenger

seat there are reading lights and attendant call lights. These lights are connected to the OEMs, which in turn are connected to the ZMU for the entire tone.

IN-FLIGHT ENTERTAINMENT (IFE) —BOEING 777



Each passenger seat in the 777 also has IFE equipment. The IFE equipment typically includes a video display an audio headphone jack, and controls for selecting specific video or audio content to be enjoyed by the passenger. It can also include other items, such as a credit card reader, or a port for connecting the passenger's own personal equipment to the system. The system may allow for connection of a device to the internet.

These functions are provided by a central source. This central source may be called the main IFE computer. As stated previously, various versions at IFE systems are available from various manufacturers, and their terminology and features differ somewhat.

In the 777 system, the main IFE computer is connected to the Cabin Services System via an ARINC 628 data bus. The data routes from the IFE computer, through the CSMU and the ZMUs, to the individual passenger seats. This is a two-way connection. Passenger requests for particular content (e.g. specific audio, video, games, etc) are sent to the main IFE computer, and the content is then sent from the main IFE computer to the passenger.

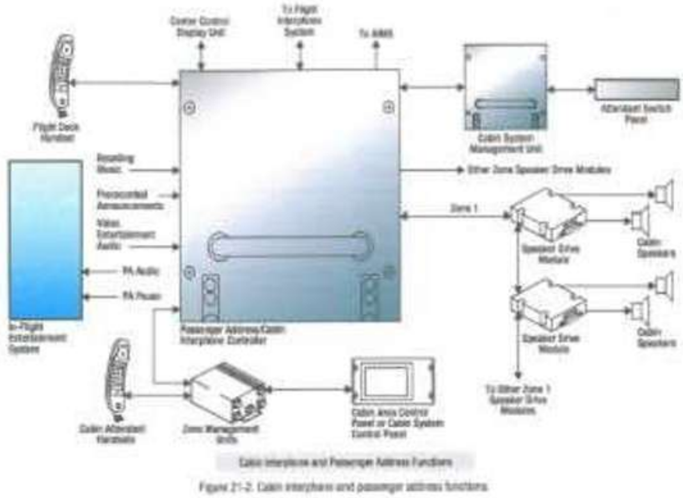
In flight, the main IFE computer is constantly receiving requests for different content from different passengers, and it must route the requested content to the correct passenger. This content, which may be audio, video, text, or something else, has been stored as digital data. It will be transferred as a data stream. The computer may be called upon to send out a large number of different data streams to the various passenger seats, all at the same time. This is accomplished by breaking up the data into small packets, attaching an address to each packet, then sending the packets at high speed down the data bus. The addresses indicate which passenger seat will receive which data packets. The data packets for all the seats are placed onto the data bus. The order in which they are sent does not matter, because the addresses will allow them to be sorted out at the receiving end. 'This process is known as multiplexing.

At the receiving end, the data packets are sorted by address. All the data packets labeled with a particular address (for a particular passenger) are reassembled. This is known as de-multiplexing. The data is converted back into audio, video, text, or whatever content it originally represented and delivered to the passenger. Each passenger receives only the data that was addressed to his or her seat. A high-speed digital data transfer system can multiplex and de-multiplex many channels, and carry information to many destinations simultaneously.

EXTERNAL COMMUNICATION SYSTEM

Although a large amount of data can be stored on board the aircraft and accessed by the cabin system, it is also useful to communicate with points outside the aircraft. This involves connecting with the Internet, and with the telephone system. To do this, the aircraft's external communication system is used to connect with either a ground-based network or a satellite-based network. There are several commercial networks available, and each of them has advantages and disadvantages. Changes and upgrades to these networks occur frequently.

Direct communication from the aircraft to the ground is used if the sender/receiver links are within line-of¬



sight range. Network systems automatically decide the routing at specific time communication is taking place. There are systems that are designed for just direct ground to air, for example, if an aircraft is operated just in one defined geographical location. Depending on the systems employed, ground based is faster than Satellite — it also must be expected that, generally, all air Internet services will be slower than ground domestic/business systems. However as technology contents moan& it can be expected that this difference in data speed, between systems, will progressively become less. Using two antennas on aircraft, one up-link and one down-link, enables the data speed to be dramatically increased.

For connecting with the external networks, the aircraft uses a router that is connected to dedicated radio transmitting and receiving equipment. This radio equipment uses various frequencies, depending on which external network is being accessed. Antenna size and shape also varies according to the network being used. For satellite-based systems, the antennas must be mounted to the upper surface of the aircraft, typically on the top of the fuselage or in the tail section. For ground-based systems, they will be mounted on the lower surface of the aircraft, usually on the belly of the fuselage.

Ground-based networks use radio towers to communicate with the aircraft. These towers have a limited range, so ground-based systems can provide coverage only over limited geographical areas. Ground-based systems tend to have a wide bandwidth, allowing for higher-speed data transfer. This makes web cages load faster, and results in good-quality audio and streaming video. Satellite networks, while providing a larger coverage area than ground-based networks, tend to operate at somewhat slower speeds, and with narrower bandwidths than ground-based networks. Several satellite systems are available for use by aircraft systems. The coverage areas and capabilities of these systems vary.

The Inmarsat system uses 4 geostationary satellites, which orbit at an altitude of approximately 22 000 miles (approximately
36 000 km) in equatorial orbits. The coverage is over about 95% over the earth,
with minimal coverage in the polar regions. It was the first system to be put into operation for voice/data links. Until it's uniform world-wide coverage is able to be utilized, other systems, such as Iridium, are generally preferred. However the Inmarsat system has very good reliability and connectivity and is relatively cheap.
The Iridium system uses low-earth orbit (LEO) satellites, which orbit at around 500 miles (800 km) altitude, and has world-wide coverage capability. Its characteristic low earth orbit ensures minimal voice/data delay, in comparison to geostationary higher orbit satellites.
Iridium utilizes a constellation of 66 satellites, evenly spaced in 6 orbital planes (11 satellites equally spaced in each plane) in near-polar orbits. Horizon-to-horizon time for each satellite is approximately 10 minutes — automatic linking, or hand-over, between satellites ensures continual data connectivity for users. The system operates in the L- band: 1616 to 1625.5 MHz for aircraft to satellites links. Cross-links between satellites is in the K Band: 23.18-23.38 GHz. Data is linked to the nearest satellite, and then relayed between satellites until the optimum, or overhead, satellite is able to link directly to the aircraft. A similar linking procedure occurs in the opposite direction.

The main advantage of the Iridium system is the world-wide coverage capability, but it can have disruptive connections and is more expensive, relative to the Inmarsat system.

CABIN MASS MEMORY SYSTEM

The cabin mass memory system is where the data that is used in the cabin is stored. The mass memory system can be programmed with cabin configuration information. This is information about the way the seats are configured. For example, which seats are first class, and how many zones the cabin is divided into. The cabin configuration programming varies, even the same make and model of aircraft can have several possible cabin configurations.

The mass memory system can store many types of data files. This includes audio and video files that are used for passenger briefing and announcements, and can include IFE such as pre-recorded music. An aircraft's mass memory system is accessible to the cabin crew through user interfaces (control panels). Although it is stored in a central location, data from the mass memory system can be transferred to access points around the cabin using digital data buses. The system can be updated when necessary through a data loading system.

CABIN MONITORING SYSTEM

Cabin monitoring systems are used to monitor various conditions in the passenger cabin. These conditions can include (among others): Cabin temperature, lighting, status of passenger warning lights such as "Fasten Seat Belts" and "No Smoking" lights, cabin entry and exit door status (open or closed), Status of smoke and fire detectors in troubleshooting and trend monitoring.

lavatories and elsewhere in the cabin, and Status of galley equipment, such as water tanks. Information about the conditions being monitored can be accessed by flight attendants in real time. It can also be stored and accessed later for purposes such as Cabin Wi-Fi, allows passengers to connect their own personal devices to an on-board Wi-Fi system. Such Wi-Fi systems are becoming more common, through installation on new aircraft and also through retrofitting older aircraft. Secure connections are achieved using encryption for passengers that require access to their own specific sensitive data. Satellite links are utilized for world-wide coverage.

Advantages of using WI-FI inside an aircraft cabin in comparison to hard-wired aircraft systems include:

- Higher speed, easier and improved access to data and information.
- Better accessibility throughout the aircraft cabin.
- Hard points are eliminated (Ethernet) for connection to the internet However, aircraft Wi-Fi service to the World Internet is still in its growing and developing phase, in technology and acceptable commercial models.

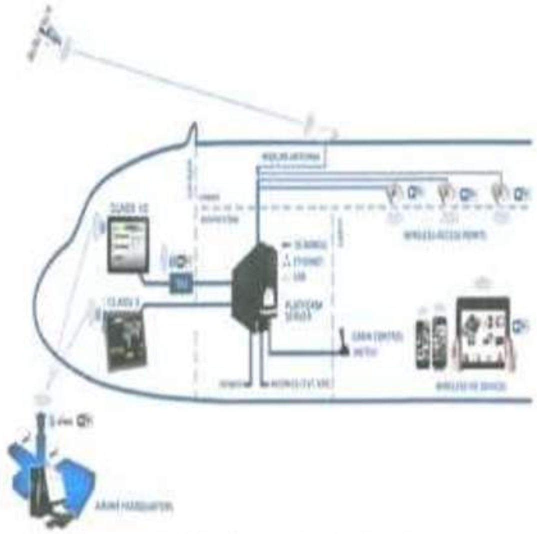


Figure 21-4. A typical components required for an installed aircraft Wi-Fi system.

MISCELLANEOUS CABIN SYSTEMS

Cabin systems, particularly in-flight entertainment systems, are the subject of intensive commercial development. New systems are being marketed, and some existing aircraft that were built without modern cabin systems are being retrofitted with them. Because these systems are highly visible to an airline's customers (its passengers), they can affect the customers' choice of airline. Since the airline industry is highly competitive, further refinements and improvements have recently occurred in the areas such as Cabin WI-FI.

AIRCRAFT WI-FI

Limited bandwidth is a major consideration, typically only 10Mbps, using a ground based system with dual modems and antennas. Note that this bandwidth is for the entire aircraft, not individual passengers, so obviously the general concept of video streaming is not yet an acceptable consideration. The bandwidth reduces considerably when satellite based systems are required, such as flying over the ocean. Future developments would include Satellites having expanded bandwidth capabilities in the Ka Frequency Band to accommodate data.

EXTERNAL CAMERAS

Outside video cameras are becoming a very popular feature on modern airliners. They essentially provide a real-time presentation of the aircraft's position for passengers. "Forward looking" and "down looking" cameras are commonly used and the video is usually incorporated in other flight data as part of the individual passenger network displays.

Figure 21-5 shows a typical external camera arrangement which is used on a rotorcraft for cargo sling operations. This underside cowling covers two separate cameras that can be switched for views of the: cargo attachment (lens shown); or looking directly downwards (lens hidden from view).

FLIR
Other applications of external cameras are FLIR (Forward Looking Infra Red). This equipment is able to take still photos, video, and utilize infra-red pictures for night vision. A dedicated crew member (FLIR operator) is invariably required for the operation of the equipment. Additional support equipment employed can be digital recording for onboard disc cards for analysis of pictures and video after flights. Applications for FLIR is mainly for law enforcement and defense agencies, for obvious reasons.
Figure 21-6 shows a FLIR module on the underside of a Rotorcraft near the nose. When switched off the camera lens is rotated back within the module for protection.

Figure 21-7 shows the operators position and associated equipment with viewing screen. Hand held controls (similar to commercial video

game apparatus complete the operating system for manipulating the FUR.

PICTORIAL MOVING MAP DISPLAYS

Other information available, as part of the flight data network, include Moving Map displays. These provide a real-time position of the aircraft relative to it's geographical location, with additional information such as: altitude, speed, direction, relative prominent ground features and similar data.

Figure 21-5. A typical FLIR external camera arrangement.

Figure 21-6. FLIR module on the underside of a Rotorcraft

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

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

INFORMATION SYSTEM INTRODUCTION

Aircraft operations and maintenance involve dealing with large quantities of information. This information must be stored in some manner, and, ideally, it should be capable of being accessed as efficiently as possible. Also, the information changes from time to time. To keep up with the changes, there must be a way to update the information stored in an information system.

Aircraft Information systems have evolved over the years. Initially, paper was the storage medium used. Manuals, drawings, charts, and other publications were available only in printed form. Information was retrieved by physically locating and reading the pages containing the information needed. Paper information sources have the advantage of being self-contained. No special equipment is needed to access the information. When revisions are required, new pages are printed and distributed. The old pages are removed and discarded, and are replaced by the new, revised pages.
Because large amounts of paper are bulky and heavy, other methods for storing large quantities of information were developed. Microfilm and microfiche were methods that saved space and weight. These methods involved using tiny photographic images of the pages contained in manuals, drawings, charts, etc. A disadvantage of microfilm and microfiche was that special equipment was needed to magnify the images so that they could be viewed. The microfilm or microfiche was useless if the viewing equipment malfunctioned, or if it was not available when the information needed to be accessed. Updates to microfilm and microfiche came in the form of new rolls of microfilm, or new microfiche sheets. After the new rolls or sheets were received, the old media was removed and discarded.

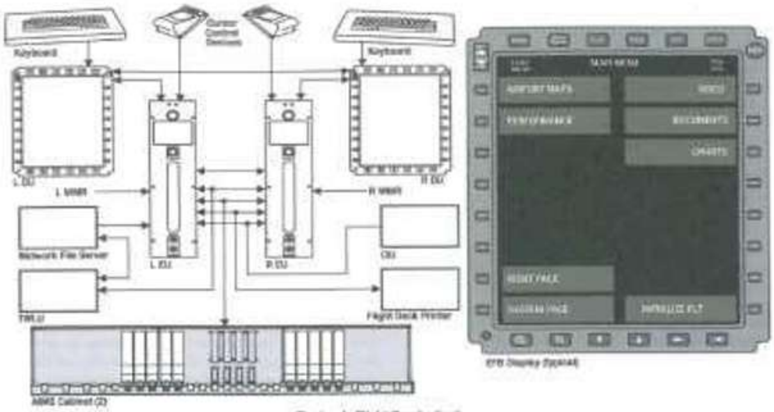
Digital computers represent a major advance in information technology. Modern computers can store and process very large amounts of information. They are compact, and they are very lightweight compared to other storage media, such as printed books. Computers store information in a type of memory known as RAM (random access memory) and ROM (read-only memory). This RAM and ROM can be internal to the computer, as in a hard drive. It can also be portable, contained in compact disks, flash drives, and other forms. The information contained in the digital memory can be accessed through display screens. It can be transferred in and out of the computer on wired data buses and wirelessly, and it can be printed. When updates are needed to the information contained in the system, the memory can be electronically erased and written over. There is no need to physically remove and replace paper or film. Aircraft information systems can be used to store and retrieve many kinds of information for various users. Examples are flight deck information systems, maintenance information systems, and passenger cabin information systems.

BOEING INFORMATION SYSTEMS

FLIGHT DECK B777 ELECTRONIC FLIGHT BAG

An example of a flight deck information system is the Electronic Flight Bag (EFB), an optional system used on the Boeing 777 and on other aircraft. (Figure 22-1)

The flight crew uses the EFB to access information that would traditionally have been found in various printed publications and carried aboard in a flight bag. Such publications include sectional charts and approach plates. The system also provides advanced capabilities beyond those available in printed publications.



Electronic Flight Bag (option)

Figure 22-1. Electronic flight bag.

On a 777 equipped with the EFB, the system displays information on the two Display Units (DUs) that are installed on the flight deck. One DU is for the Captain and one is for the First Officer. These two DUs are touch-screen devices, and operate independently of each other. The system can be navigated by the touch-screen DUs, and through keyboards and cursor-control devices. The DUs are connected to two electronics units (EUs) located on the airplane information management system (AIMS) rack in the main equipment center. The AIMS is an integrated system for processing information from many sources in the aircraft.

In the 777 system, the EUs send information stored in databases to the DUs through fiber-optic cables. The two DUs are connected to
each other through wired conn ectio ns. Data bases accessible through the EFB include aeronautical charts, airport maps and charts with real-time position monitoring, manuals, minimum equipment lists, and logbooks.

The system can also display video. The EFB system receives data from the AIMS and from multi-mode receivers (MMRs). MMRs are GPS and Instrument Landing System receivers designed for use in instrument landings. The MMRs provide aircraft position information that is extremely precise. This allows the EFB to pinpoint the aircraft's position on an airport map. The system also receives video from the Camera Interface Unit (CIU). This video is from the flight deck entry surveillance camera.

The 777 Electronic Flight Bags databases can be updated through the aircraft's data loader, but they can also be updated wirelessly. For wireless updates, the system uses a Terminal Wireless LAN Unit (TWLU). The TWLU contains a radio transmitter and receiver that create a Local Area Network (LAN) between the airplane and a ground-based network. This allows the databases to be updated wirelessly while the airplane is parked at the gate.

MAINTENANCE INFORMATION SYSTEM

The 777 EFB is one example of an aircraft information system. That particular system is used by the flight crew. However, the same kind of technology is also used by maintenance personnel. Flight crews need to access the EFB's information while in flight, but maintenance personnel need to access different kinds of information while the aircraft is on the ground. Maintenance Information Systems work along similar lines as a flight deck information system (such as the EFB), but the information being stored is different. Instead of maps and charts, maintenance crews use maintenance manuals, illustrated parts catalogs, wiring diagram manuals, service bulletins, and other technical data. Maintenance Information Systems provide access to these publications electronically, just as Flight Deck Information

Systems electronically provide the flight crew with access to maps and charts.
An advantage of using an electronic format, in addition to the space and weight savings it provides, is the ability to quickly locate the desired information. Instead of leafing through a large, paper maintenance manual, for example, a technician using a maintenance information system uses hyperlinks which allow easy navigation within the system. A typical method of achieving this is to use a menu containing links to each of the ATA 100 chapters within the manual. Within each chapter, the table of contents contains links that will quickly access a particular page. This allows the technician to locate the desired page with a few "clicks."
Aircraft manufacturers, which previously published their manuals only on paper or microfilm/microfiche, now offer their manuals in electronic format. Laptop computers are very well suited for storing and retrieving maintenance information in the aviation maintenance environment. Laptops can be taken practically anywhere on the aircraft that the technician might need to go while performing maintenance. For this reason, laptop computers are widely used to access maintenance information. (Figure 22-2)

PASSENGER CABIN INFORMATION SYSTEM

Flight attendants need to be able to access information to do their jobs, just as pilots and maintenance technicians do. In paper form, a typical flight attendant's manual weighs about five pounds (2.3 kg). Each flight attendant is required to carry the manual while

serving as cabin crew. The flight attendant's manual contains vital information for cabin crew members, such as checklists, procedures, security information, and information about safety devices and systems on the aircraft. Passenger cabin information systems provide the same advantages to the cabin crew that Flight Deck Information Systems provide to the flight crew, and that maintenance information systems provide to the maintenance crew. The use of the digital electronics allows flight attendants to carry a small device, such as a tablet, that contains the manual they are required to have on hand. (Figure 22-3)
The device weighs much less than a paper manual. This saves the airline money, as even small reductions in weight affect fuel economy. The airline also saves money on the cost of printing revisions to the flight attendant's manual.

AIRBUS INFORMATION SYSTEMS

The information system developed for the Airbus A380 is based on different system architecture than that used on Boeing airplanes however, its primary function is similar in that, it provides efficient data management for flight deck, maintenance and passenger cabin operations. The information system comprises two primary sub systems.
A Network Server System (NSS) serves as the primary platform hosting On-board Information System (OIS) software. The NSS divided into two parts, a secured area of data management known as the "Avionics world" and a less secure area of data management and exchange outside of the aircraft, such as communication with airline operations and service providers, known as the "Open world".
Data cannot flow from the Avionics World section into the less secure Open World. However Open World can flow into the secured Avionics World section of the NSS.
The second system is the Air Traffic Control (ATC) system which uses data link communications to transfer communication, navigation and surveillance data between the airplane and air traffic control services.

NETWORK SERVER SYSTEM (NSS)

The primary function of the NSS is to host the On¬board Information System (OIS) software using a high speed protocol to communicate with a network of servers and routers. The servers and routers provide a means to manage data

Figure 22-2. Laptop computers are used to access maintenance information.

Figure 22-3. Passenger Cabin information can be accessed with a computer tablet.

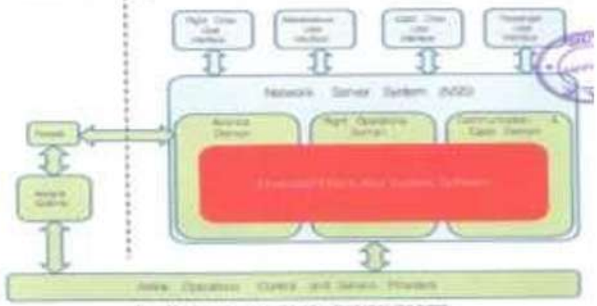
to and from user interfaces for flight crews, maintenance personnel, cabin crew and passengers. The user interfaces may be flight crew control panels and displays, maintenance terminals, cabin terminals and in-flight entertainment systems. The network server system also provides a means of receiving data from the secure avionics area via a firewall and communications externally with Airline control centers, such as flight operations and maintenance control, and with contracted service providers such as telecoms and e-mail.

The On-board Information System (OIS) software consists of a number of applications and document templates to replace paper documents and to increase operational efficiency both on the ground and in the air. The applications provide a means for the flight crews, maintenance crews and cabin crews to access data quickly and efficiently to allow for smooth operations and quick turn-around times for the airplane.

In addition, passengers are provided with access to e-mail, internet services and a number of entertainment channels including movies and games. The Network Server System (Figure 22-4) is divided into three areas known as 'Domains'. These are the Avionics, Flight Operations and Communication & Cabin Domains. The OIS software applications are also shared across these three domains

AVIONICS DOMAIN

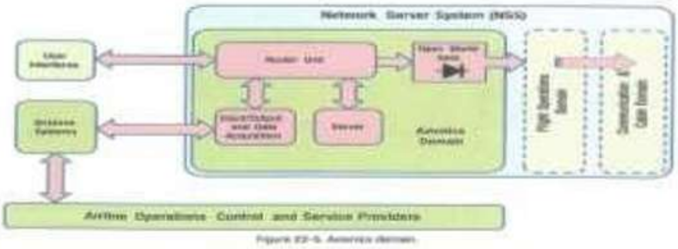
The avionics domain (Figure 22-5) consists of Network Servers, Router Units, Input/Output interfaces, Data Acquisition Units and a unidirectional data gate.
Dedicated processing units and memory devices provide computing and storage functions for the relevant OIS applications operating in the Avionics Domain. Data communication and acquisition of data from the 'Avionics world' is achieved via the input/output interfaces and the data acquisition units. This data is then routed to user interfaces, the NSS and recording devices where necessary.
Given the critical and sensitive nature of the data used in the 'Avionics world' any data processed and used within the avionics domain must be checked and 'cleaned' to ensure no malware



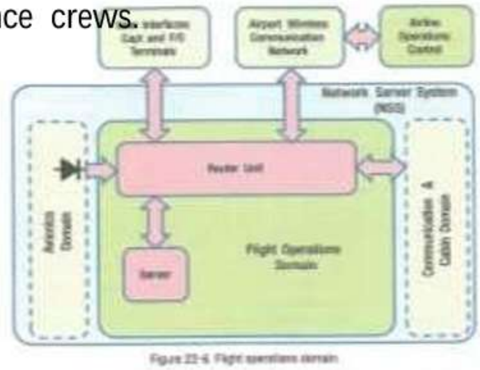
or virus is allowed to corrupt the system therefore, an additional function of the avionics domain is to 'virus check' and decontaminate any data received from and transmitted to the 'Avionics world' via the avionics domain.
The unidirectional gate ensures that data flows from the avionics domain to other domains but prevents data flowing into the avionics domain from the NSS, thus protecting aircraft avionic systems from any malware or virus that may exist in the 'Open world' environment.
Access to the avionics domain is achieved through Flight Deck terminals (Captain and First Officer), Maintenance
terminals, printers and data recorders. Limited access is also provided for Cabin Crew for communication and systems status (Doors armed/not armed, smoke and fire detection, etc.).
The On-board Information System applications used in the avionics domain provide on-board and air-to-ground communication functions, recording of aircraft data, electronic documents, maintenance and servicing operations within the 'Avionics world' and avionic domain.

FLIGHT OPERATIONS DOMAIN

The flight operations domain (Figure 22-6) consists of a server, router and terminals in the flight deck. As in the avionics domain the flight operations domain server contains the processor and memory devices for computation and storage facilities to support the relevant applications in the On-board Information System software suite. The flight deck terminals used by the captain and first officer consist of portable laptops connected via a docking station.



An airport wireless communication system also
provides a link with operational control centers whilst the airplane is on the ground. Access to the flight operations domain is provided by use of the Flight Deck terminals and/or portable laptop computers for captain, first officer and maintenance crews.



Universal Serial Bus (USB) ports are also available for loading data. Outputs can also be provided to an on-board dedicated printer for flight and maintenance crew use. The On- board Information System applications installed in the flight operations domain provide flight operations electronic documentation, airplane performance computation functions, navigation and communication functions, data management and maintenance operations associated with the flight operations domain.

COMMUNICATION AND CABIN DOMAIN

Like the other two domains the communication 8c cabin domain consists of network servers and routers but, unlike the other domains, it also controls a number of wireless units in the cabin area. The servers provide the processing and memory functions needed to support the 015 applications installed in the domain. (Figure 22-7)

The router manages data between the communications & cabin domain and the flight operations domain, server, on-board video surveillance systems, satellite communications systems (SATCOM) and passenger in-flight entertainment systems. The cabin installed wireless units are connected to antenna in the cabin and provide a wireless connection for portable electronic devices (laptops, tablets, smartphones, etc.) to enable access to electronic mail and internet services. The communication & cabin domain is accessed from the airplane cabin via a cabin terminal used by cabin crew and maintenance crews. OIS applications hosted by the communication & cabin domain include data management for data transferred between the airplane and operations centers, maintenance functions and passenger services (internet and electronic mail).

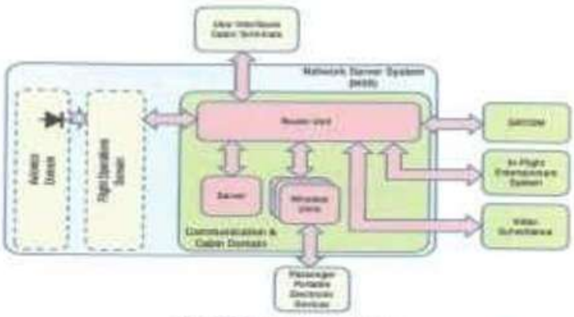
AIR TRAFFIC AND INFORMATION MANAGEMENT SYSTEM

Until recently the primary means of communication between flight crews and ground air traffic services have been by the use of voice communication systems such as High Frequency (HF) and Very High Frequency (VHF) Amplitude Modulated (AM) radio.

Although effective,

these systems have traditionally suffered from the effects of atmospheric interference/ disturbance resulting in

poor and unreadable communications. The bandwidths used by these systems were such that the number of available channels was limited. As air traffic increased to the level we see today it was only inevitable that the 'system would become overburdened and there would not be enough channels to support an efficient data exchange in today congested airspace.



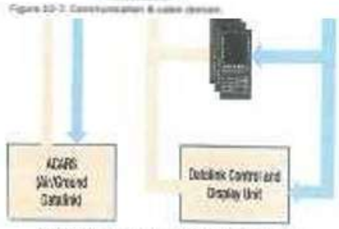


Figure 22-8. Air traffic information management system

The development of High Frequency Data Link (HFDL) and Very High Frequency Data Link (VDL) systems with much narrower

bandwidths means that there are many more channels available which are not prone to the levels of interference as the traditional AM systems. These systems provide the means to transfer data more efficiently, accurately and quickly. This also enables air traffic services to handle busier and more congested airspaces.

The Air Traffic Information Management System (ATIMS) (Figure 22-8) developed by Airbus Industries provides a means for a swift transfer of data via data link between airplane and ground stations. Air traffic