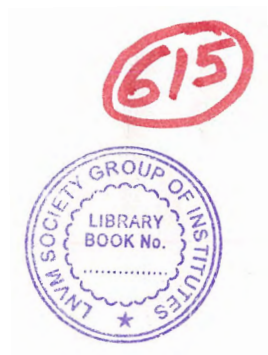# Automatic Flight Control

Automatic Flight Control
Fourth Edition

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Blackwell
Science

Preface to First Edition

At the present time there is hardly an aircraft in either civil or military operation without some form of automatic flight control system comprising part of its standard operational equipment. The systems available are as diverse as the aircraft themselves, varying from a simple roll stabiliser or 'wing-leveller' in a single-engined private aircraft, to the sophisticated flight-guidance systems capable of automatically controlling the flight paths of large transport aircraft from take-off to touchdown and roll-out. It is then a little difficult perhaps to realise that the development of such systems has arisen from foundations laid years before man himself took to the air to become the controller of his own 'flight path destiny'.

The early inventors of 'heavier-than-air flying machines' were, of course, faced with many problems, the most prominent of which was the one associated with the attainment of stabilised flight. Although there was an awareness that stability should be inherent in the basic design of a machine, little was known of the separation of stability into dynamic and static elements in relation to the various degrees of freedom possessed by a machine. As a result, and as recorded history indicates, efforts were directed more towards keeping a machine straight and level and free from the effects of external disturbances, and to derive the requisite stability by applying some form of artificial stabilising device.

It is of interest to note that possibly the first machine to use such a device was an unmanned glider designed by the Frenchman Charles Renard in 1873. The device consisted of a transverse pendulum coupled to two 'steering wings', the idea being that if the machine turned from its intended flight path, the pendulum would raise one wing and lower the other, and thereby straighten the machine's path. The first flight test indicated that such a device could work, but that lateral instability would have to be much less than that exhibited by Renard's machine to be really successful! Apart from the pendulum, the stabilising properties of a gyroscope were also considered, and a noteworthy 'first' in this connection was the stabiliser patented in 1891 by Sir Hiram Maxim and installed in his steam-powered machine. The design concept was somewhat ahead of its time in that it also comprised a servo control loop and other features which are basic to today's

automatic flight control systems. Maxim's flying machine unfortunately, came to an untimely end before the stabiliser could be tested under 'live' conditions.

When later pioneers took up the challenge of designing machines in which they themselves ventured to fly, the possibility of manoeuvring their machines away from straight and level flight was realised. However, this was to present another problem; namely, how to cater for the changes in stability which would result when control for initiating a manoeuvre was applied. Thus, 'controllability' was to become an important feature of flying machine design, and one which the Wright brothers were to incorporate in the machine which gained for them the distinction of making that historic flight in 1903. The Wrights' approach to aerodynamic and in-flight problems was more advanced than that of their predecessors, and although the machines built and flown by them were not completely stable, the incorporation of the controllability feature permitted a number of successful flights to be made without artificial stabilisation.

The introduction of control systems by the Wright brothers and sub-sequent pioneers in their aeroplanes (as they were becoming known) was to establish an additional role for stabilisation devices to play because, if a device could be coupled to the controls, then it alone could correct any departure from a stabilised condition. This was not to go unchallenged of course, and the first practical demonstration of a coupled gyroscopic two-axis control device was given by Lawrence Sperry during his historic flight in Paris in 1914. Thus, it can be said that the foundation for automatically-controlled flight was laid in the early years of this century. By the mid-twenties and in the 'thirties', the development of systems in the United States, the United Kingdom and Europe, became a separate field of engineering technology, and a number of 'automatic pilots' and 'gyropilots' demonstrated their capabilities in commercial and military aircraft oper-ations, and in several historic long-distance record flights. As the technology has continued to develop, system designs have been influenced not only by the advances made in aerodynamics and aircraft controllability charac-teristics, but also by the advances taking place in other technological fields. For example, the changeover from pneumatic operation of gyroscopes to electrical operation; the processing of control signals by electron tubes and magnetic amplifiers; the introduction of the semiconductor; and, perhaps the greatest influence of all at this moment in time, the vast potential of digital processing technology.

The diversity of present-day automatic flight control systems arises prin-cipally because they need tailoring to suit the aerodynamic and flight handling characteristics of individual types of aircraft. It is possible to compromise, and by virtue of this, many of the systems installed in aircraft designed for operation in the general aviation sector are, in fact, highly versatile in their applications; however, there are limitations particularly where the more complex types of transport aircraft are concerned. Thus,

any attempts at describing the range of systems and their operating fundamentals would be a mammoth task involving the writing of several volumes. However, any one automatic flight control system may be considered as being composed of four principal elements, which although differing in design and construction, perform functions common to all other control systems. The element functions concerned are progressively: attitude sensing, error signal sensing, signal processing, and conversion of processed signals into powered control, and they set a convenient pattern for a general study of control fundamentals. The material for this book has, therefore, been structured accordingly, and it is hoped that the selected examples of devices performing such functions, will usefully illustrate how relevant principles are applied.

A basic understanding of the priciples of flight and aircraft stability, and of servomechanisms, is a pre-requisite to a study of the main subject and they are therefore covered in the opening chapters. With the development of flight director systems and of the concept of integrating basic attitude and navigational data, it became logical to share data and servomechanism links such that a director system could provide guidance commands to an automatic flight control system. Thus, manufacturers develop and make available a wide range of complementary systems, the basic principles of which have also been included in this book. Chapter 10 deals with what may be termed the ultimate in automatic flight control evolution, namely automatic landing and autothrottle systems.

In preparing the material on systems, I have been greatly assisted by data and illustrations supplied by manufacturers, and would in particular, like to express grateful thanks to Collins Radio Company of England Ltd, Smith's Industries, Marconi Avionics Ltd, and Sperry Rand Ltd, for their permission to use certain of the data, and to have photographs reproduced. My thanks are also extended to friends and colleagues for useful suggestions, comments and assistance in proof reading, and finally to the publisher's editoral staff for their patience.

Copthorne   E.P.
Sussex

Preface to the Fourth Edition

The continuing demand for this book has been most encouraging, and it has therefore, been particularly gratifying to meet the publisher's requirement for the production of this, the fourth edition.

Since the book was first published in 1979, considerable technological advancement of systems designed for the automatic control of aircraft has, inevitably, taken place. As is generally the case however, such advancement has been in the methods by which established fundamental principles are applied. It is hoped that the coverage of these principles in previous editions has been of help to those readers who, having encountered practical examples of 'new applications technology' in the course of their particular specialisation in aviation, have had to gain more detailed knowledge of such applications.

In preparing the material for this edition, it was considered appropriate to retain the same sequence of chapters, in accordance with the functions of the principal elements that comprise any one type of control system.

New information has been added on

- principles of flight
- automatic control of helicopters
- autothrottle control
- digital computer-based systems.

Information on the fundamentals of fly-by-wire control systems has also been expanded and now forms the subject of a new chapter.

The authors would like to thank the following manufacturers for suppling new illustrations for this edition: SFIM, Honeywell, Airbus Industrie, and Westland Helicopters.

Copthorne, Sussex   E.P.
Roade, Northampton   S.C.

Contents

1
Principles of Flight

In order to understand the operating fundamentals of any automatic flight control system (AFCS) and its application to an aircraft, it is first necessary to have some understanding of how an aircraft flies, its stability characteristics, and of the conventional means by which it is controlled. There are two classes of aircraft with which we are concerned, namely fixed-wing and rotary wing or helicopter class, and the contents of this chapter are therefore set out under these appropriate headings.

A. FIXED-WING AIRCRAFT

Lift

It is a well-known fact from common experience that all material objects are attracted to the earth by a force which is in proportion to the mass of the object; such a force is called gravity. In order for an object to rise from the earth's surface, and to maintain itself in a continual ascent or at a constant height above the surface, the attraction which gravity has for the object must be opposed by the development of a force called lift. A variety of methods can of course be adopted, the choice being dependent on the object to be lifted. The method with which we are concerned, however, is the one applied to the wings of an aircraft. In this method, wings are designed so that they conform to specific plan forms, and aerofoil-shaped cross-sections, chosen on the basis of size, weight and performance requirements of the particular aircraft. The geometry of some typical wing plan forms, aerofoil cross-sections, and associated terminology are shown in fig. 1.1.

In order to generate the required lifting force there must be relative movement between the wing and the surrounding air. Theoretically, it makes no difference whether air flows over a stationary wing or whether the wing is moved through the air; in practice, however, the latter movement takes place as a result of the propulsive thrust from a propeller or turbine engine exhaust gases.

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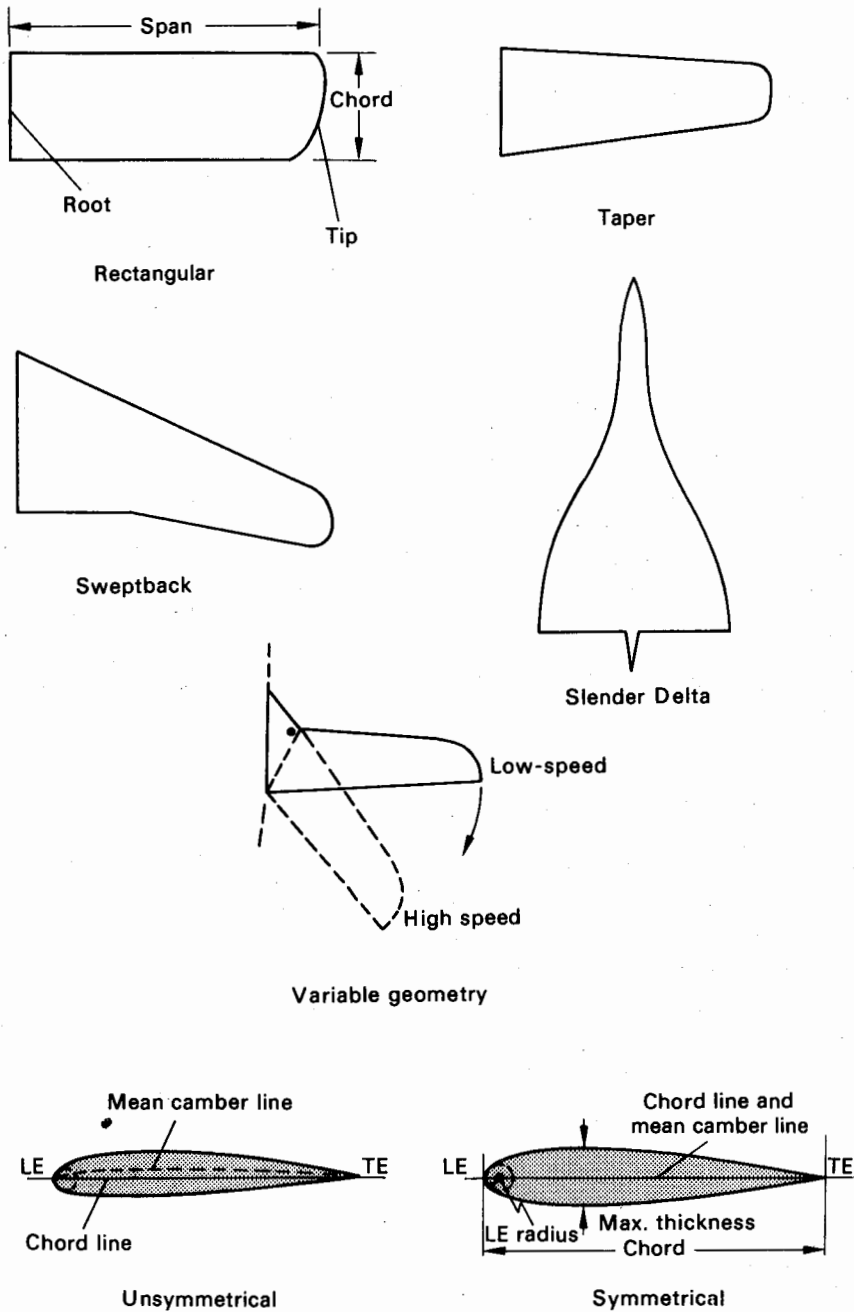
Fig. 1.1 Wing planforms and aerofoil terminology.
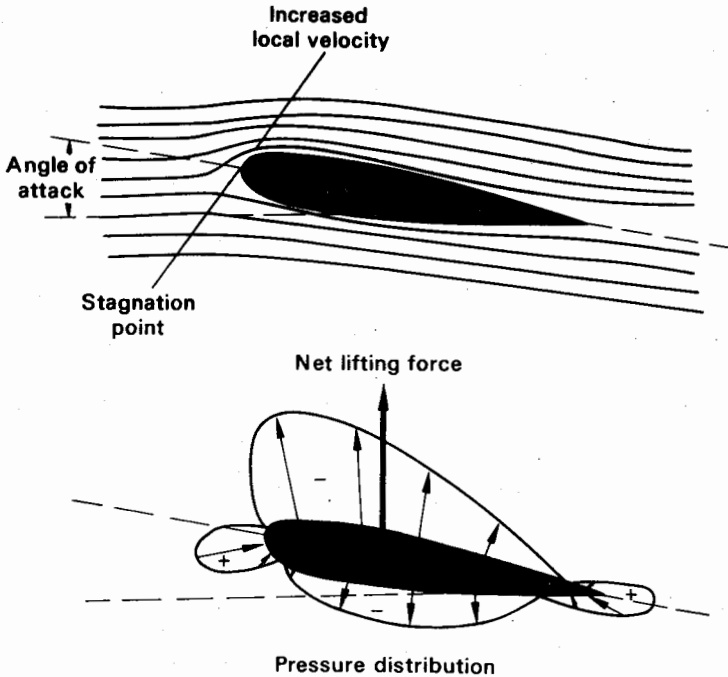
Fig. 1.2 Generation of lift.

Referring to fig. 1.2 it will be noted that when the air strikes the leading edge of the wing, it divides into a flow over the upper and lower cambered surfaces of its aerofoil section. The mass of continuity of flow is constant, but as a result of differences between the amount of upper and lower surface camber, and also because the wing is at an angle of attack, i.e. at an angle relative to the airflow, the velocity of the airflow over the upper surface will be greater than that of the air flowing along the lower surface. Since the pressure of fluid (liquid or gas) decreases at points where the velocity of the fluid increases, then for an aircraft wing at small angles of attack the pressures acting on both surfaces of the wing will decrease. However, the decrease is greater on the more highly cambered upper surface, and it is the resulting pressure difference across the wing aerofoil section which generates the net lifting force. The greatest pressures occur at the stagnation point, at points around the leading edge, and at the trailing edge.

The measurement of the pressures acting on the surfaces are in absolute values, and they are represented by vectors drawn perpendicular to the surfaces. The length of a vector is proportional to the difference between absolute pressure at a point and free stream static pressure. It is usual to convert this to a non-directional quantity called the pressure coefficient

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by comparing it with free stream dynamic pressure. The convention for plotting these coefficients is (i) measured pressure higher than ambient gives a positive coefficient, and the vector is plotted towards the surface; (ii) measured pressure lower than ambient, the coefficient is negative, and the vector is plotted away from the surface.

From the foregoing it is apparent that variations in angle of attack are an important factor in controlling the magnitude of the lift generated by a wing. For example, when angle of attack is increased the velocity of airflow over the upper surface increases at a faster rate than that over the lower surface, thereby changing the pressure distribution such that the net lifting force is further increased. At some critical angle of attack, called the stalling angle, the airflow separates from the upper surface and becomes turbulent, with the result that the lifting force is drastically reduced. In practice, the wings of each type of aircraft are fixed at an optimum angle of the chord line relative to a longitudinal datum (generally called the 'rigging angle of incidence') and the aircraft then flown within a small working range of angles of attack so that in combination the highest lift/drag ratio and economic performance may be obtained.

Other important factors which control pressure distribution and lift are the velocity of the free air flow, its viscosity and its density, the shape and thickness of the aerofoil section adopted for a wing, the wing plan form and its area, and condition of wing surfaces.

Delta wing

The lift coefficient (C_L) of a delta wing continues to increase up to very steep angles of attack because the system of leading edge vortices strengthens as angle of attack increases. The form of the vortex system of a wing with a large angle of sweep is shown in fig. 1.3.

When the wing is at zero angle of attack, the airflow remains attached to both surfaces of the wing and no lift is generated. As soon as the angle of attack departs from zero, the flow separates along the entire length of the leading edges in the form of two free vortex layers joined to the leading edges and rolled up in the manner of two conically-shaped spiralling coils above the upper surface of the wing. The coils, or leading edge vortices, induce a suction on the upper surface which remains constant along the chord except in the neighbourhood of the trailing edge. The size of the vortices increases with angle of attack and they cover a progressively greater proportion of the wing surface. When the angle of sweep is sufficient, these vortices remain in a broadly similar form through a wide range of angles of attack and the flow is characteristically steady throughout the range appropriate to the required flight conditions. Secondary vortices flow between the leading edges and the cores of the main vortices, but because
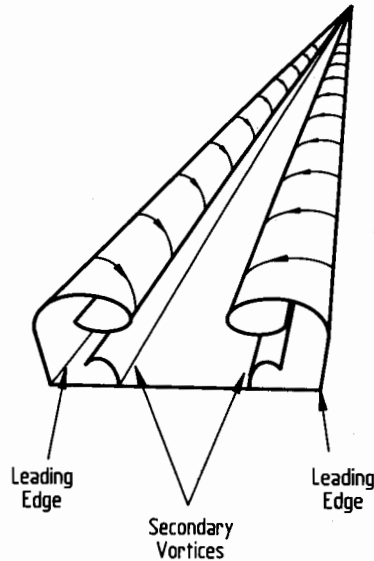
Fig. 1.3 Leading edge vortices.

these also develop progressively they do not interfere with the stability of the main flow.

As the vortices increase with angle of attack, the suction force generated on the upper surface of the wing increases and thereby contributes to the total lift produced.

The slender delta plan form shown in fig. 1.1 is of the type adopted for the 'Concorde'. The curved shape of the wing is such that maximum sweepback is obtained at the inboard sections of the wing without destroying the best effects of the vortex system.

Centre of pressure

In connection with the pressure variations occurring across the surfaces of a wing, it is usual to consider the total lift force as acting from one point along the chord line; this point is known as the centre of pressure (CP). As will be noted from fig. 1.4 a, the total lift force is resolved into two principal components: (i) the lift component acting at right angles to the direction of the free airflow, and (ii) a total drag component acting in the direction of the free airflow. The ratio of lift to drag is a measure of the efficiency of any aerofoil section adopted for an aircraft wing.

The location of the CP is a function of camber and the factor known as the lift coefficient, and it varies with the angle of attack. As the angle of

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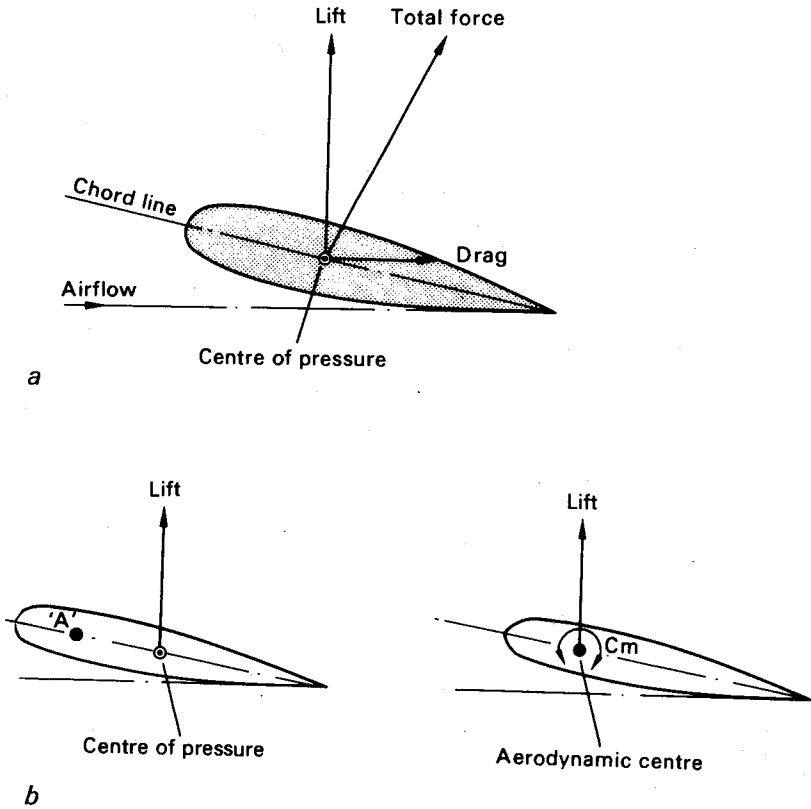
Fig. 1.4 Centre of pressure and aerodynamic centre.

attack increases, there is a change in the distribution of pressure above and below the wing such that the magnitude of lift force increases and the CP moves forward. At a certain angle of attack, known as the stalling angle, there is a sudden decrease in the magnitude of the lift force and the CP moves rearward.

Aerodynamic centre

Movement of the CP with changes in angle of attack also causes the pitching moment of a wing to vary, to an extent which depends on the position of the moment reference point 'A'. The pitching moment is equal to the product of the total lift force and the distance from the point 'A' to the CP (fig. 1.4 *b*). It is, however, possible to locate a reference point about which the pitching moment is constant (C_m) regardless of the angle of attack. Such a point is known as the aerodynamic centre, and for flight at

subsonic speeds, it is usually located at or near 25% of the chord. In the mathematical treatment of stability and control of aircraft, allowance is made for the constant pitching moment and it is assumed therefore that the total lift force acts from the aerodynamic centre rather than the CP.

Drag

The movement of a body through a fluid, whether it is a liquid or air, always produces a force that tends to oppose the movement; such a force is known as drag. Thus the wings of an aircraft, and all its other structural parts exposed to the airflow, experience components of a total drag which must be reduced to a minimum. The drag components arise in several different ways and they can be considered as constituting two principal types of drag, i.e. profile, and induced or vortex; these are summarised in tabular form in fig. 1.5.
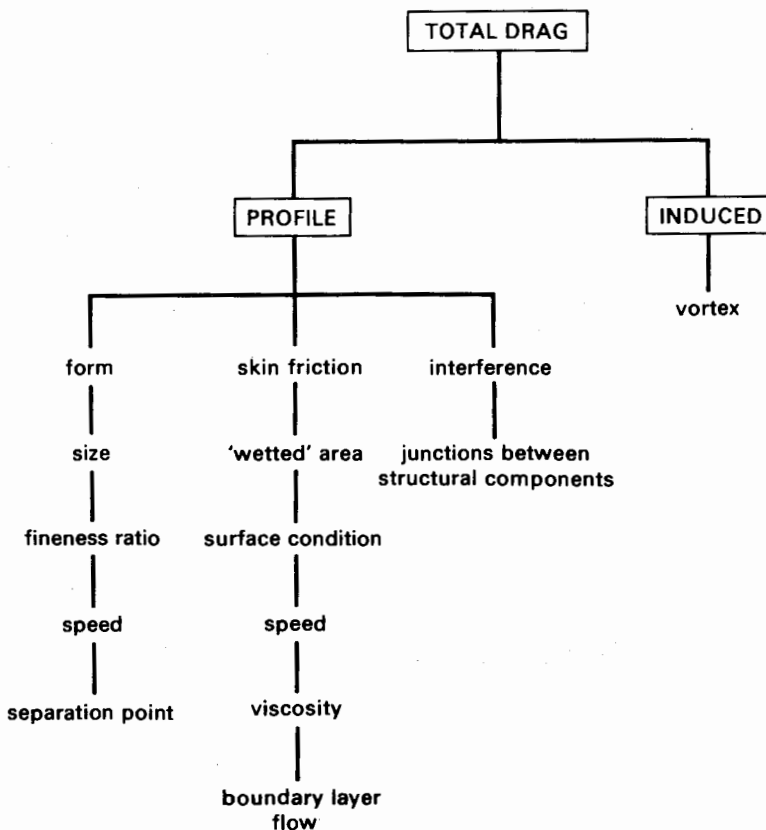
Fig. 1.5 Total drag of an aircraft.

Profile drag

Profile drag is composed of the drag components produced by the surface or skin friction created when a body is exposed to airflow, and also by the form or shape of the body. A controlling factor in determining, among other things, the nature of these components, is the very thin layer of air extending from the surface of the body, and referred to as the *boundary layer*. The whole surface area of an aircraft has a boundary layer and therefore has surface friction drag.

If the streamlines of an airflow over the wing of an aircraft are considered as the boundary lines between layers of air, then because air has viscosity, variations in the velocity of each layer will occur as a result of viscous adhesion. Such variations are governed by the distance from the wings' surface, and also by the condition of the surface, i.e. whether it is rough or smooth. The layer adjacent to the surface will adhere to it and so its velocity will approximate to that of the wing. The viscous adhesion between
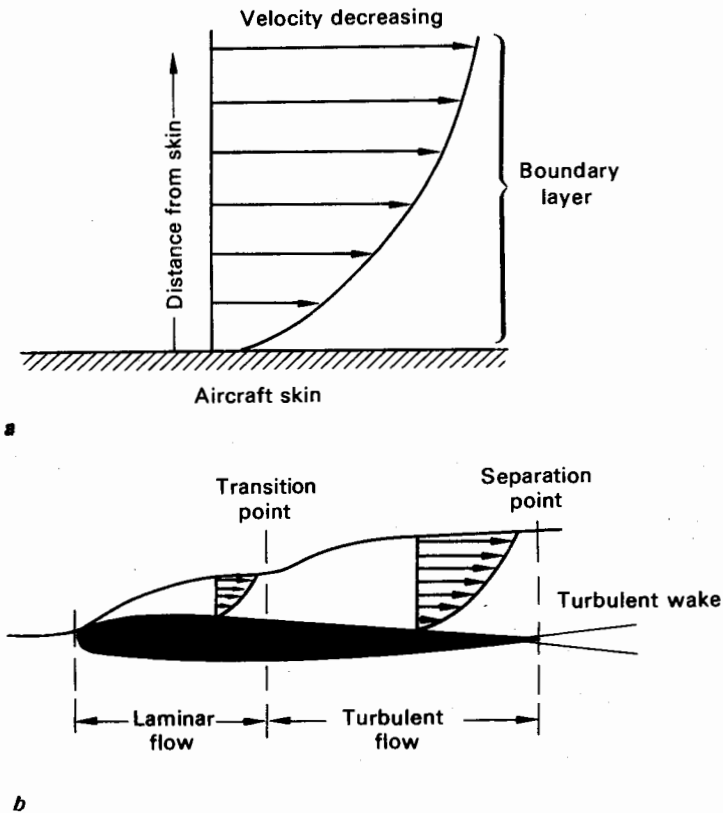
Fig. 1.6 Boundary layer.

this layer and the one above it will cause the second layer to flow in the direction of wing movement, but at a slightly lower velocity. Similarly, the velocity of the adjacent layers will be lowered until a point is reached where the movement of the wing causes no movement whatsoever of layers of air at some distance 'd' from the wing surface (see fig. 1.6 *a*). Thus, boundary layer may be more closely defined as the layer of air extending from a surface to the point where no viscous drag forces are discernible.

Boundary layer airflow may be either laminar, i.e. streamline, or turbulent as shown in fig. 1.6 *b*. Usually the airflow starts by being laminar over the forward part of the surface, and then at some point, called the *transition point*, the layer tends to break away from the surface and becomes turbulent. The turbulent air mixes with the air above the boundary layer causing a thickening and spreading out of the layer and, as this increases the distance at which viscous drag forces can act, surface friction drag will accordingly increase. Eventually, at a point close to the trailing edge of the wing, the boundary layer separates from the surface resulting in a wake of turbulent air. The separation depends on the rate at which the pressure changes around the body, the rate of pressure change in turn depending on the shape of the body.

The position of the transition point in an airflow of a given density and viscosity depends on the velocity of the airflow and the thickness of the body in the airflow. When applied to a wing of a given thickness, an increase of velocity causes the transition point to move towards the leading edge with the result that more of the wing surface is covered by a turbulent boundary layer and so surface friction drag is further increased. However, a turbulent layer has more kinetic energy than a laminar layer and, since this has the effect of delaying boundary layer separation, the maximum value of lift coefficient is increased.

Form drag

This type of drag, as the name suggests, is dependent on the shape of the body exposed to the airflow and as noted earlier, the body shape governs the boundary layer separation and the rate at which the pressure around the body changes. For this reason, therefore, form drag is also referred to as boundary layer normal pressure drag. In order to appreciate the difference between surface friction drag and form drag, let us consider for a moment that the body exposed to the airflow is in the form of a very thin plate. When the plate is at zero angle of attack with respect to the airflow the direction of the airflow will not be materially changed and neither will the velocity or pressure. Thus the boundary layer in this case is purely laminar and the drag results solely from surface friction. When the plate is set at an angle of attack it will cause a change in airflow direction, velocity and pressure, so that the boundary layer now becomes turbulent and begins to separate from the upper surface of the plate. If the angle of attack is further

increased such that the total surface area of the plate is presented to the airflow then there is a complete breakdown of the boundary layer and the drag is wholly form drag.

Interference drag

Interference drag is a result of disturbances to the airflow over an aircraft by the many junctions between major parts of its structure, e.g. between wings and fuselage, engine nacelles and wings. They can all cause changes in the pressure distribution and early separation of the boundary layer.

Induced drag

When a wing is producing lift, the airflow over both the upper and lower surfaces join at the trailing edge, and leave it in the form of a vortex motion the direction of which imparts a downward velocity component to the air. This downwash, as it is called, has the effect of inclining the lift force rearwards so that it will have a component acting in the direction of the drag force. This additional drag component is called the induced or vortex drag, and is affected by such main factors as plan form and aspect ratio of a wing, lift and weight, and speed of the aircraft.

One method of reducing vortex drag is to fit what are termed 'winglets' such that they are virtually wing-tip extensions turned through an angle compatible with the aerodynamic characteristics of the aircraft concerned. An example of the method as applied to a current type of swept-wing aircraft is shown in fig. 1.7.

Aircraft stability

Stability is the property of a system whereby the latter returns to a state of equilibrium after it has been displaced from a state of rest or a state of

Fig. 1.7 Gulfstream III Corporate Jet (reproduced by courtesy of *Grumman Aerospace Corporation*).
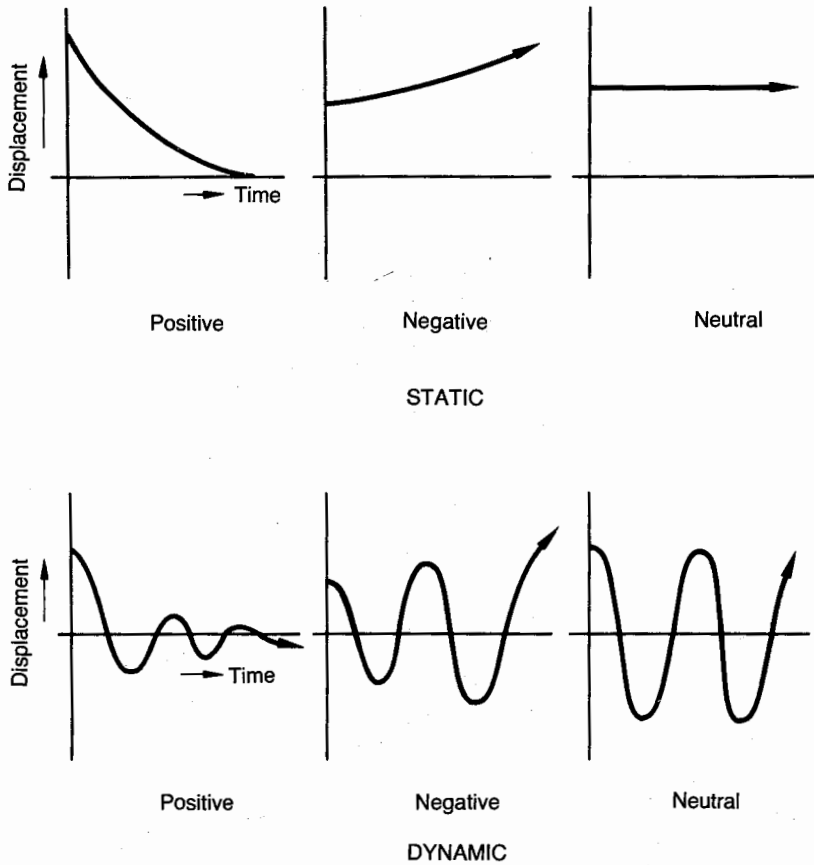
Fig. 1.8 Stability.

uniform motion. In applying this definition to an aircraft, it can be stated therefore, that following a displacement from an original steady flight path, an aircraft has stability if it returns to that path without movements of its flight control surfaces having to be applied.

In practice however, there are two types of stability to consider: static stability and dynamic stability (see fig. 1.8). Static stability refers to the immediate reaction of the aircraft and its tendency to return to equilibrium after displacement, while dynamic stability refers to the subsequent long-term reaction which is of an oscillatory nature about a neutral or equilibrium position. It is usual to classify both types of stability according to the nature of an aircraft's response to displacements from its original steady flight path: thus, stability is *positive* when, subsequent to the displacement, the forces and moments acting on the aircraft return it to its original steady flight path; *neutral* if the forces and moments cause the aircraft to take up a

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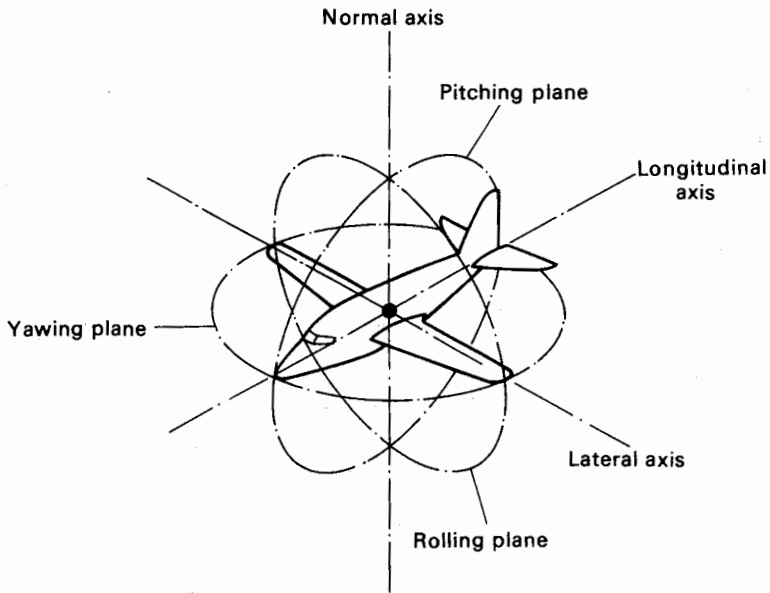
Fig. 1.9 Aircraft axes and displacement planes.

new flight path of constant relationship to the original; and *negative* if the aircraft is caused to diverge from the original steady flight path (an unstable condition). Static stability is a prerequisite for dynamic stability, although the converse is not true; it is possible to have a system which is statically stable, but dynamically unstable.

The displacements of an aircraft which, for example, result from an air disturbance, or by the operation of its flight control system, can take place in any one of three planes; known as the pitching, yawing and rolling planes. This also applies to the aircraft's stabilising motions in response to the displacements. The planes are not constant relative to the earth but, as indicated in fig. 1.9, they are always constant relative to the three body axes passing through the centre of gravity of the aircraft. Both forms of stability relate to the three axes in the following manner: longitudinal stability about the lateral axis, directional or 'weathercock' stability about the normal axis, and lateral stability about the longitudinal axis. In addition to the forces and moments set up by any displacement, forces are also set up as a result of the velocities of motion. These forces are a necessary contribution to the stability of an aircraft, and provide what is termed aerodynamic damping so that motions may be limited or eventually eliminated. The damping in roll, for instance, is the rolling moment due to angular velocity in roll and, since it acts in the opposite sense to the rolling velocity achieved by deflection of the ailerons, the velocity is limited. Damping also applies to pitch and yaw displacements. When this natural form of damping cannot

be obtained, it must be furnished by artificial means, e.g. electrically-controlled yaw dampers (see page 223).

Longitudinal stability

Static

When an aircraft has a tendency to return to a trimmed angle of attack position following a displacement, it is said to have positive static longitudinal stability; it thus refers to motion in the pitching plane, and is influenced largely by the design of the horizontal stabiliser, and on the position of the aircraft's centre of gravity under the appropriate flight and load conditions.

The horizontal stabiliser together with the elevators in the neutral or streamlined position form an aerofoil which produces lift at varying angles of attack, the lift in turn producing either an upward or downward restoring moment to balance wing pitching moments about the aircraft's centre of gravity. The lift and restoring moments produced are governed by such factors as the area and planform of the stabiliser, the distance of its aerodynamic centre from the centre of gravity, i.e. the moment arm, and also by the effects of airflow downwash from the wings. When the elevators are maintained in the neutral or streamlined position, static stability is referred to as stick-fixed stability, as opposed to stick-free stability which refers to the condition in which the elevators are allowed to float in the airflow, i.e. 'hands-off' flight condition.

Assuming that in the stick-fixed position the aircraft is displaced nose up, the angle of attack of the wings and, therefore, the lift produced, will be temporarily increased by an amount d_L, resulting in an increase of the wing pitching moment about the aircraft's centre of gravity. Thus, if the aerodynamic centre is forward of the centre of gravity giving a moment arm of length x as shown in fig. 1.10 *a* the wing pitching moment (M_{wp}) is increased by the amount $d_L x$, the nose-up displacement is thereby worsened and the effect is a destabilising one. Since the nose-up displacement lowers the horizontal stabiliser, then its angle of attack and corresponding lift force will also be increased, but as the position of the aerodynamic centre with respect to the aircraft's centre of gravity provides the longer moment arm, stabiliser lift force produces a stabilising nose-down moment. When the aerodynamic centre of the wings is to the rear of the centre of gravity (fig. 1.10 *b*) the increase in M_{wp} will be stabilising in its effect so that in conjunction with that produced by the horizontal stabiliser a greater restoring moment is provided.

For a given weight in level flight there is one speed and angle of attack at which an aircraft is in equilibrium, i.e. tail moments equal to wing moments. The speed and angle of attack depend upon the difference in rigging incidence between the chord lines of the wing and horizontal

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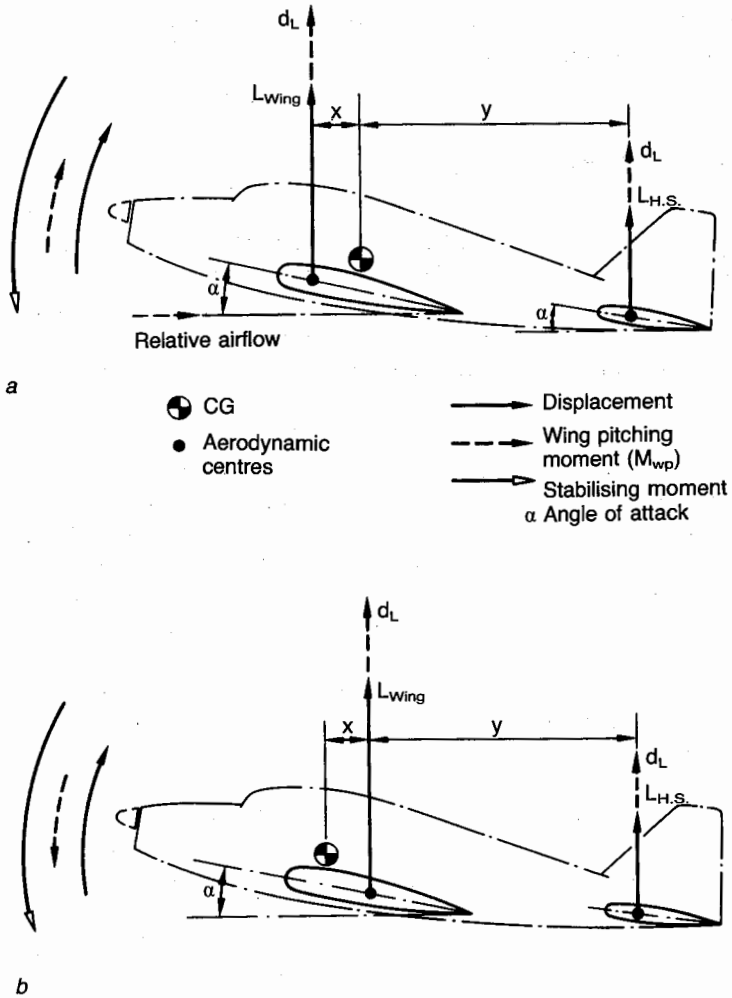
Fig. 1.10 Longitudinal stability.

stabiliser; a difference known as the longitudinal dihedral angle. The angle of attack at which equilibrium is obtained is called the *trim point*.

It is thus apparent from fig. 1.10 that the ratio of the wing moment to stabiliser moment, and therefore the degree of longitudinal stability, is affected by the relative positions of both aerodynamic centres, and of centre of gravity. An indication of this is given in fig. 1.11, which is a graphical representation of the conditions appropriate to the wing of an aircraft. Since stability is evidenced by the development of restoring moments, for the wing to contribute to positive static longitudinal stability the aircraft's centre of gravity must be forward of the aerodynamic centre. In this case,
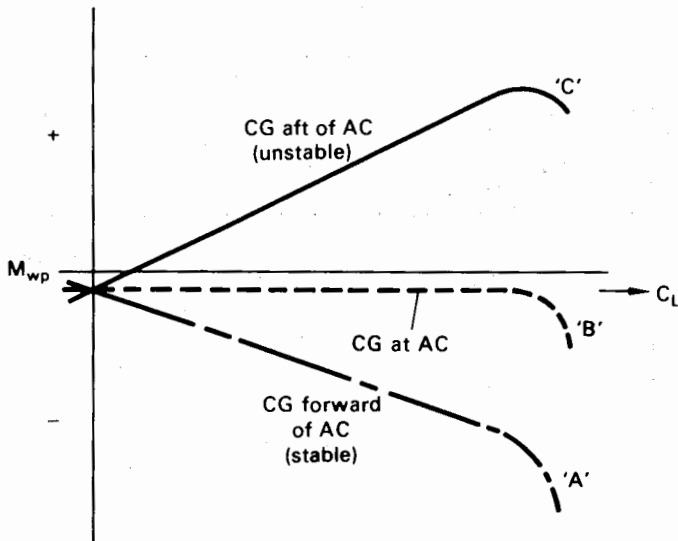
Fig. 1.11 Contribution of wing to longitudinal stability.

the wing contribution is a stable one, and the curve of M_{wp} to lift coefficient (C_L) would have a negative slope (curve 'A'). If the centre of gravity were located at the aerodynamic centre, all changes of lift would take place at the centre of gravity and so the wing contribution would be neutral (curve 'B'). An unstable contribution would be made with the centre of gravity to the rear of the aerodynamic centre, and the M_{wp}/C_L curve would then have a positive slope (curve 'C').

In addition to the wings and horizontal stabiliser, other major components of an aircraft such as the fuselage and engine nacelles can also influence the degree of longitudinal stability since, under varying angles of attack, the conditions of airflow and pressure distribution will produce individual pitching moments which can be either stabilising or destabilising in their influence. In plotting the total pitching moments against C_L, and the contribution of the major components to stability, curves similar to those shown in fig. 1.12 a are obtained (it is assumed in this example that the centre of gravity is at 30% of the mean aerodynamic chord). The contribution of the wing alone is destabilising as indicated by the positive slope of the curve, an effect which is further increased by the fuselage contribution. The large negative slope of the curve of the horizontal stabiliser contribution indicates its highly stabilising effect, which must be sufficient for the complete aircraft to exhibit positive static stability at the anticipated locations of the centre of gravity.

The typical effect of varying locations of the centre of gravity on static stability is indicated in fig. 1.12 b. As the centre of gravity moves rearward,

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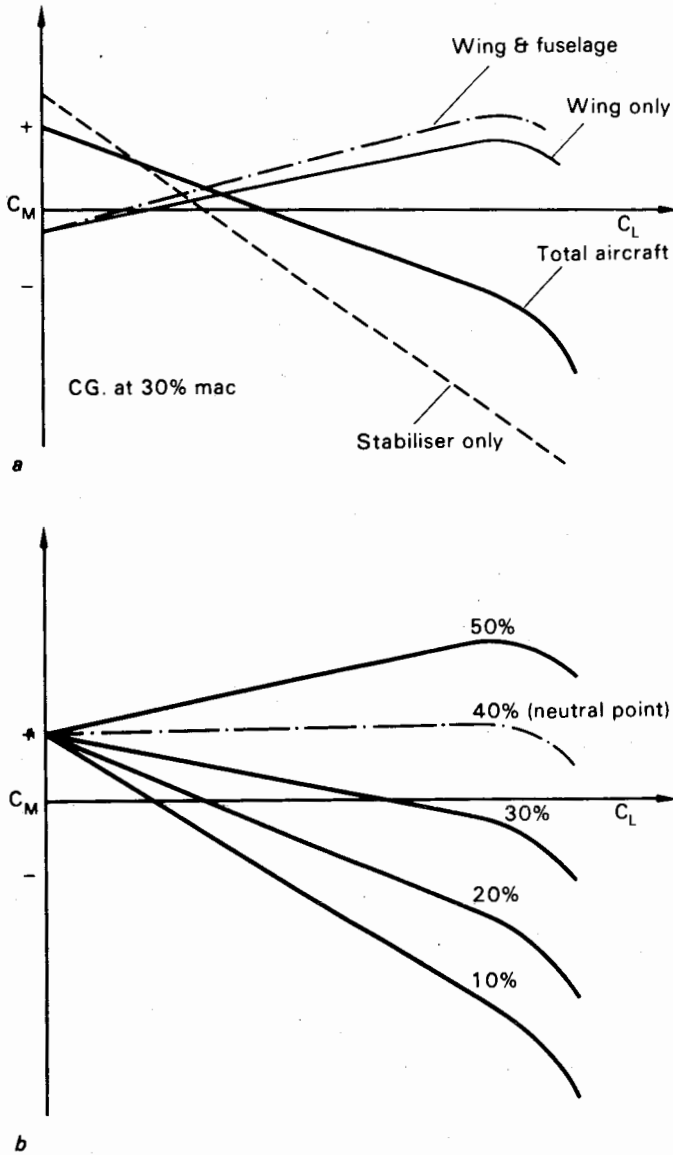
Fig. 1.12 Total pitching moments and effects of varying CG locations.

the static stability decreases, then becomes neutral, and finally results in an unstable condition. The centre of gravity location which produces zero slope and neutral static stability is referred to as the *neutral point*. The distance of the centre of gravity at any time from the neutral point is known as the *static margin* and is an indication of the degree of longitudinal

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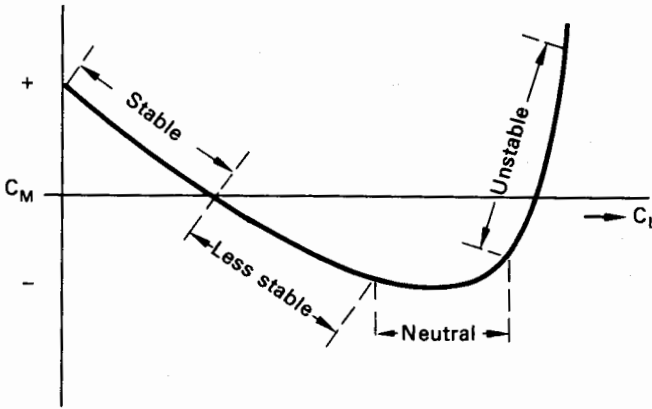
Fig. 1.13 Changes in longitudinal static stability.

stability. Noticeable changes in static stability can occur at varying values of C_L, particularly when power effects contribute largely to stability, or when significant changes in downwash at the horizontal stabiliser occur. Such changes are illustrated in fig. 1.13. At low values of C_L, the slope of the curve indicates good positive stability, but this gradually starts decreasing with increasing C_L. With continued increase in C_L the slope becomes zero indicating that neutral stability exists. Eventually the slope becomes rapidly positive indicating an unstable 'pitch-up' condition.

If the elevators are allowed to float free, they may have a tendency to float or streamline relative to the airflow, when the angle of attack of the horizontal stabiliser is changed. Thus, if the angle of attack is increased and the elevators tend to float up, the change in lift produced by the horizontal stabiliser is less than if the elevator remain fixed; stick-free stability of an aircraft is, therefore, usually less that the stick-fixed stability. Elevators must therefore be properly balanced to reduce floating, and so minimise the difference between stick-fixed and stick-free stability. In the case of fully powered control systems actuated by irreversible mechanisms, the elevators are not free to float and so there is no difference between stick-fixed and stick-free stability.

Dynamic
Longitudinal dynamic stability consists of three basic modes of oscillation, and these are illustrated in fig. 1.14. The first mode (fig. 1.14 *a*) is of very long period and is referred to as a *phugoid* which involves noticeable variations in pitch attitude, altitude and airspeed. The period of oscillation is quite large, and may be counteracted by very small displacements of the elevator control system. The pitching rate is low, and as also only negligible

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a phugoid

Unstable oscillation (porpoising)
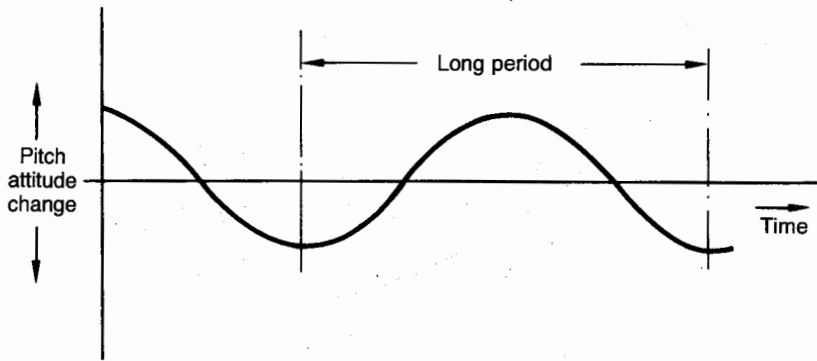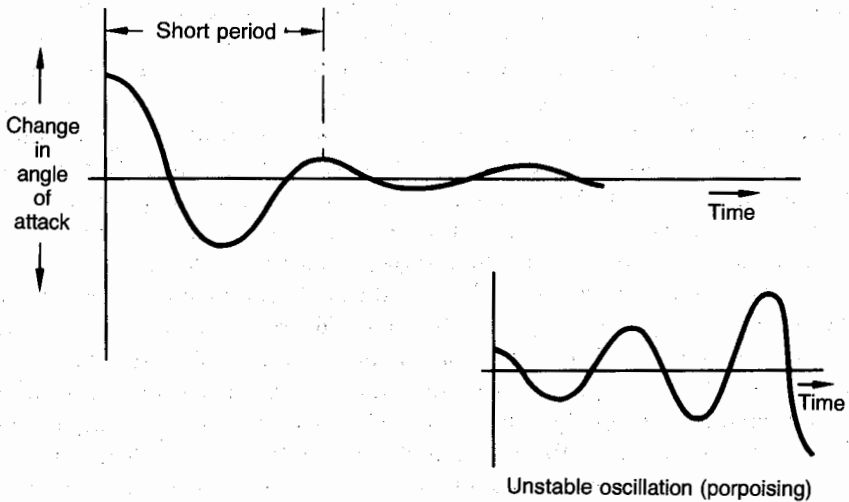
b short period oscillation

Fig. 1.14 Longitudinal dynamic stability.

changes in angle of attack take place, damping of the phugoid is weak and possibly negative.

The second mode (fig. 1.14 *b*) is a relatively short period motion that can be assumed to take place with negligible changes in velocity. During the oscillation the aircraft is restored to equilibrium by the static stability, and

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the amplitude of oscillation decreased by pitch damping. If the aircraft has stick-fixed static stability, the pitch damping contributed by the horizontal stabiliser will usually assume sufficient dynamic stability for the short-period oscillation. The second mode, stick-free, has the possibility of weak damping or unstable oscillations, and for this reason elevators must be statically balanced about their hinge line, and aerodynamic control must be within certain limits. If instability were to exist in the second mode, 'porpoising' of the aircraft would result, and because of the short period of oscillation the amplitude can reach dangerous proportions with the possibility of structural damage resulting from the severe flight loads imposed.

The third mode occurs in the stick-free case, and is usually a very-short-period oscillation. The motion is essentially one whereby the elevators flap about the hinge line and in most cases the oscillation has very heavy damping.

Directional stability

Directional or 'weathercock' stability involves the development of yawing moments which will oppose displacements about the aircraft's vertical axis and so restore it to equilibrium. Unlike longitudinal stability, however, directional stability is not independent in its influence on the behaviour of an aircraft, because as a result of what is termed aerodynamic coupling effect, yaw displacements and moments also producing roll displacements and moments about the longitudinal axis. Thus, directional motions have an influence on lateral motions and vice versa, the motions involved in each case being yawing, rolling, sideslipping or any combination of these.

As far as yawing displacement, forces, and moments only, are concerned conditions are, in fact, analogous to those relating to longitudinal stability but, whereas in the latter case a horizontal stabiliser has the greatest influence, directional stability is influenced by a vertical stabiliser. This may be seen from fig. 1.15 *a*.

Assuming that with the rudder in neutral position the aircraft is yawed to starboard by a disturbance (fig. 1.15 *b*), the vertical stabiliser will be at some angle of attack with respect to the airflow and a corresponding side force (lift) will be produced. Since the position of the aerodynamic centre of the stabiliser with respect to the aircraft's centre of gravity provides the longer moment arm, a stabilising yawing moment to port is created and equilibrium is restored. In addition to the stabiliser moment arm, other factors affecting the size of the stabilising moment are the area of the stabiliser, its aerofoil section, angle of attack, aspect ratio and sweepback. As in the case of longitudinal stability other major components of the aircraft can also influence the degree of directional stability, notably the fuselage and engine nacelles.

When an aircraft is at some angle of yaw, its longitudinal axis is con-

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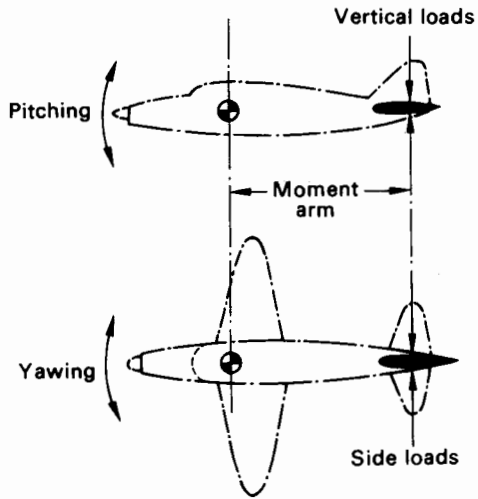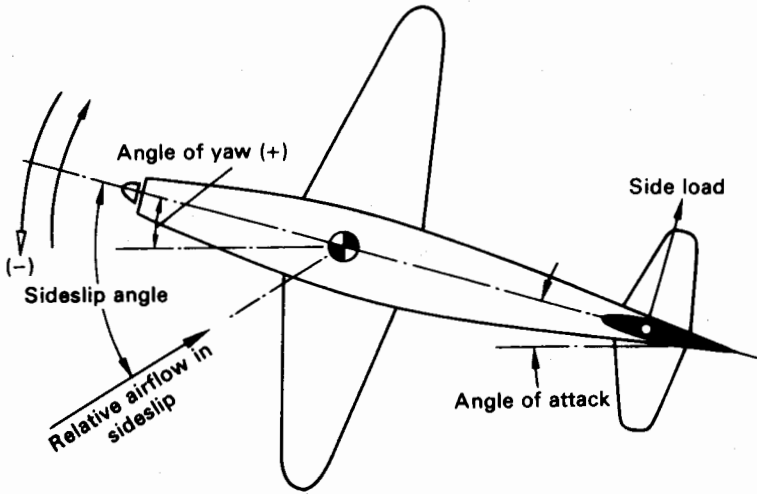
Fig. 1.15 Directional stability.

sidered as being displaced from a reference azimuth, and by convention, a displacement from this azimuth to starboard constitutes a positive angle of yaw, while a displacement to port constitutes a negative angle of yaw. In the yawed condition, and ignoring aerodynamic cross-coupling, the aircraft is maintaining a forward flight path so that, alternatively, the aircraft can be described as being in a condition of sideslip; thus from the example shown in fig. 1.15 and by convention, an aircraft yawed to starboard is sideslipping to port at a negative angle. The angle of sideslip, therefore, is minus the angle of yaw, and since it relates to the displacement of the aircraft's longitudinal axis from the relative airflow rather than a reference azimuth, it becomes a primary reference in directional stability considerations. This is illustrated graphically in fig. 1.16 a.

When an aircraft is subject to a sideslip angle (relative airflow coming from starboard in the case illustrated) static directional stability will be evident if a positive yawing moment coefficient results. Thus, a yawing moment to starboard would be created which tends to 'weathercock' the aircraft into the relative airflow. This is indicated by the positive slope of curve 'A'. If there is zero slope there is of course no tendency to return to equilibrium and so static directional stability is neutral. When the curve has negative slope the yawing moments developed by sideslip tend to diverge, thereby increasing sideslip such that the aircraft would be directionally unstable.

Figure 1.16 b serves to illustrate the fact that the instantaneous slope of the curve depicting yawing moment coefficient/sideslip angle will indicate the static directional stability. At small angles of sideslip, a strong positive slope depicts strong directional stability. Large angles produce zero slope and neutral stability; if the sideslip is very high the slope would indicate instability.

Lateral stability

Static

An aircraft has lateral stability if, following a displacement about the longitudinal axis (called a roll displacement), a rolling moment is produced which will oppose the displacement and return the aircraft to a wings-level condition. In practice however, and because of aerodynamic coupling, rolling moments can also set up yawing or sideslip motions so that the opposing of lateral displacements is not so simple as it seems.

When an aircraft experiences a roll displacement, the effective angle of attack of the down-going wing becomes greater than that of the up-going wing resulting in the appropriate changes in the lift produced (see fig. 1.17). These changes produce a rolling moment which although opposing the initial roll displacement will do no more that provide a damping effect proportional to the rate of displacement. In other words, the aircraft would possess neutral static stability and so would remain in the rolled or banked
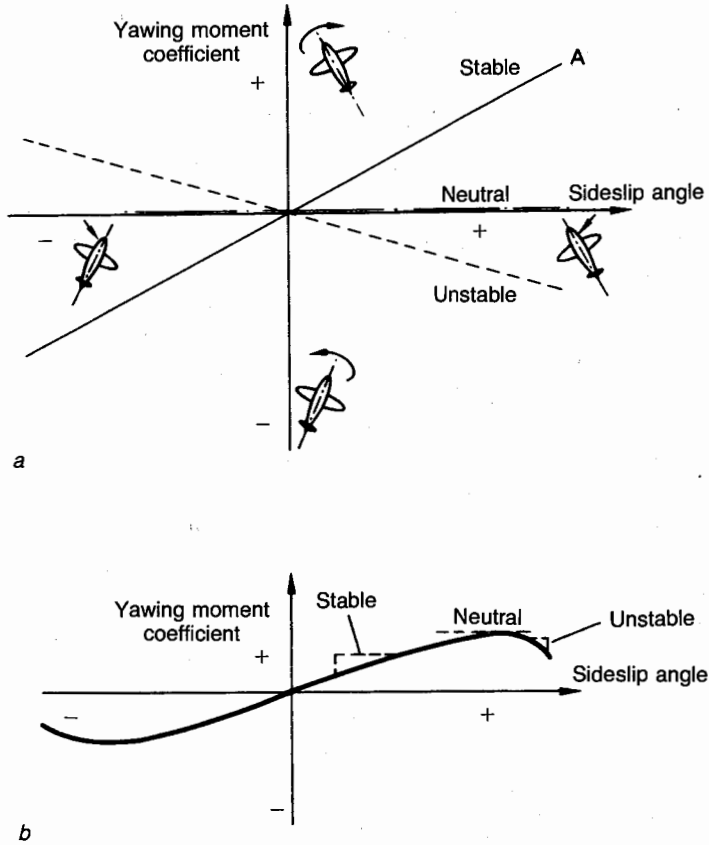
Fig. 1.16 Static directional stability.

position. However, the aircraft also experiences a sideslipping motion which is caused by the inclination of the lift vectors at the roll or bank angle. This motion, in turn, causes the airflow to exert forces on the different parts of the aircraft, and it is the rolling moment induced by sideslipping which establishes static stability reaction and return of the aircraft to a wings-level condition. This may be illustrated by a graph of rolling moment coefficient versus sideslip angle such as that shown in fig. 1.18. When the aircraft is subject to a positive sideslip angle, i.e. it sideslips to starboard, positive static stability will be evident if a negative rolling moment to port is produced; the curve will have a negative slope (curve 'A'). If the slope of the curve is zero, lateral stability is neutral, while a positive slope indicates lateral instability.

The overall value of the lateral static stability will depend on the effects contributed in varying magnitudes, by each different part of the aircraft, these in turn depending on the configuration of the aircraft and on the
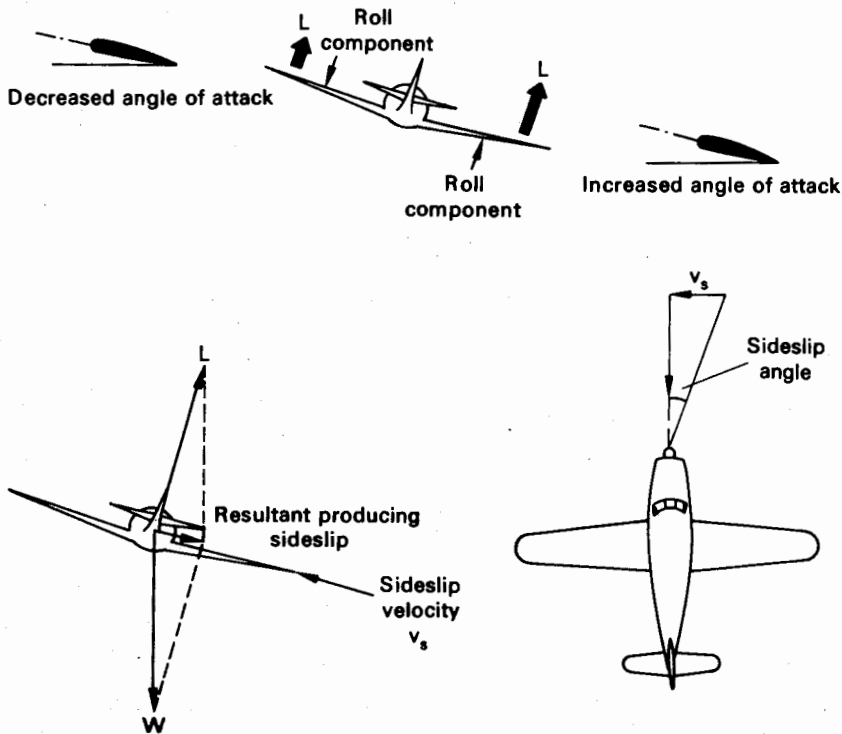
Fig. 1.17 Lateral stability.

condition of flight. The principal contributions to overall lateral static stability are as follows.

1. *The dihedral angle* or upward setting of the wings relative to the horizontal. Dihedral angle is one of the most important contributions to lateral stability which, for this reason, is often referred to as dihedral effect.

2. *The angle at which the wings are swept back* relative to the longitudinal axis. Swept-back wings are a characteristic of many types of high-performance aircraft, producing additional lateral stability which has a greater effect in sideslip at low speeds. In some types of swept-wing aircraft, it may be necessary for stability to be reduced at low speeds, and this is done by setting the wings downwards relative to the horizontal, a setting referred to as *anhedral*. One example of an aircraft utilising anhedral is illustrated in fig. 1.19.

3. *The vertical location of the wings with respect to the fuselage.* In a sideslip, air flows spanwise over the aircraft and causes changes in the effective angle of attack of the wings, such that in the case of a high-wing aircraft, the rolling

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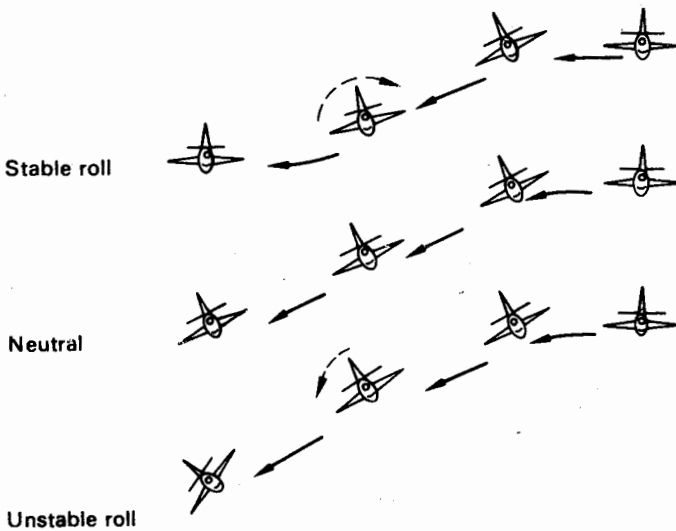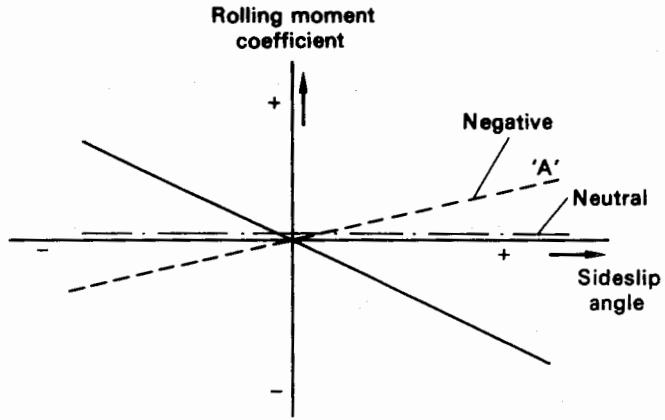
Fig. 1.18 Static lateral stability.

moment produced will be a stabilising one, and in the case of a low-wing aircraft a de-stabilising rolling moment will be produced. There is a zero effect on lateral stability of an aircraft with wings in the mid-position.

4. *The keel surface.* The side load produced in a sideslip acts on an aircraft's fuselage and on the vertical stabiliser (fin) which together form the keel surface. This side load produces a rolling moment that is generally stabilising, but to a smaller degree than the moments produced in other ways.

Fig. 1.19 BAe 146 (reproduced by courtesy of British Aerospace).

5. *The flaps.* When flaps are lowered, they alter the spanwise distribution of pressure and lift, and since they are usually located at the inboard sections of wings, the overall centre of lift is located closer to the fuselage centre line, i.e. the moment arm is reduced. Therefore, any changes of lift resulting from sideslip produce smaller rolling moments thereby reducing the overall lateral stability.

Dynamic
The relative effect of the combined rolling, yawing and sideslip motions produced by aerodynamic coupling, determine the lateral dynamic stability of an aircraft. If the stability characteristics are not satisfactory the complex interaction of the motions will produce three possible forms of dynamic instability: (i) directional divergence, (ii) spiral divergence, and (iii) an oscillatory mode termed Dutch roll.

Directional divergence

This form of instability is a simple divergence in yaw which may occur if the aircraft is statically unstable about the vertical axis: thus, if the aircraft is flying straight and level and it experiences a small displacement in yaw to port, say, the result will be a yawing moment in the same direction thereby increasing the displacement. In addition, a side force will act on the aircraft in the yawed attitude, so that it will curve away from its original flight path. If the aircraft has lateral static stability directional divergence will occur without any significant degree of bank angle, and the aircraft would still fly a curved path with a very large amount of sideslip.

Spiral divergence

This form of instability exists when the directional static stability is very large compared with lateral stability. Assuming once again that a yaw diaplacement to port is experienced, because of the greater directional stability the yaw would be quickly eliminated by a stabilising yaw moment set up by the keel surface. A rolling moment to port would also have been set up by the yaw displacement and if it were strong enough to overcome the restoring moment due to lateral stability, and to the damping-in-yaw effect, the angle of bank would increase and cause the aircraft nose to drop into the direction of yawing. The aircraft then begins a nose spiral which gradually increases to a spiral dive.

Dutch Roll

This is an oscillatory mode of instability which may occur if the aircraft has positive directional static stability but not so much, in relation to the lateral stability, as may lead to spiral divergence. Dutch Roll is commonly found to a varying degree in combinations of high wing loading, sweepback, and high altitude, and where weight is distributed towards wing tips, e.g. engines mounted in pods under the wings. Assuming yet again that the aircraft is yawed to port, it will roll in the same direction. The directional stability will then begin to reduce the yaw to the extent that the aircraft will overswing and start a yaw, and a roll, to starboard. Thus, every period of the continuing oscillations in yaw acts in such a manner as to cause further displacement in roll, the resulting motion being a combination of rolling and yawing oscillations which have the same frequency, but are out of phase with each other.

Controllability

In order that an aircraft may fulfil its intended operational role it must have, in addition to the varying degrees of stability, the ability to respond to requirements for manoeuvring and trimming about its three axes, so that all desired flying attitudes can be achieved and equilibrium be established; in other words, it must have controllability.

Controllability is a different problem from stability in that it requires aerodynamic forces and moments to be produced about the three axes of the aircraft, such forces always opposing the natural stability restoring moments and causing the aircraft to deviate from an equilibrium condition. There is, therefore, a clear relationship between the two which may be illustrated by the analogy of a ball placed on various surfaces (fig. 1.20). Diagram *a* represents the condition of positive static stability and, as we have already learned, any displacement in this condition will always be
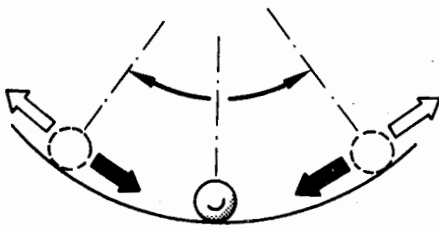
Increased

a positive
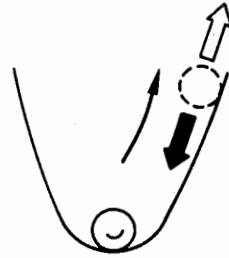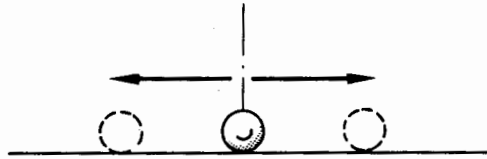
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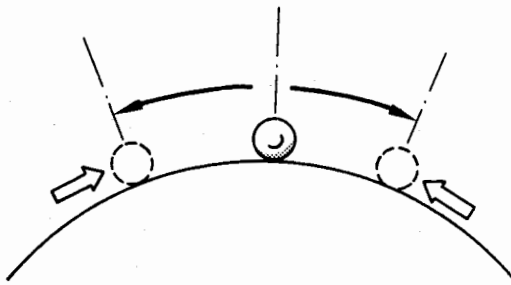
c negative

→ Displacement

➡ Return tendency

⇨ Control balancing

Fig. 1.20 Static stability and controllability.

opposed by a tendency of the ball to return to equilibrium. If, however, it is required to control the ball, and so maintain it in the displaced position, a balancing force must be applied *in the direction* of displacement. When stability is increased a greater balancing force is required to control the ball to the same displaced position. A large degree of stability therefore tends to make for less controllability, so that for aircraft it can be stated that the upper limits of its stability are set by the lower limits of controllability.

In a neutrally stable condition (fig. 1.20 *b*) there is no tendency for the ball to return to equilibrium from a displaced position, and since a new point of equilibrium is always obtained, then no control balancing force is required to maintain the displacement. As the static stability approaches zero, controllability increases to infinity and the only resistance to a displacement would result from damping effects, e.g. the viscosity of air is a damping factor which is proportional to the speed of the displaced body. Thus for an aircraft it can be stated that the lower limits of its stability may be set by the upper limits of controllability.

The effect of negative static stability, i.e. instability, is shown in fig. 1.20 *c*. If the ball is displaced from equilibrium it will tend to continue in the displaced direction and in order to control its displacement a balancing force must, in this case, be applied in a direction *opposite* to the displacement. In applying this reversed form of controllability to an aircraft it would mean that the pilot, in attempting to maintain a state of equilibrium, would also be providing the stability.

It will be apparent from the foregoing that for an aircraft, proper balance must be achieved between stability and controllability, the latter being provided by means of a primary flight control system, and a secondary 'trimming' system.

Primary flight controls

In its basic form, a primary flight control system consists of movable control surfaces connected by cables and rods to cockpit controls which are directly operated by the pilot. The surfaces are aerodynamically balanced to reduce the pilot's physical effort in controlling the aircraft. In high-performance aircraft, the mechanical sections of systems also include powered-actuators (see page 34).

Conventionally there are three sets of control surfaces, and these are situated at the extremities of the wings and stabiliser units (see fig. 1.21) so as to obtain the largest possible controlling moments, consistent with stability, about the three principal axes and centre of gravity. Movement of a control surface causes a change in the aerodynamic profile and therefore a change in the forces produced as shown in fig. 1.22. The pressures acting through the centre of pressure of the control surface, produce a hinge moment which tries to return the surface to its neutral or 'faired' position.

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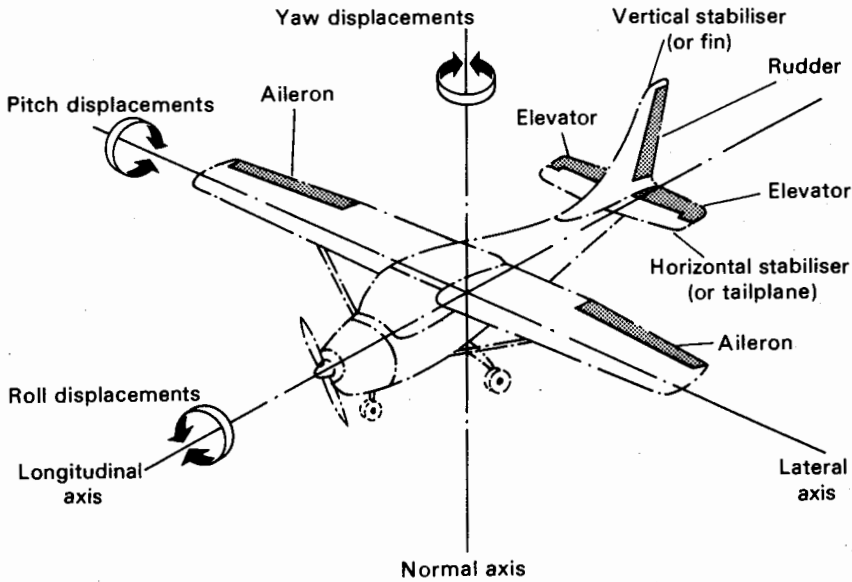
Fig. 1.21 Disposition of flight control surfaces.
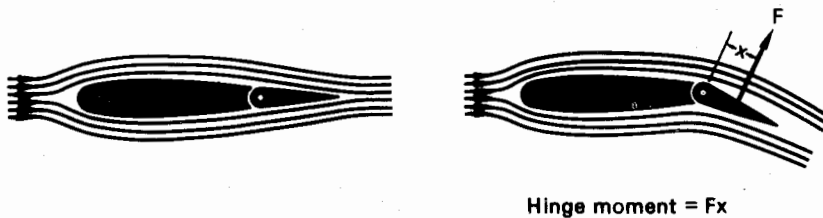
Fig. 1.22 Control surface deflection.

The size of the hinge moment is given by the product of the force on the control surface and its distance x from the hinge point. In order to maintain the surface in its deflected position, the hinge moment is balanced by a control force applied to the control system either manually or automatically.

It is desirable that each set of control surfaces should produce a moment only about the corresponding axis. In practice, however, the cross-coupling effects, which arise from interaction between directional and lateral stability, apply equally to the flight control system; e.g. a yawing moment in addition to a rolling moment is produced when the ailerons are deflected. This will be described in more detail on page 40.

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Ailerons

These provide lateral control, or roll displacements, about the longitudinal axis, the rolling moment produced being opposed by aerodynamic damping in roll. When the two are in balance the aircraft attains a steady rate of roll; ailerons are therefore essentially rate control devices. As will be noted from fig. 1.23 ailerons are, in most cases, operated by a control 'wheel' pivoted on a control column. They are always connected so that in response to instinctive movements of the control wheel by the pilot, they move in opposite directions thereby assisting each other in producing a roll displacement. Thus, when the control wheel is turned to the left, the left-wing aileron is raised and the wing lifting force is decreased, while the aileron of the right wing is lowered causing the lifting force of that wing to increase and thereby initiate a roll displacement to the left. A similar but opposite effect occurs when the control wheel is turned to the right.

Elevators

Elevators provide longitudinal or pitch control about the lateral axis, and they also assist the horizontal stabiliser to maintain longitudinal stability. Elevators are usually in two separate halves and are normally mounted on a common hinge line at the rear of the horizontal stabiliser. They are connected to the control column, which can be moved backwards and forwards.

When the column is moved backwards, the elevators are raised, thereby decreasing the lift of the horizontal stabiliser so that the aircraft is displaced by a pitching moment about the lateral axis into a nose-up, or climbing attitude. Forward movement of the control column lowers the elevators to increase the lift of the horizontal stabiliser and so the pitching moment causes the aircraft to assume a nose-down or descending attitude. Pitch displacements are opposed by aerodynamic damping in pitch and by the longitudinal stability (see page 13) and as the response to elevator deflections is a steady change of attitude, elevators are essentially displacement control devices. It will also be noted from fig. 1.23 that control column and control wheel movements are independent of each other so that lateral and longitudinal displacements can be obtained either separately or in combination.

Rudder

This surface provides yawing moments or directional control about the normal axis of the aircraft, such control being opposed by damping in yaw, and by the directional stability. The rudder is operated in response to instinctive movements by the pilot of a foot-operated rudder bar or, more usually, of a pair of rudder pedals. Thus, if the left pedal is pushed forward

Control wheel

Rudder pedals
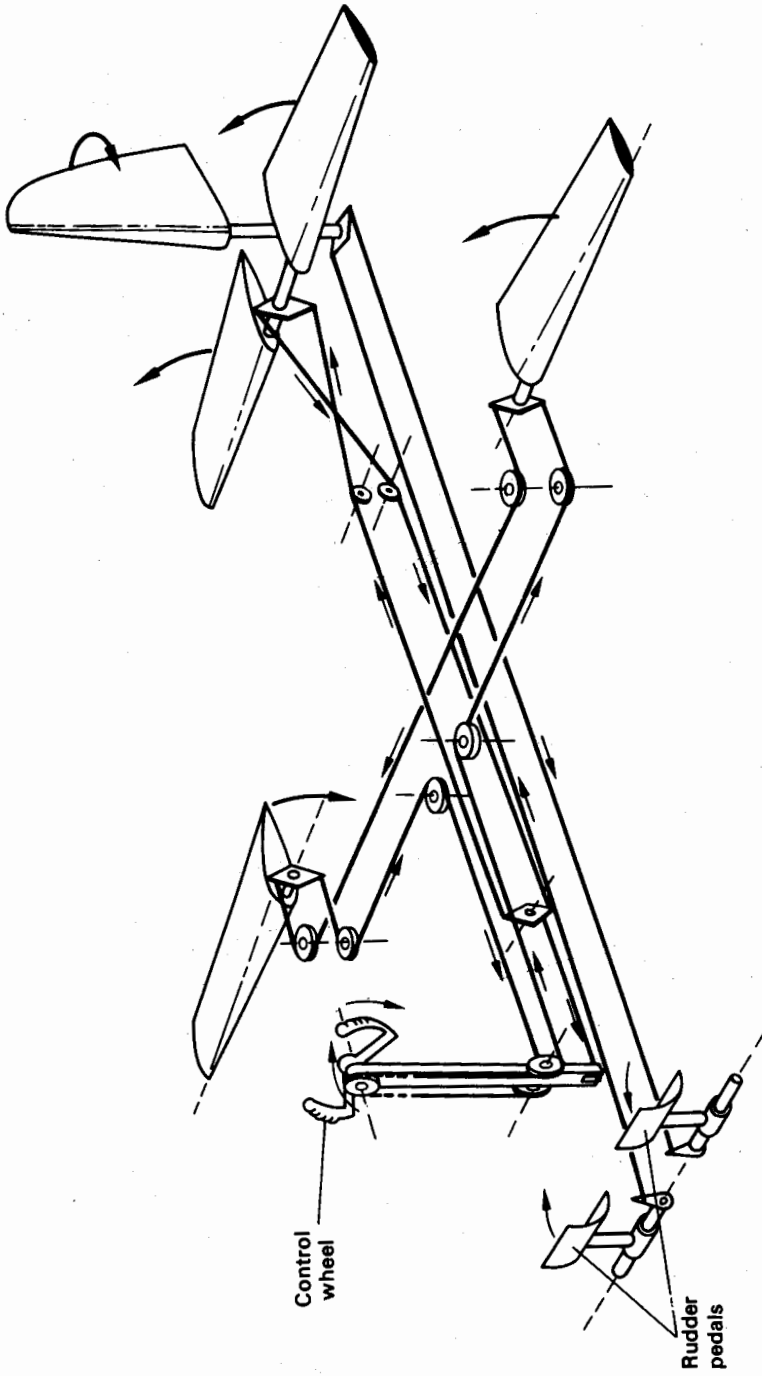
Fig. 1.23 Primary flight control system.

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the rudder is turned to the left and the force produced on the vertical stabiliser sets up a yawing moment which displaces the aircraft's nose to the left. A corresponding displacement to the right is set up when the right rudder pedal is pushed forward. The response to rudder deflections is a steady state of change of angle of attack on the keel surfaces and so, like the elevators, the rudder is a displacement control device.

Effectiveness of controls

The effectiveness of a control system, i.e. the moment produced for a given control surface deflection, depends on the magnitude of the force produced by the control surface and also on the moment arm, i.e. distance from the centre of gravity. The aerodynamic forces acting on an aircraft depend, among other important factors, on the airstream velocity, so the effectiveness of flight controls varies accordingly. At low speeds, large control surface movements are needed, while smaller ones are necessary at high speeds. Movement of a control surface alone may also alter control effectiveness, since the loads set up tend to twist and bend the aircraft's structure which has a certain amount of inherent flexibility.

In many types of large aircraft it is usual for control surfaces to be arranged in pairs, i.e. an inboard and an outboard aileron on each wing, an inboard and an outboard elevator, and an upper and lower rudder (see fig. 1.24 which illustrates the arrangements in the McDonnell Douglas DC–10). The reasons for such arrangements are to ensure control ef-
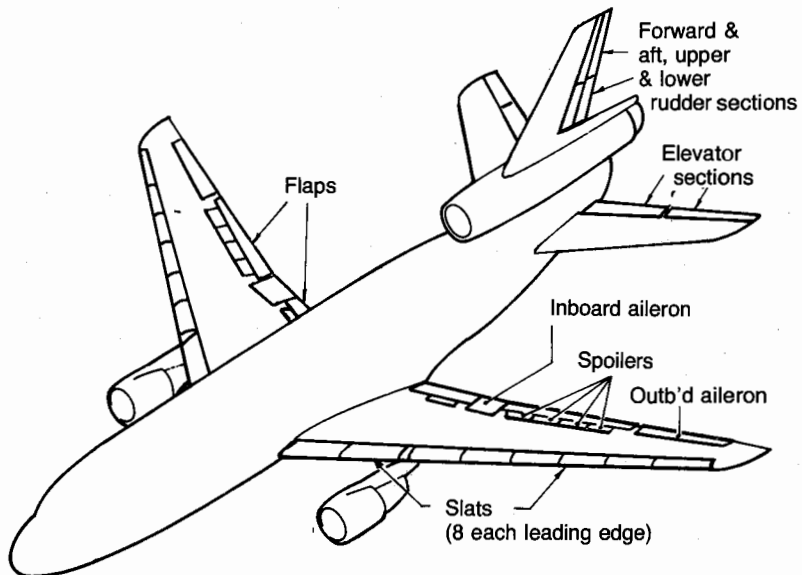
Fig. 1.24 Paired flight control surfaces.

fectiveness at both low and high speeds, particularly where lateral control is concerned, and also to ensure control in the event of any failures in the system. At low speeds, only the outboard ailerons provide lateral control, while at high speeds they are locked and lateral control is taken over by the inboard ailerons so that the deflecting forces act closer to the longitudinal axis thereby reducing wing twisting. The duplication of elevators and rudder is done primarily as a safety precaution. Each elevator and rudder is operated by an independent control system so that should a failure of a system occur one control surface of a pair can still be effective.

It will also be noted from fig. 1.24 that the rudder sections are in forward and aft pairs. The forward sections are actuated by the rudder pedals and associated hydraulic system, while the aft sections are hinged to the forward sections and are mechanically connected by pushrods to the vertical stabiliser. This arrangement provides for proportional displacements of the forward and aft sections and thereby an overall increase in rudder efficiency. The travel of each forward section is approximately 23° left or right of the neutral position, while the travel of each aft section is approximately 22° left or right relative to the forward sections.

Control gearing

Control gearing refers to the relationship between movements of the pilot's controls and the displacements of the corresponding control surfaces, and in the design of the flight control systems of any one type of aircraft, this relationship is a variable one so as to produce the required handling characteristics in all conditions of flight. The variation, or gear change ratio to be more precise, is effected through the mechanical linkages in the control circuits in such a way what when the pilot's controls are moved a certain extent around their neutral position, a relatively small displacement of the control surfaces will be produced; the same extent of movement near the extremities of control movement range on the other hand, will produce a much larger displacement of the control surfaces.

The control gearing relationship can also be varied at different airspeeds by means of a dynamic, or 'q', pressure sensing gear change unit. In this case, the pressure/speed signals are fed to 'ratio changers' in hydraulic power actuators which operate so as to reduce control surface displacements as airspeed increases. This method is, for example, applied to the rudder control system of the Boeing 747.

Combined controls

In certain types of aircraft, the primary flight control system is arranged so that one type of control surface may combine its function with that of another; e.g. on a delta wing aircraft such as Concorde, a control surface at each trailing edge can perform the function of both ailerons and elevators;

such a control surface is called an *elevon*. When the control column is moved either backwards or forwards both surfaces move together in the manner of elevators, but when the control wheel is turned, one elevon is raised and the other lowered as in the case of conventional ailerons. The interconnection between the two control systems is such that the surfaces can be deflected simultaneously to produce combined pitching and rolling moments.

Another example of combined controls is the one applied to some light aircraft having a 'V' or 'butterfly' tail. In this case, the control surfaces operate as either a rudder or as elevators, and for obvious reasons, they are known as *ruddervators*. They are connected to the control column and are moved up or down to produce pitching moments as in the case of conventional elevators. They are also connected to the rudder pedals, so that they move equal amounts in opposite directions to produce the required yawing moment. The control column and rudder pedal systems are connected to the surfaces through a differential linkage or gearing arrangement, so that combined pitching and yawing moments can be obtained.

In some aircraft, elevators are dispensed with and they are substituted with a movable horizontal stabiliser. Thus when the control column is moved the angle of attack of the stabiliser is varied such that a negative angle produces a nose-up attitude, and a positive angle produces a nose-down attitude. Such a stabiliser is known as a *stabilator*.

Powered flight controls

Powered flight controls are employed in high-performance aircraft, and are generally of two main types (a) power-assisted and (b) power-operated. The choice of either system for a particular type of aircraft is governed by the forces required to overcome the aerodynamic loads acting on the flight control surfaces. In basic form, however, both systems are similar in that a hydraulically-operated servo-control unit, consisting of a control valve and an actuating jack, is connected between the pilot's controls and relevant control surfaces. The major difference, apart from constructional features, is in the method of connecting actuating jacks to control surfaces and this may be seen from fig. 1.25.

In a power-assisted system, the pilot's control is connected to the control surface, e.g. control column to elevators, via a control lever. When the pilot moves the control column to initiate a climb say, the control lever pivots about point 'X', and accordingly commences moving the elevators up. At the same time, the control valve pistons are displaced and this allows oil from the hydraulic system to flow to the left-hand side of the actuating jack piston, the rod of which is secured to the aircraft's structure. The reaction of the pressure exerted on the piston causes the whole servo-unit, and control lever, to move to the left, and because of the greater control effort produced the pilot is assisted in making further upward movement of the elevators.
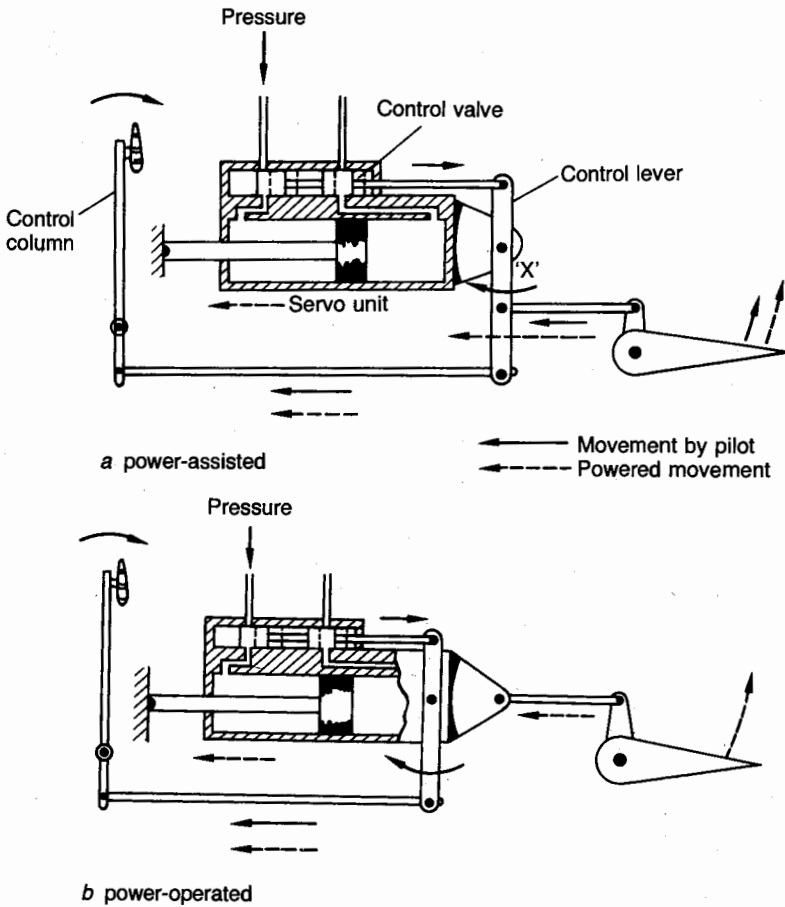
Fig. 1.25 Powered flight controls.

In a power-operated system the pilot's control is connected to the control lever only, while the servo-unit is directly connected to the flight control surface. Thus, in the example considered, the effort required by the pilot to move the control column is simply that needed to move the control lever and control valve piston. It does not vary with the effort required to move the control surface which, as will be noted from the diagram, is supplied solely by servo-unit hydraulic power. Since no forces are transmitted back to the pilot he has no 'feel' of the aerodynamic loads acting on the control surfaces. It is necessary therefore, to incorporate an 'artificial feel' device at a point between the pilot's controls, and their connection to the servo-unit control lever.

A commonly used system for providing artificial feel, particularly in elevator and horizontal stabiliser control systems, is the one known as 'q'-

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feel. In this system, the feel force varies with the dynamic pressure of the air (i.e. $\frac{1}{2}\rho V^2$ or 'q') the pressure being sensed by a pilot-static capsule or bellows type sensing element connected in the hydraulic powered controls, such that it monitors hydraulic pressure, and produces control forces dependent on the amount of control movement and forward speed of the aircraft.

'Fly-by-wire' system

Another system which may be considered under the heading of powered flight controls, is the one referred to as a 'fly-by-wire' (FBW) control system. Although not new in concept, complete re-development of the system was seen to be necessary in recent years, as a means of controlling some highly sophisticated types of aircraft coming into service. The problem associated with such aircraft has been one of designing conventional forms of mechanical linkage to suit the complex flight control systems adopted. Thus, an FBW system, as the name very aptly suggests, is one in which wires carrying electrical signals from the pilot's controls, replace mechanical linkages entirely. In operation, movements of the control column and rudder pedals, and the forces exerted by the pilot, are measured by electrical transducers, and the signals produced are then amplified and relayed to operate the hydraulic actuator units which are directly connected to the flight control surfaces.

In some current types of aircraft, the application of the FBW principle is limited to the control of only certain of the flight control surfaces; for example, wing spoiler panels in the case of the Boeing 767. This particular application is schematically illustrated in fig. 1.26, and serves as a useful basis for the general understanding of FBW operating principles. Other FBW systems are covered in chapter 11.

For lateral control, the deployment of the panels is initiated by movement of the pilot's control wheels to the left or right as appropriate. This movement operates position transducers, in the form of rotary variable differential transformers (RVDTs) via mechanical gear drive from the control wheels. The RVDTs produce command voltage signals proportional to control wheel position and these signals are fed into a spoiler control module for processing and channel selection.

The spoiler control module output signals are then supplied to a solenoid valve forming an integral part of a hydraulic power control actuator. The valve directs hydraulic fluid under pressure to one or other side of the actuator piston which then raises or lowers the spoiler panel connected to the piston rod. The actuator is mounted so that it pivots to allow for the required angular movement of the spoiler panel. As the actuator piston rod moves, it also actuates a position transducer of the linear variable differential transformer (LVDT) type, and this produces a voltage feedback signal proportional to spoiler panel position. When the feedback signal

Spoi

Hydraulic
Pressure

Power Control
Actuator

LVDT

Processed Command Signal

Feedback Signal

Signal
Control
Module

Command Signal

Command Signal

Position Transducer
(LVDT)

Position Transducer
(RVDT)

Hydro-Mechanical

Electrical

osition Transducer
(RVDT)
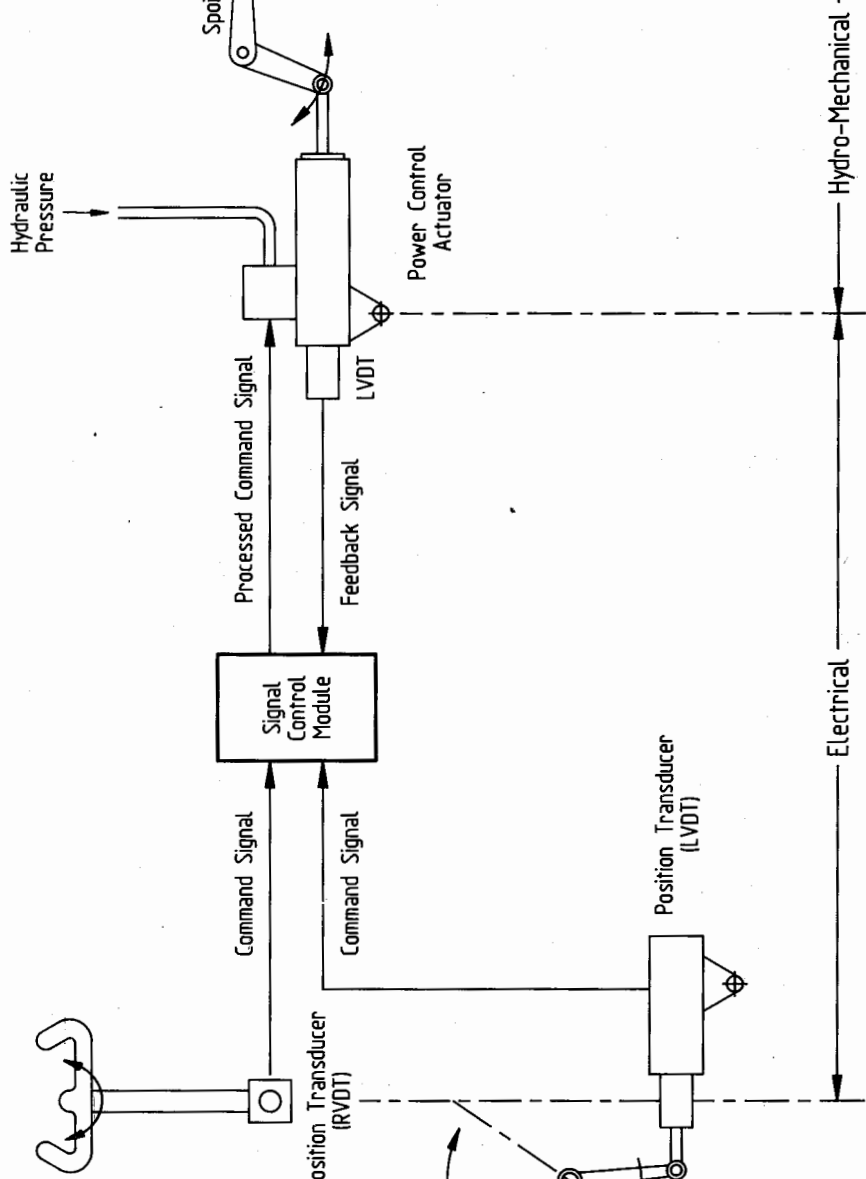
-by-wire system.

equals the command signal, a 'null' condition is reached and spoiler panel movement stops.

Deployment of spoiler panels for the purpose of acting as speedbrakes is initiated by movement of a speedbrake lever. The lever operates an LVDT type of transducer which produces a command voltage signal for processing by the signal control module. The output signal operates the actuator in the same way as for lateral control except that the spoiler panels are deployed to their fullest extent. 'Nulling' of the command signal is also produced in the same way.

Lateral control and speedbrake signals are mixed in the signal control module to provide the proper ratio of simultaneous operation.

As a further advance in the 'fly-by-wire' (FBW) concept, systems utilising fibre-optic cables for conveying flight command signals have also now been developed. The principal advantage of this method is its immunity to electromagnetic interference, and the consequent elimination of heavy shielding required to protect the more conventional 'signal wires'.

In a fibre-optic cable system, signals are transmitted in the form of light through a number of glass fibres, and where applications to aircraft are concerned, this has given rise to the term 'fly-by-light'. In relation to currently developed systems, however, the term is a misnomer because in these systems, light transmission applies only to command signalling and not to signal processing which is performed electronically within control system computers.

Side stick controller

An attraction of an FBW system is the ability to replace the conventional control wheel/column with a small side stick or side arm controller. Apart from size and location and lack of movement, it acts in the same way as a normal cockpit control.

Manoeuvring and forces affecting an aircraft

The displacements resulting from the various movements of the flight control surfaces are those intentionally set up by the pilot in order to manoeuvre his aircraft into required flight attitudes. Such attitudes are: straight and level, climbing, descending, rolling, turning and a combination of these, e.g. a climbing turn. There are four principal forces affecting an aircraft in flight and the directions in which they act are shown in fig. 1.27. Lift, as we learned at the beginning of this chapter, acts at right-angles to the direction of the airflow from the centre of pressure, the position of which can vary with changing angle of attack. Weight acts vertically downward through the centre of gravity which can also vary in position with changing load conditions. Thrust is the forward propulsive force produced by either a turbine engine or a propeller to overcome the opposing total drag force.
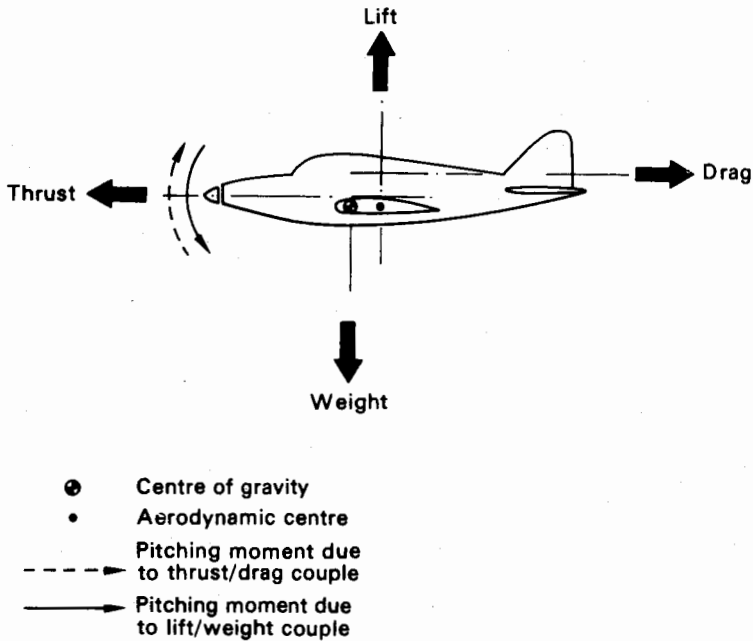
Fig. 1.27 Balance of forces.

Straight and level flight

To be in equilibrium in a straight and level flight attitude at constant speed, lift must equal weight, and thrust must equal drag, and it is arranged that these forces act from points which are not coincident, thereby producing couples which give rise to pitching moments. For example, because the relative positions of the centre of pressure and centre of gravity can vary during flight, the lift and weight forces produce couples which cause either a nose-up or a nose-down pitching moment. Similarly, pitching moments result from the displacement couples of thrust and drag. Ideally, the moments arising from these two couples should balance each other, and by design it is usual for a nose-down moment due to the lift/weight couple, to be balanced by a nose-up moment due to the thrust/drag couple as indicated in fig. 1.27. If the thrust is then decreased either by a deliberate reduction in engine power or by an engine failure, the lift/weight couple will overcome the reduced thrust/drag couple and cause a nose-down moment thereby putting the aircraft into a nose-down attitude.

Climbing and descending

These are pitch attitude manoeuvres which are set up by upward and downward movements respectively of the elevators. In the case of stabila-

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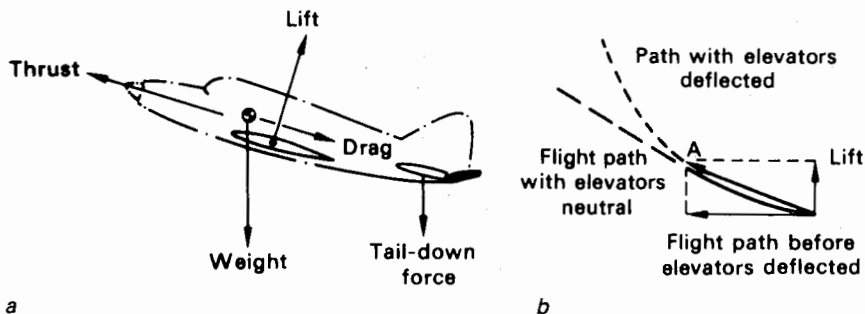
Fig. 1.28 Climbing.

tors referred to earlier, the horizontal stabiliser is deflected to produce a negative angle of attack for the setting up of a climbing attitude, and a more positive angle of attack for a descending attitude. As an example of how the forces act we may consider the case of an upward deflection of elevators to produce a climbing attitude from straight and level flight as shown in fig. 1.28. When the elevators are deflected a pitching moment is produced to rotate the aircraft about the centre of gravity causing the lift vector to be inclined, and thereby constitute an accelerating force at right angles to the direction of flight causing the aircraft to initially follow a curvilinear path. Fig. 1.28 *b* is a vectorial representation of the conditions obtaining. The lift force in a climb is normally less than the weight, a component of which acts in the direction of drag; therefore, more power, i.e. greater thrust is required to lift the aircraft at a vertical speed, otherwise known as the rate of climb. At the required rate and angle of climb the elevators are returned to their neutral position, as at point A, and the aircraft will fly along a path tangential to the original curve.

Rolling and turning

Rolling of an aircraft takes place about the longitudinal axis, and is initiated by deflecting the ailerons in the required direction. As the effective angle of attack of each wing is thereby changed, then the down-going wing produces a greater lift than the up-going wing (see page 21) and so a rolling moment is established. The total lift vector (see fig. 1.29) rotates through the same angle as the aircraft and gives rise to two components, one vertical and equal to $L \cos \theta$ and the other horizontal and equal to $L \sin \theta$. The latter component acts through the centre of gravity, and it establishes an inwards or centripetal force, thereby causing the aircraft to accelerate into a curvilinear flight path; thus displacing an aircraft into a rolled attitude also results in turning.

For any given airspeed and turn radius, however, turning of this nature
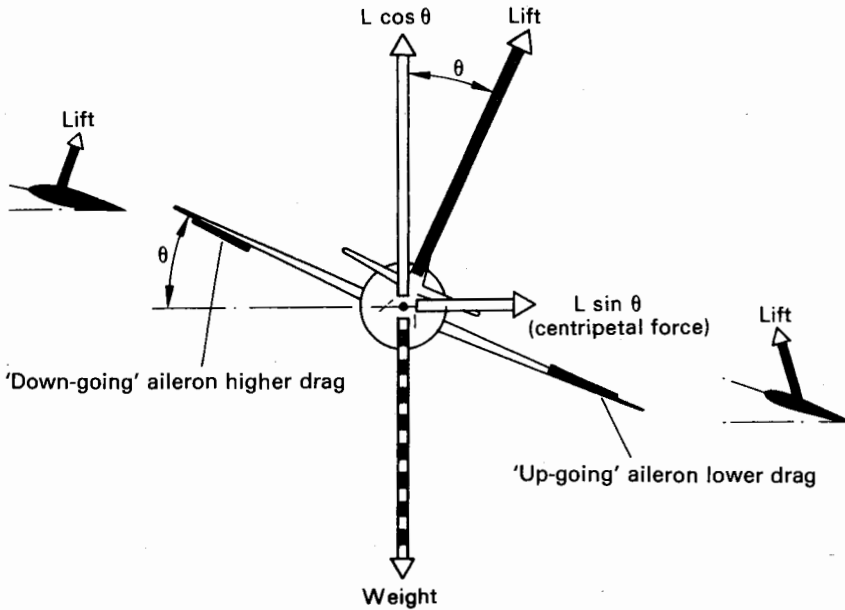
Fig. 1.29 Rolling and turning.

would only occur at one correct angle of bank. At any other angle, and because of a cross-coupling response (see page 29), a yawing moment is produced which opposes the rolling moment. The yawing moment arises in this case, because the lift vector of the down-going wing is inclined forwards, while that on the up-going wing is inclined rearward. Furthermore, the up-going wing, being further from the centre of the turn, has the greater angular velocity, and so it sustains more drag. The ailerons themselves, when in their deflected positions, can also create what is termed adverse aileron yaw (see fig. 1.30). When an aileron is lowered, it moves into a region of high pressure and so the aileron drag is increased; it is greater than that of the opposite aileron which, on being raised, is moved into a region of low pressure. Thus, with drag greater on one aileron than on the other, the effect is to yaw an aircraft in the direction opposite to that intended. This effect is reduced as much as possible by operating the ailerons differentially, i.e. when control is applied the 'up-going' aileron moves through a greater angle than the 'down-going' one.

A yawing moment also results in sideslip (see page 21) and this must be prevented if the aircraft is to be manoeuvred into a steady turn at any speed and roll angle; in other words, the turn must be a co-ordinated one. If an aircraft has a well-designed aileron system, a co-ordinated turn can be achieved by aileron deflection alone. As is more often the case, however, the ailerons must be assisted by deflection of the rudder in the direction of its

41

'Down' aileron (higher drag)   'Up' aileron (lower drag)

- - - → Aileron rolling moment
———▷ Rolling moment due to yaw
———→ Intended direction of turn
- - -▷ Adverse yaw

Rudder correction
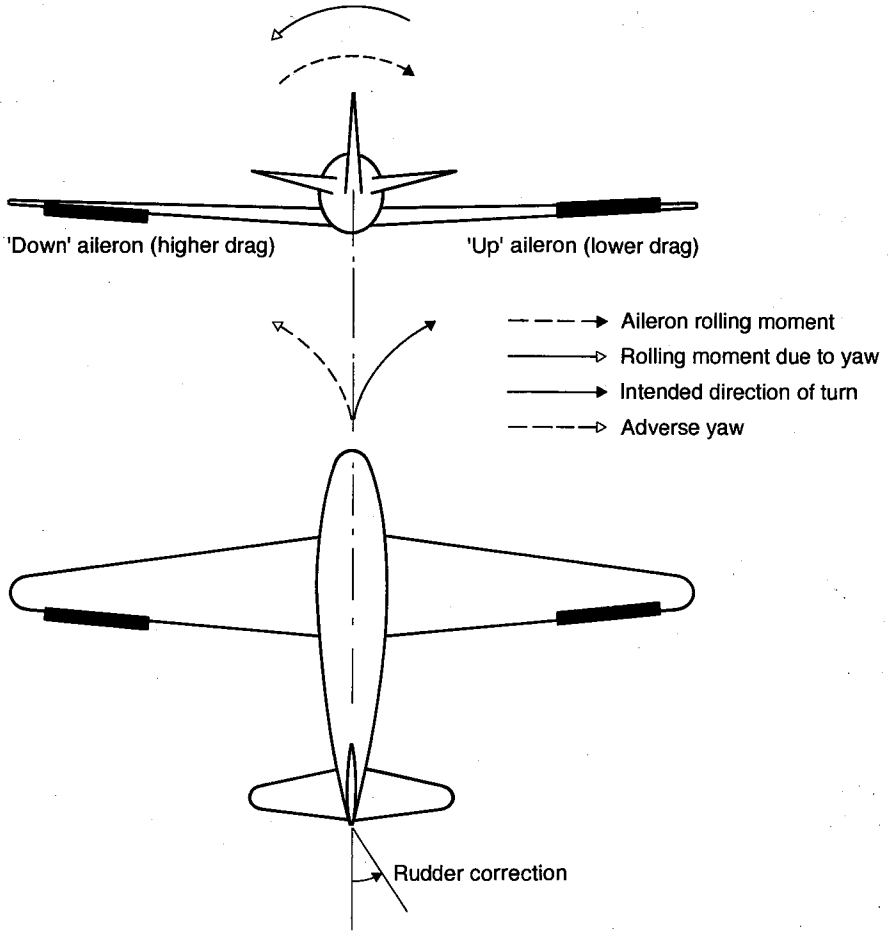
Fig. 1.30 Adverse aileron yaw.

turn, thereby creating a side force on the vertical stabiliser to overcome adverse aileron yaw.

Trimming

The balancing of the aerodynamic forces and moments and the establish-ment of desired flight attitudes are continuous processes and, as we have already observed, are governed by the degree of inherent stability of an aircraft and by the manoeuvring capability afforded by its primary flight control system. In flight, however, control must also be exercised over changes in weight and centre of gravity locations which occur as a result of

the consumption of fuel, disposition of passengers and cargo, flight under asymmetric power conditions, etc.; in addition, the attitude changes resulting from the lowering of flaps must also be controlled. Although the required control could be maintained by repositioning relevant primary flight control surfaces, varying degrees of physical effort on the part of a pilot would be needed to keep the control surfaces in specifically displaced positions. It is usual, therefore, to provide a secondary control system which can be separately adjusted so that it will displace the primary control surfaces, thereby reducing the effects of aerodynamic loads on the primary control system, and so relieving the pilot of undue physical effort. The operation of such a system is referred to as 'trimming', and some typical methods by which it is operated are described briefly in the following paragraphs.

Trim tabs

In this method, which may be considered as the basic form of trimming, an auxiliary surface known as a tab is hinged at the trailing edge of a primary control surface and is connected via a cable, linkage and gearing system to a trim wheel in the cockpit. The wheel is arranged so that it can be rotated in the same sense as the required trim change.

As an example of tab operation, let us consider the case whereby balancing of forces and moments to maintain straight and level flight requires that the aircraft adopt a nose-up attitude. In order to obtain this the elevators must be displaced in an upward position, but for this to be done by movement of the control column the pilot would have to maintain a constant pull on the control column. However, by means of an elevator trim tab, the pilot can set up the required elevator displacement simply by rotating the trim wheel in the appropriate direction, in this case rearwards. As will be noted from fig. 1.31 the tab will move downwards, so that it is
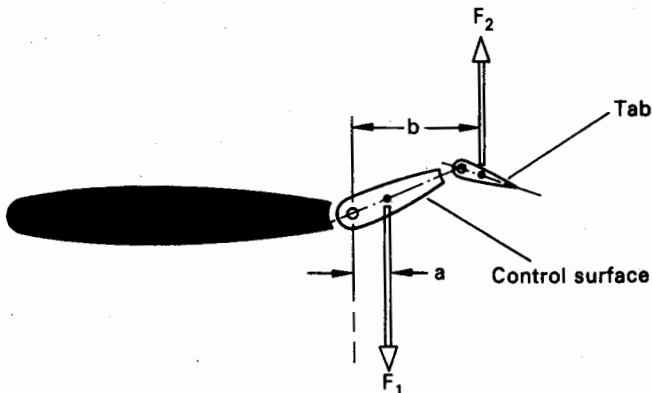
Fig. 1.31 Principle of a trim tab.

the airloads acting on the tab which will deflect the elevators upwards, and also move the control column to some rearward position. In terms of moments, the one produced by the elevators is $F_1 \times a$, and this is balanced by that produced by the tab, i.e. $F_2 \times b$.

Tabs may also be designed for purposes other than trimming, depending on the aerodynamic and flight control system characteristics appropriate to a particular type of aircraft. For example, they may be designed for (i) balancing the flight controls by varying the hinge moments produced by the controls; these are known as balance or geared tabs, (ii) reducing control loads at high speeds (spring tabs), and (iii) reducing control loads necessary to manoeuvre an aircraft (servo tabs).

Balance panels

In some types of aircraft, aerodynamic loads acting on ailerons and elevators are further reduced by means of balance panels that operate in conjunction with balance tabs. A typical arrangement as applied to an aileron is illustrated in fig. 1.32. The panel is interconnected to the leading edge of the aileron and the wing rear spar by hinged fittings, so that at the wing trailing edge it divides the area between the upper and lower surfaces into two vented compartments.

With the aileron and tab in their neutral positions, the pressure of the air vented into each compartment is in balance. When the aileron is displaced, e.g. upward, a higher pressure is developed in the upper compartment so that the resulting force acting on the balance panel, together with that acting on the downward displaced tab, assists in the application of aileron control.

Variable-incidence stabiliser

On many high-performance aircraft, trimming is effected by varying the angle of incidence of the horizontal stabiliser, the latter also operating in conjunction with elevators. This provides a much wider range of trim capability, and more precise manoeuvring in the pitching plane is possible because the full range of elevator movement is always available. Furthermore, this all-moving tail configuration overcomes the loss of effectiveness from which the use of elevators alone can suffer, especially at high speeds.

It is beyond the scope of this book to go into any detail relating to all the various methods of stabiliser operation, but the fundamental principles involved may be understood from fig. 1.33. The incidence of the stabiliser is varied by an actuator assembly which can be operated by any one of three methods: (i) by energising an electric actuator motor through switches on the control columns; (ii) by a trim servo-actuator (see page 213) which responds to signals from the automatic flight control system; if the latter is not in control, the servomotor can also respond to signals from a Mach trim system (see page 231), and (iii) by manually-operated trim wheels and
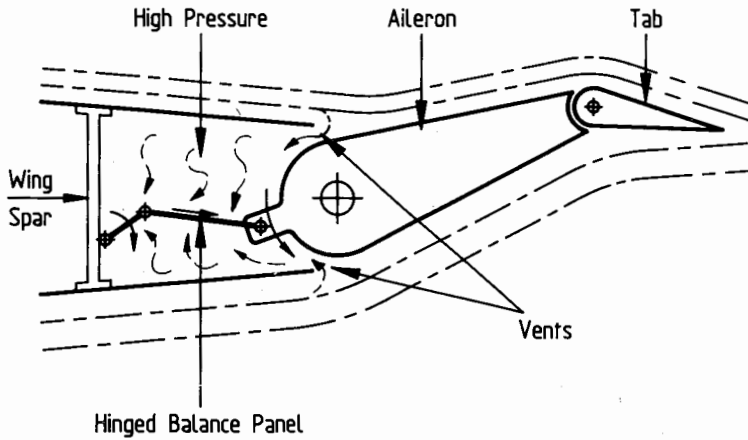
Fig. 1.32 Balance panel.

control cables; this method is for emergency operation in the event of malfunction of the other two methods.

The actuator assembly consists of a jackscrew on which is threaded a ball nut connected to the leading edge section of the horizontal stabiliser. The lower portion of the assembly is connected to the fuselage structure by a gimbal which allows fore and aft angular movement of the actuator during stabiliser positioning. Similarly, the ball nut operates in a gimbal to prevent binding of the jackscrew. The operating range of the ball nut, and stabiliser movement, is limited by stop nuts at each end of the jackscrew.

The primary method of rotating the jackscrew is by means of a 115-volt three-phase motor connected to gearing which also forms part of the actuator. When the motor is energised via a switch on the pilot's control wheel, it rotates in one direction only, and its drive is transmitted to the jackscrew through either one of two electro-magnetic clutches which, on being energised, permit jackscrew rotation in either direction, corresponding to 'nose-up' or 'nose-down' trim. To prevent the stabiliser from being motor-driven onto either of the stop nuts, limit switches are provided, and on being mechanically operated by a striker on the stabiliser they de-energise the appropriate electro-magnetic clutch circuit. In the event that the actuator ball nut is driven onto either of the stops, a mechanical torque limiter is operated to prevent damage being done to the jackscrew actuator. When the trim system is not in operation, rotation of the jackscrew under the influence of air loads acting on the stabiliser is prevented by a braking system.

In the event of malfunction of the main actuator motor system, or of the automatic trim servo-actuator system, trimming may be carried out manually by means of trim wheels in the cockpit, and by cables connected to cable drums and a disconnect clutch.

FIXED TO AIRCRAFT
STRUCTURE

MACH TRIM
ACTUATOR

ELEVATOR TRIM
CENTRING FEEL UNIT

MAIN CONTROL
VALVE LINKAGE

MANUAL ELEVATOR CONTROL

STABILIZER & ELEVATOR CO

NEUTRAL
SHIFT RODS

AFT
QUADRANT

STABILIZER

AUTOPILOT
SERVO MOTOR

STABILIZER
AFT DRUM

JACKSCREW

STABILIZER
TRIM WHEEL

BRAKE RELEASE
KNOB

MAIN ELECT
ACTUATOR

ELECTRIC TRIM
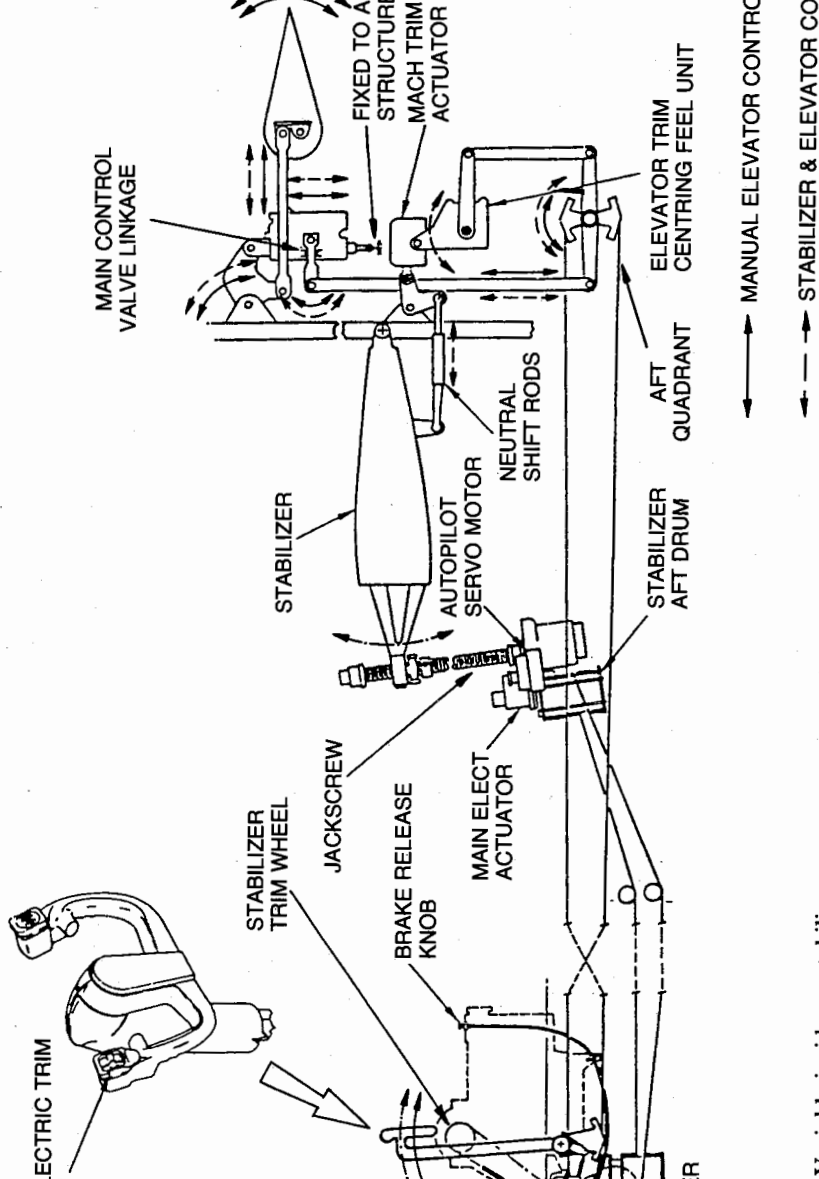
Variable incidence stabiliser.

The elevators provide for the control of manoeuvres in the pitching plane with the stabiliser in a trimmed position. They are operated by any one of four methods: (i) manually and by movement of the control column, (ii) stabiliser movement through a neutral shift rod mechanism, (iii) inputs from a Mach trim system, and (iv) inputs from an automatic control system.

When the control column is manually operated, then through the cable system the aft quadrant is rotated and this, in turn, operates a main control valve within the elevator power control unit, to admit hydraulic pressure into the unit's main actuator. The unit is thereby displaced to position the elevators in the manner of a power-operated system. The operation of a power control unit is described in more detail in page 209. A parallel input from the aft quadrant into the trim centring and feel unit provides the pilot with artificial feel.

The elevators are also displaced whenever the stabiliser is moved to a trimmed position, and as will be noted from fig. 1.33, this is effected through a neutral shift rod mechanism. The reason for elevator displacement, or neutral shift as it is called, is to augment the control authority of the stabiliser. The shift rod mechanism transmits its movement to the main control valve of the elevator power control unit via bodily displacement of the Mach trim actuator and the linkages from the aft quadrant of the control system. Fig. 1.34 illustrates the range of elevator neutral shift as the stabiliser is trimmed, and with the aircraft on the ground; in flight the range is slightly less due to aerodynamic loads acting on the stabiliser and elevators. It will be noted from the graph that the range also varies, dependent on whether the Mach trim actuator is extended or retracted.

The Mach trim actuator (see also page 231) is part of the neutral shift mechanism and at speeds below a set value, e.g. 0.7M, it is fully extended. As speed increases above this value, the actuator retracts to rotate the trim centring and feel unit, and through the aft quadrant and linkages the main control valve of the power control unit causes the unit to move down and so give an upward displacement of the elevators.

When the automatic control system is engaged the linkage to the main control valve of the power control unit is locked in its central position. Displacement of the elevators as required by a command signal from the pitch channel is carried out by supplying the signal to a transfer valve in the elevator power control unit. This admits hydraulic pressure to the unit's main actuator, and as in the case of manual control, the unit is displaced bodily to move the elevators to the commanded position.

Flying tail

The stabiliser is connected directly to the pilot's control column via cables, linkages and hydraulic actuator. It is also interconnected with geared elevators which are positioned as a function of stabiliser movement. In the
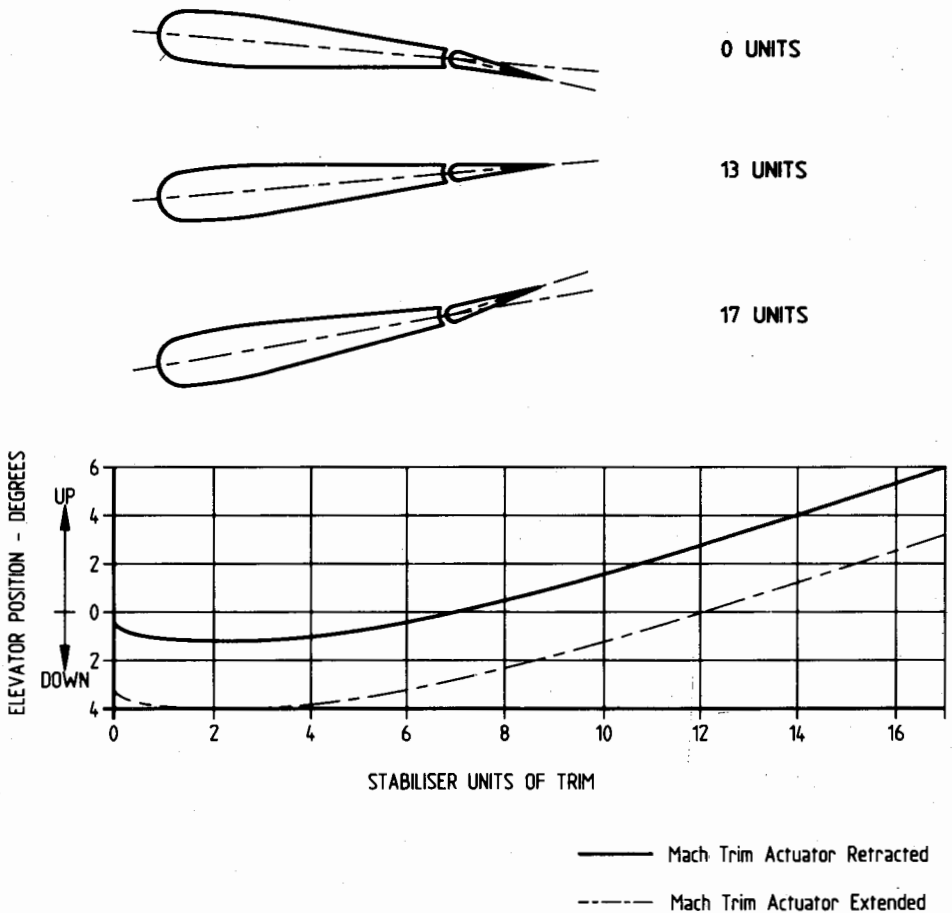
Fig. 1.34 Elevator neutral shift.

high-speed flight regime, the stabiliser serves as a pitch control surface that is significantly more efficient than an elevator whose effectiveness can be down-graded by shock-induced flow separation and structural flexibility effects. At cruise speed, the elevators are faired with stabiliser to provide minimum drag. At approach speeds, the elevators are up in relation to the stabiliser to produce a negative camber for maximum low-speed pitch control power.

The elevators are connected to the stabiliser by means of 'up' and 'down' cables that pass around a forward cable drum and a rear quadrant; the ends of the cables are fixed to the fuselage structure. When the stabiliser is actuated such that its leading edge is moved either up or down, then, as may be seen from fig. 1.35, the cable drum also moves with the stabiliser.
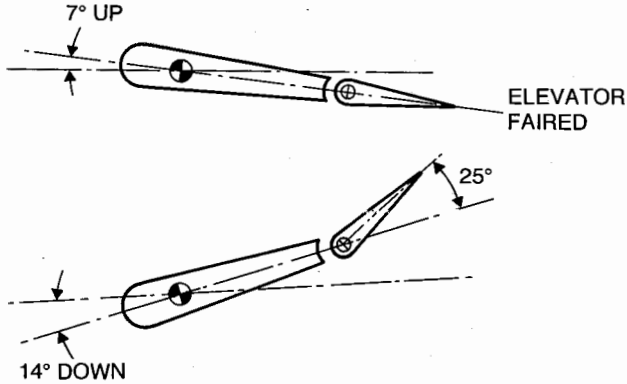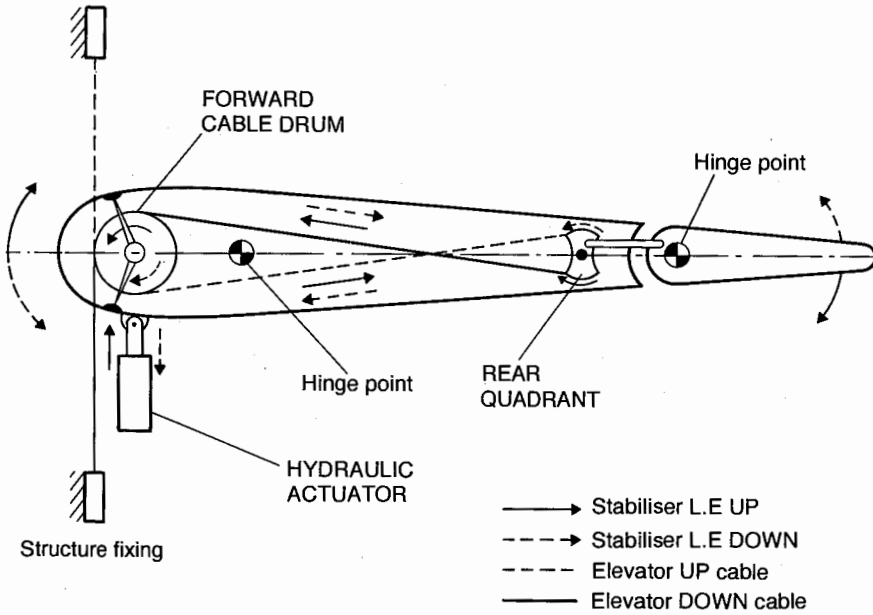
Fig. 1.35 'Flying tail'.

Since the 'fixed' cables pass around the drum, its movement has the effect of lengthening one cable and shortening the other, thereby producing drum and rear quadrant rotation, and also elevator deflection in a direction opposite to that of the stabiliser. The 'gear ratio' is such that as the stabiliser leading edge moves from 1° up to 14° down, the elevators are automatically moved from the fixed position to 25°, trailing edge up.

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Mach trim

In aircraft which are capable of flying at high subsonic speeds, and of transition to supersonic speed, larger than normal rearward movements of the wing centre of pressure occur and in consequence large nose-down pitching moments are produced; the attitude change being generally referred to as 'tuck under'. The attitude change is, of course, corrected and trimmed out by designing such aircraft so that they have the essential stability characteristics and trimming method, e.g. the variable incidence horizontal stabiliser already referred to. However, at a certain ratio of aircraft speed to the local speed of sound, or Mach number as it is called, compressibility effects arise which make the counteracting nose-up pitching moment produced by trimming the horizontal stabiliser to a negative angle of attack position, less effective as aircraft speed increases. Under manually controlled flight conditions, this would necessitate the pilot having to make prolonged trim changes and hold higher forces on the control column when displacing the elevators relative to a specific trimmed condition. It is usual therefore to install what is termed a Mach trim system (see also page 231) which automatically senses increases of speed above the appropriate datum Mach number and, by means of servo coupling, automatically re-adjusts the position of the horizontal stabiliser thereby maintaining the pitch trim of the aircraft.

Another example of Mach trimming is the one adopted for 'Concorde' and shown schematically in fig. 1.36. During transition from subsonic to supersonic speed, the nose-down pitch moment (diagram *a*) is counteracted by transferring fuel from a front 'trim tank' to a rear 'trim tank' so that the centre of gravity is moved rearwards thereby removing the couple introduced by the movement of the centre of pressure (diagram *b*).

Lift augmentation

During take-off, approach and landing, the speed of an aircraft corresponds to values in the low end of its overall speed range, and since there is a fixed relationship between these speeds and stalling speed, then as high a maximum lift coefficient as possible is required at the lower speeds. In order to achieve this, secondary control devices are provided and when operated they augment the lift generated by the wings. The devices most commonly used are flaps, slats and slots.

Flaps

Flaps are devices that form part of the trailing edge sections of the wings of an aircraft, and in such a manner that they can be lowered through selected angles and thereby increase the camber of the wing sections. In many cases,

a nose-down pitching moment

b balancing of pitching moment
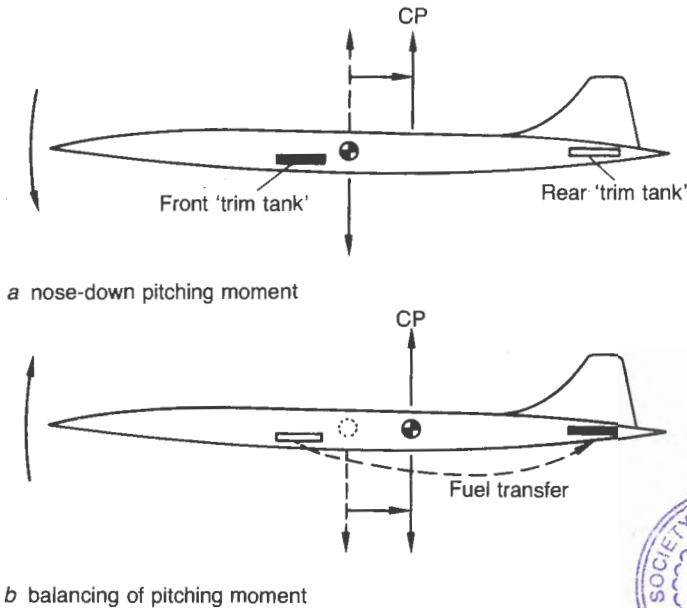
Fig. 1.36 Mach trimming.

they can be extended simultaneously so that the wing area is also increased. Flaps vary in their configuration and methods of operation, dependent upon the type of aircraft in which they are installed. The split flap is the simplest type, deriving its name from the fact that it is part of the lower section of the wing, split from the remainder of it and deflected about a simple hinge arrangement by a mechanical linkage system throughout. The plain flap is a moveable section of a wing trailing edge deflected about its hinge access by a hydraulic actuator. The most widely used type is the Fowler flap, which not only increases the camber of the wing section but also increases the wing area. They are lowered and extended by hydraulic actuators although in some smaller types of aircraft, actuation is by means of an electric motor.

When flaps are lowered, the angle of attack under the prevailing conditions is reduced, and as a result of the change in wing section camber, there is a fixed and lower stalling angle for each increasing flap angle. The change in pressure distribution, and rearward movement of the increased lift-force also produces a nose-down pitching moment which is generally opposed by an increase in the downwash at the tail of the aircraft, this being an added effect of lowering flaps.

Although lift is increased by lowering flaps, drag also increases progressively but under approach and landing conditions this can be used to advantage, and for each type of aircraft flaps are lowered to corresponding angular positions. At the approach position, the increase of lift enables a

lower approach speed to be made and since the stalling speed is decreased the aircraft can touch down at a lower speed. For landing, the flaps are fully lowered and a higher drag permits a steeper approach without speed becoming excessive; in other words, the flaps have a braking effect during approach and during the period an aircraft 'floats' before touch-down.

The increased lift coefficient when flaps are lowered, can also shorten the take-off run of an aircraft. Therefore, for each type of aircraft a recommended angle or 'take-off flap' position is always specified and which will give the correspondingly best lift/drag ratio.

The effectiveness of flaps may be considerably increased if the air is made to follow the lowered surface without a tendency to separate or become turbulent, particularly at high angles of attack and large flap angles. One method of achieving this is by designing flaps such that when lowered a slot is opened at their leading edges. This slot allows air to flow over the upper surfaces of the flaps, thereby re-energising the air so that a smooth flow is maintained over the whole cambered surface formed by the wings and flaps. In Fowler type flaps, two or even three slots may be opened during lowering and extension as shown in fig. 1.37 *a*.

An example of the use of Fowler flaps in the take-off position is illustrated in fig. 1.38.

In the case of swept-wing aircraft, a large increase in lift-dependent drag can occur at high angles of attack, mainly because of air-flow separation at the wing leading edge, and a spanwise flow of the boundary layer. These aircraft are therefore provided with leading edge flaps which are extended to increase the leading edge camber, the extension being in conjunction with that of the trailing edge flaps.

The most common type of leading edge flap is the Krueger shown in fig. 1.37 *b*. Another type is the variable-camber flap which has a flexible surface made of fibreglass so that as it extends, the actuating linkage causes the surface to vary its contour in order to maintain a more efficient lift characteristic. In some aircraft, both may be used in combination; for example in the Boeing 747, Krueger flaps are located at the inboard leading edge sections, and variable camber flaps are at the outboard sections. Another method of increasing leading edge camber and of reducing air flow separation, is 'leading edge droop', the droop being obtained by lowering a section of the wing leading edge itself.

Slats

For some types of swept-wing aircraft, the alternative slat method of augmenting lift and of reducing spanwise flow of the boundary layer may be adopted. For example, both the McDonnell Douglas DC–10 and the 'L–1011 Tristar' have slat installations (see fig. 1.39) as opposed to leading edge flaps which, as already noted, are adopted in the Boeing 747.

Slats are cambered surfaces which, in the retracted position, lie flush

Outboard flaps

Inboard flap

Landing position

Slots

Take off position

Triple-slotted Fowler flap

(a) Trailing edge flaps

Folding nose

Krueger flap

Folding nose

Variable camber

(b) Leading edge flaps
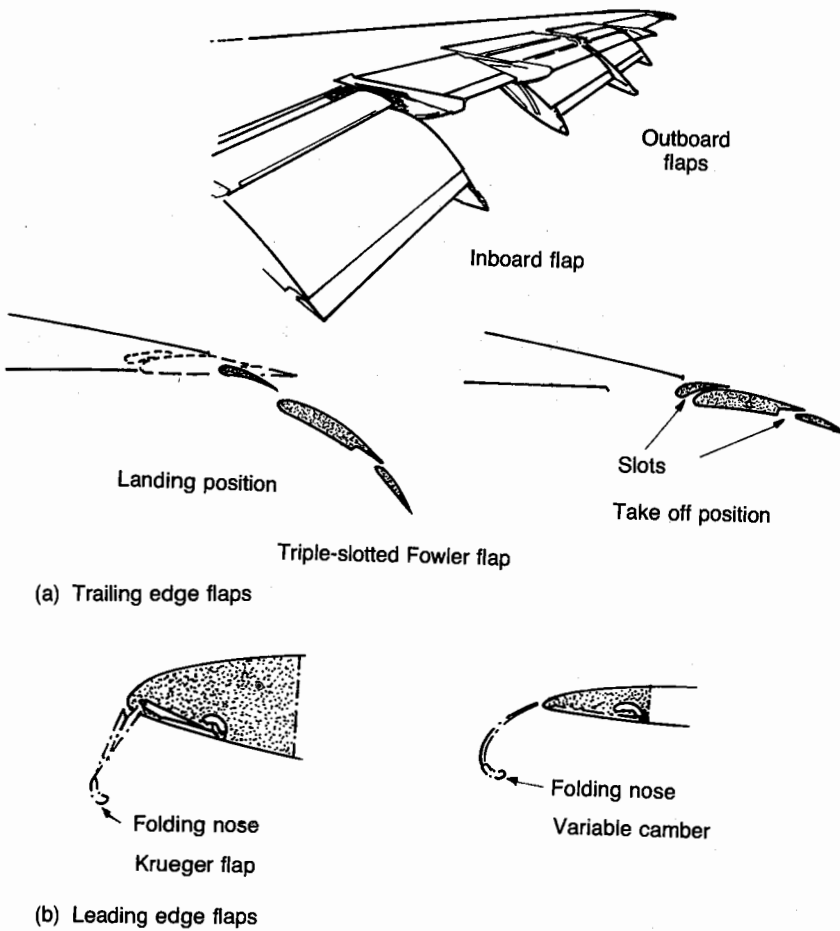
Fig. 1.37 Flaps.

with the upper surface of the leading edge section of a wing. They are arranged in sections along the leading edge and are extended and lowered by hydraulic actuators thereby providing an increase in leading edge camber and resulting increase in the lift coefficient. As will be noted from the diagram in figure 1.39 when a slat is in the extended position, a slot is formed between it and the upper surface of the wing leading edge. The air flowing through the slot is accelerated by venturi effect and this re-energises the boundary layer and prevents separation of air flowing from the wing. Since a number of slat sections are installed, then the combined effect of the airflow through each open slot is to prevent spanwise flow of the boundary layer, and any tendency for wing-tip stalling to take place.

Fig. 1.38 Airbus A300 (reproduced by courtesy of *Airbus Industrie*).

Fixed leading edge
panels & fairings

Leading
edge SLATS
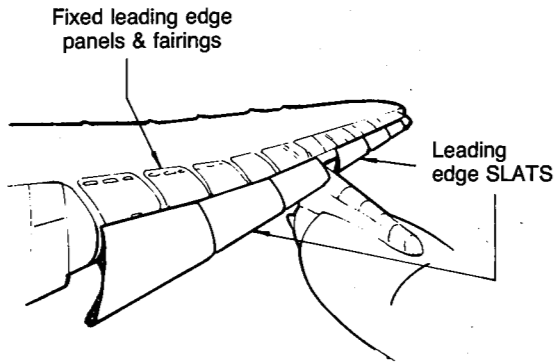
Fig. 1.39 Slats.

Spoilers

Spoilers are flap-type control surfaces located in the upper surface of a wing
in such a way that they can be raised to selected angular positions. As their
name suggests they 'spoil' the airflow over the wing, and since this pro-
duces a change in both lift and drag they can be used to provide specific lift
control features for the purpose of: improving lateral control, aerodynamic
braking for vertical speed control during descent, and for reducing roll-out
after landing. In other to illustrate the basic function and operating funda-
mentals of spoilers, we many consider the system as applied to the McDon-
nell Douglas DC–10 (see fig. 1.24). Five spoilers are located in each wing
immediately forward of the trailing edge flaps and are used in conjunction
with the ailerons. The commands for the various lift control functions are
applied to lateral control mixer units, the outputs from which operate the
spoilers via hydraulic actuators. During the lateral control mode of oper-

ation, displacement of the ailerons causes the mixer units to control the spoilers in such a manner that the spoilers on the down-going wing are raised thereby assisting the aileron on that wing. The spoilers on the up-going wing remain in the retracted position.

During descent, it is essential for both altitude and vertical speed to be closely monitored, and conventionally it is effected in the long-term by engine power, and in the short-term by the pitching moment established by the elevators. However, in aircraft such as the DC–10, the spoilers are used for altitude and vertical speed control during descent, since in the extended position they control lift by direct means. When this mode of operation is required, a control lever in the cockpit is operated manually and causes the hydraulic actuators to partially extend all the spoilers on each wing. If it is necessary to apply lateral control of the aircraft while under direct lift control, the ailerons are displaced in the normal way and by means of the mixer units, the spoilers on the down-going wing are extended further and those on the up-going wing are retracted.

Reduction of the landing run of the aircraft after it has touched down is achieved by fully extending the spoilers on each wing: this is generally known as the 'ground spoiler' or 'speedbrake' mode, and is also controlled by the control lever of the cockpit. The lever may be set in an 'armed' position so that the spoilers extend automatically when the main landing gear wheels spin on landing, or the lever may be positioned to provide direct manual control of the spoilers. In the armed position, the hydraulic actuators are controlled by signals from wheel-driven transducer units.

B. HELICOPTERS

The study of principles of flight thus far has been concerned with aircraft that are moved by thrust developed by propellers or exhaust gases from turbojet engines, and which derive lift from fixed wings. However, as a study of automatic flight control would not be complete without some discussion of helicopter systems, it is necessary to have some understanding of the principles of flight of rotary wing aircraft.

The helicopter is more precisely defined as 'a rotorcraft deriving lift from power-driven rotors rotating about axes that are vertical or nearly so when the aircraft is in forward flight'.

The unique aspects of the helicopter are the ability to ascend and descend vertically, hover with respect to the ground with winds from nearly any direction or speed, and to achieve forward flight.

General

The construction of the typical helicopter is shown in fig. 1.40. This serves as a useful example on which to introduce the general principles of
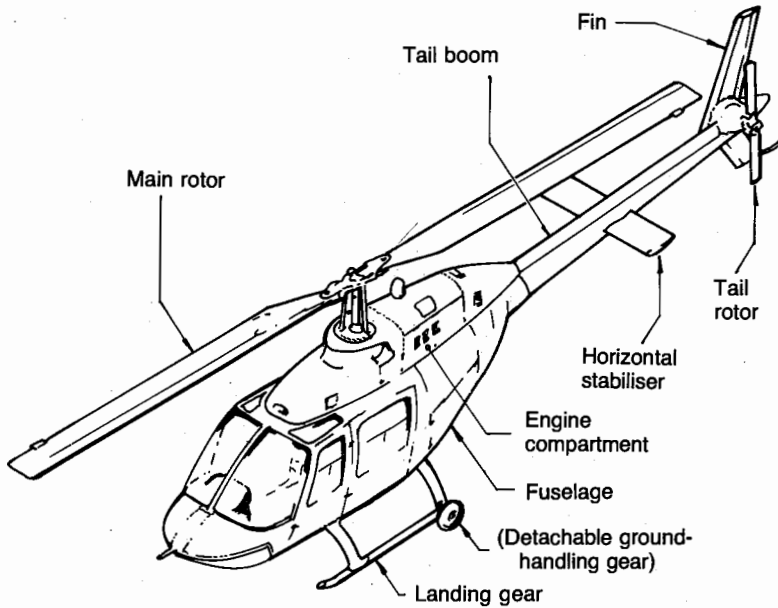
Fig. 1.40 Typical helicopter construction.

helicopter flight. Description of many of the detail features such as electrical and oil systems, transmissions, fuel controls and so on are omitted. The main features with which we are concerned are a single main rotor, a tail rotor, an engine, fuselage, and horizontal and vertical stabilisers.

The main rotor is unique in that as a single system it performs all the basic functions required for any air vehicle: namely, propulsion, support and controllability.

The rotor system on which we shall concentrate is three bladed, with hinges to permit each of the blades to flap up and down with respect to the vertical axis, lead and lag with respect to the azimuth position and to change blade pitch. This is known as a fully-articulated rotor. The rotor blades are high aspect ratio rotating wings of non-symmetrical aerofoil section.

The engine (or engines) drive the main rotor through a gearbox or transmission, and the tail rotor is directly coupled to the main rotor, normally by a drive shaft, as shown in fig. 1.41. For this example, the engine will be a gas turbine with a rotor speed governor. Only one engine is used for this discussion, as the total power is not important, and in a helicopter the loss of power from one engine does not introduce any asymmetric effect, as on a fixed-wing aircraft. In the event of a total loss of engine power, the transmission incorporates a free wheel system which during the ensuing descent, permits the rotors (main and tail) to rotate and
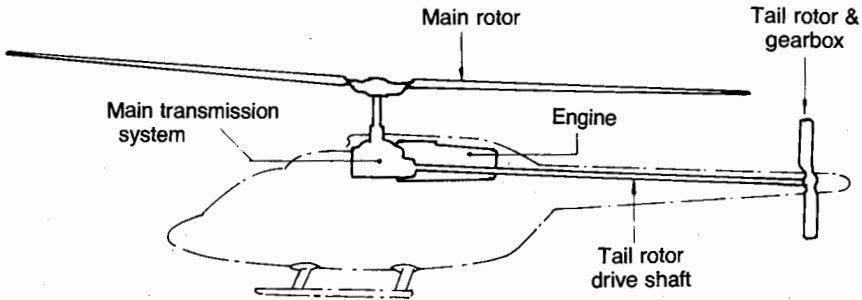
Fig. 1.41 Rotor transmission system.

also ensures continued operation of ancillary systems such as hydraulics and electrics. This descent without power is known as *autorotation*.

In order to carry out the particular flight manoeuvres associated with a helicopter, a flight control system that will produce both control and stability must be fitted. As discussed earlier, in a fixed-wing aircraft a pilot changes thrust by means of the power control (or throttle), and changes the direction of flight by means of separate pitch-and-roll control surfaces. In a helicopter the control of pitch and roll is combined with control of the magnitude of lift (or thrust). The flight control system must be capable of changing both the magnitude and size of the thrust vector. In both types of aircraft yaw control is accomplished by rear-mounted control surfaces or devices, i.e. in a fixed-wing aircraft, the rudder; and in a helicopter, the tail rotor.

The main rotor controls in a helicopter are called the *collective* and the *cyclic* and their functions are as follows:

- Collective control changes the pitch of all the main rotor blades by the same amount (or collectively) with respect to the main rotor shaft. In a general sense this controls the magnitude of the thrust vector.
- Cyclic control changes the pitch of the rotor blades individually with respect to azimuth as they move around the rotor disk. During each cycle the blades will experience a variation in pitch in a cyclical manner. In a general sense this controls the direction of the thrust vector.

An important control to counteract the reaction of the main rotor is the tail rotor. Typically, the tail rotor changes thrust only, and is controlled by the pedals. It must be able to provide thrust both to the left and the right, to cater for the great variety of wind and power conditions that the helicopter will encounter.

A helicopter is able to descend or ascend vertically, to fly with no airspeed and (within limits) to maintain a ground position with winds from

any direction. In forward flight it is able to perform nearly all of the manoeuvres of a fixed-wing aircraft.

Rotor system

A rotor blade on a helicopter is subject to a tremendous variety of airspeeds and stresses. In order to remove the stresses and prevent a large dissimilarity of lift from one side to another (discussed later) a variety of methods of hinging the blades has been developed. There are several types of rotor system in common use, namely:

- Semi-rigid or teetering,
- Articulated,
- Hingeless.

The semi-rigid or teetering type of rotor system is found only on the two-bladed helicopter. It has an underslung hinge and rigidly connects the two blades. Differences in lift are compensated by letting the two blades find their own equilibrium, and differences in drag force between the two blades are absorbed in the head. This type of rotor head has the advantage of being simple and easy to maintain; it takes up very little room in a hangar and has good hover efficiency. It can be prone to vibration, produces a pronounced two-bladed noise and does not tolerate low 'g' flight conditions unless specifically designed for these manoeuvres.

The articulated rotor system incorporates three hinges close to the hub. The reasons for these hinges is beyond the scope of this book, as they involve very complex aerodynamic and physical descriptions. The hinges are known as:

- Flapping hinges that permit the blades to flap up and down with respect to the main rotor hub to achieve equality of lift across the rotor disk.
- Dragging hinges that permit the blades to lead and lag in azimuth from a nominal position (i.e. in a three-bladed rotor the blades will not always be separated by 120°). This is due to what is termed *Coriolis effect*, whereby as a blade flaps up and down, its centre of gravity moves closer to and farther away from the hub. To conserve angular momentum, it is necessary for the blade to speed up or slow down, inducing stresses on the blade root, unless it has been permitted to lead and lag.
- Feathering or pitch change hinges that permit the blades to change pitch as they rotate to control the helicopter.

The 'hingeless' rotor is not a rigid rotor. Due to the nature of rotor systems in general, there can never be a truly rigid rotor, and the hingeless rotor has a metal or fibre composite structure that takes the place of the

hinge. These structures restrict the amount of blade movement to within certain limits. Flapping, dragging and feathering can still occur.

Axis systems

A helicopter has the same degrees of freedom of movement as a fixed-wing aircraft, and uses the same axis convention (see fig. 1.9). There is, however, an additional system of axes that relates to the rotor system, since the rotor blades can assume different angles with respect to the rotor mast (see fig. 1.42). The axes are defined as follows:

- Shaft axis – about which the blades rotate (fig. 1.42 b);
- Hub axis – that is, perpendicular to the hub, which itself is free to tilt with respect to the shaft;
- Rotor axis – that is, perpendicular to the plane of the tips of the blades as they go around the hub, as various flight conditions will change the relationship between the blades and the hub.

For descriptive purposes, the tip path plane, that is, the plane described by the tips of the rotor blades as they rotate, is assumed to be a flat disc that corresponds to the angle of the main rotor hub. It is often referred to as the rotor disc. It will also be assumed that the rotor produces a uniform lift across the disc when hovering, and that all lift acts perpendicular to the disc.

When the blades are rotating, they are producing lift, which makes them move upwards and centrifugal force is also attempting to make them fly outwards. The combination of these two forces creates what is known as coning angle (shown in fig. 1.42 a). This will vary with rotor speed, amount of lift produced, and several other minor factors.

Lift

In the zero airspeed hover, the main rotor blades produce equal amounts of lift regardless of the azimuth. As the helicopter moves forward and airspeed increases, the blades will produce different amounts of lift. The reason for this is that the blade which is advancing into the airstream experiences a total speed equal to speed due to blade rotation **plus** the forward speed. For the blade that is retreating the total speed experienced is equal to the speed due to blade rotation **minus** the forward speed. At any airspeed above approximately 20 knots, this difference in the lift created on the two sides of the disk can become significant. Fig. 1.43 shows this effect. To maintain equal lift on both sides of the disk, the pitch of the blades is changed by the cyclic, and the blades are also free to flap up and down.

Axis of
Rotation

Coning angle

Tip-path plane
(rotor disc)

Lift

Plane of rotation

Centrifugal
force

Rotor head

Rotor shaft

Shaft
axis

Chord line

Blade
pitch

(a) Axes in
perfect hover
(no wind)

Plane of rotation

Rotor
shaft

(c)

Shaft axis

Axis of
rotation

Lift

Constant pitch,
no taper

Variation in
pitch and/or taper

(b) Axes in forward flight,
disc tilted forward

100%   50%   0
Blade radius from hub

(d)
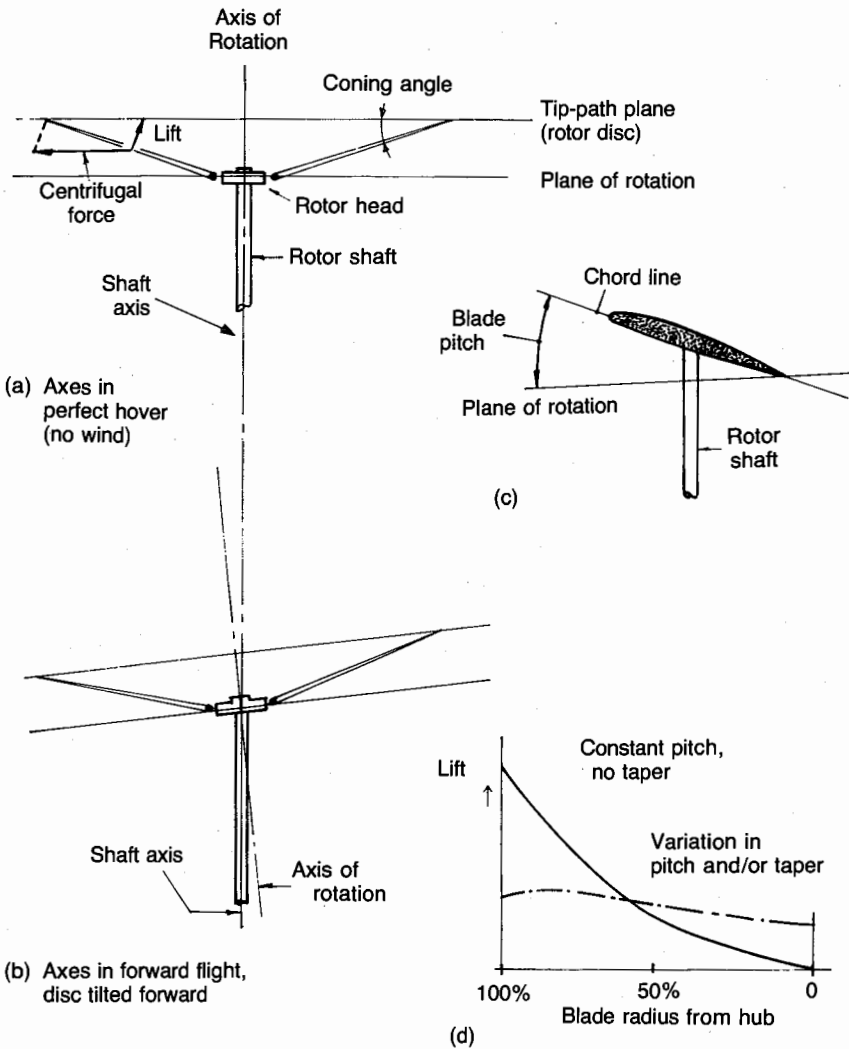
Fig. 1.42 Terminology.

Control system

The main source of controlling moments on the helicopter is the main rotor, and it must be free to change magnitude and direction of thrust. This is typically accomplished by means of a swash plate mechanism, as shown in fig. 1.44. The swash plate has a fixed portion that is attached to the airframe and is moved by the pilot's controls. Because of the high aerodynamic forces that are encountered by the rotor blades, it is difficult to control

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Direction
of Rotation

Relative airflow

Plane of rotation

θ

α

Advancing blade

Retreating blade

α

θ

θ Angle of attack
α Collective pitch
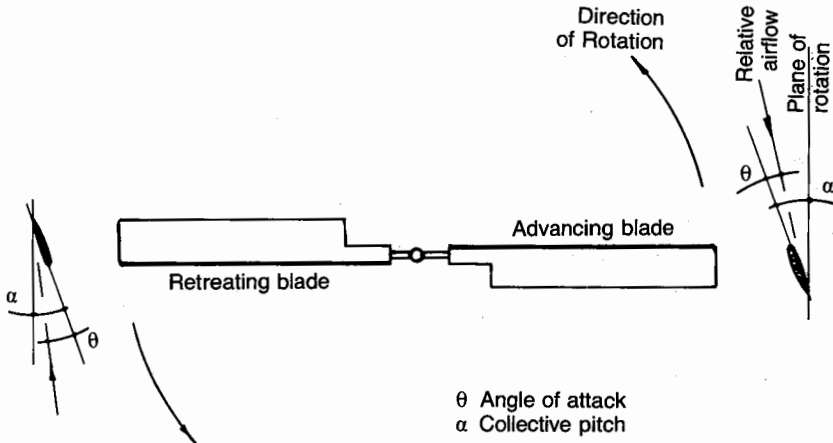
Fig. 1.43 Advancing/retreating blades.

manually any but the smallest of helicopters. For this reason, hydraulic servo-actuators are installed in the control system to assist the pilot in applying control. These position the fixed swashplate, in pitch and roll as well as for overall blade pitch. The swashplate is free to tilt fore and aft, laterally and also to move up and down on the mast. Roughly speaking, tilting is the means of controlling the direction of the thrust vector, while up and down movement controls the magnitude of the thrust vector.

Attached to the rotating part of the mast, and also connected to the fixed swashplate through bearings, is the rotating swashplate. The rotating swashplate has pitch change rods attached to it, which in turn are connected to the rotor blades. This arrangement permits the rotor blades to change their pitch with respect to the mast, and vary the magnitude and angle of the thrust vector.

Take-off and climb

As in the case of a fixed-wing aircraft, a helicopter can take off and climb only when the lift force is greater than the weight. For the fixed-wing aircraft, this is accomplished during the take off run when the airflow over the wings produces the required lift force.

For the helicopter to take off vertically, an increase in lift force is needed, and this can be considered to be overall thrust. Lift (and thrust) are produced by the pitch (and RPM) of the rotating blades and controlled by use of the collective pitch lever and the speed of the output of the engine. Raising the lever increases the pitch of the blades and the lift, and lowering the lever reduces the blade pitch and lift.
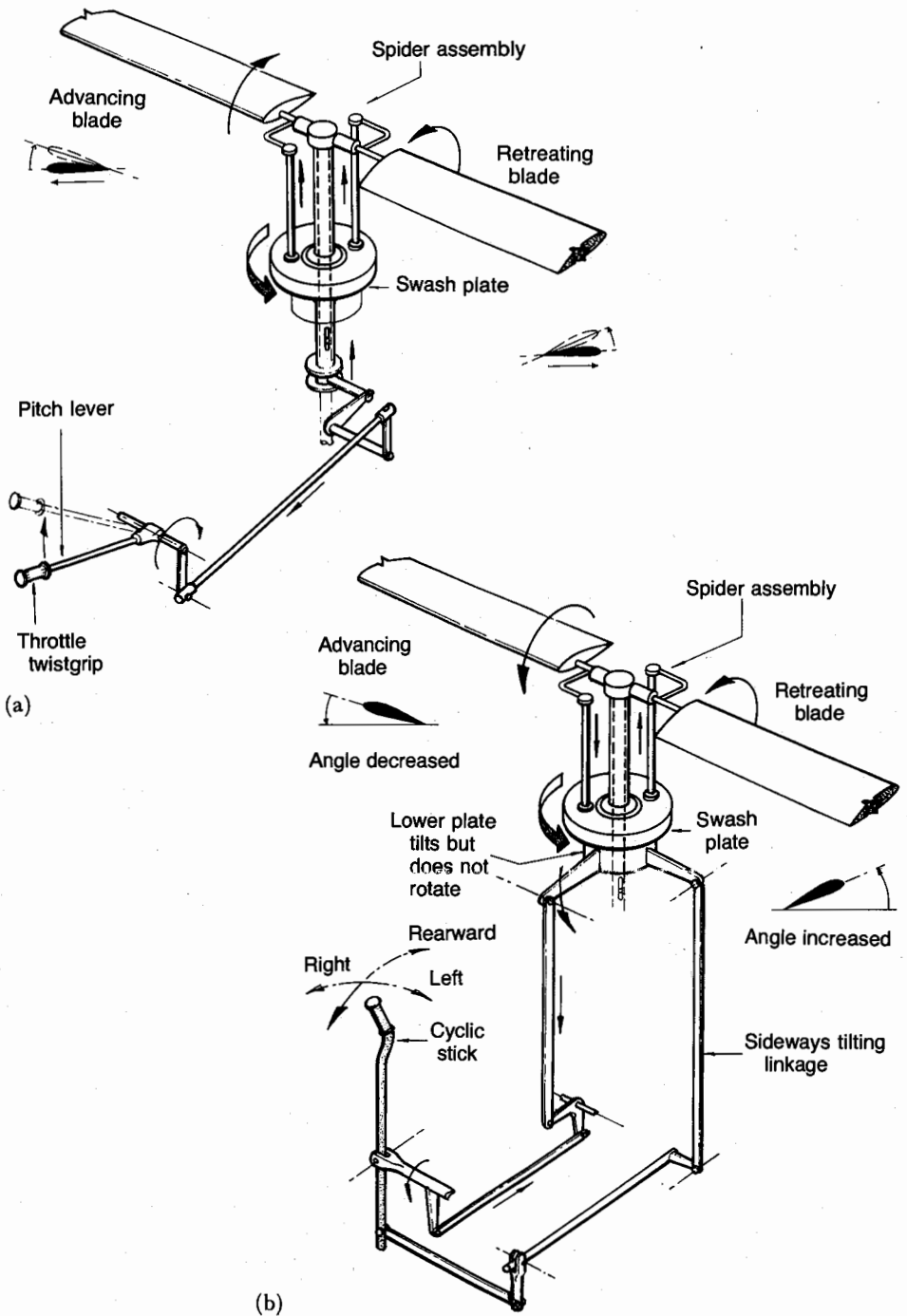
Fig. 1.44 Control of rotor, collective and cyclic.

Because changing pitch (and hence angle of attack) changes the amount / of lift of the blade, it will also change the amount of drag. The blades are designed to operate most efficiently at a set rotor RPM (RRPM). Changing the drag will cause the blades to change RRPM. This will cause the lift to change, requiring an adjustment to the blade angle to produce the same amount of lift, and so on. The combination of changing both blade angle and RRPM makes accurate manual control of the magnitude of the lift difficult. Rotor speed is normally controlled automatically by an RRPM governing system, which is part of the engine.

When the total lift (or thrust) is greater than the weight, the helicopter will lift off and climb to a height dependent upon the amount of lift that is demanded compared to the weight of the helicopter, and by a phenomenon known as ground effect.

Ground effect

In ground effect (IGE), the rotor downwash working against the ground will affect the amount of power required to maintain the helicopter at a certain height. Above a certain height, and dependent upon the size of the rotor, the disc loading and the density of the air, this effect disappears and the helicopter is now hovering out of ground effect, or OGE as it is commonly called. Out of ground effect, a change in lift or thrust from that used to maintain a steady hover will produce a constant rate of climb or descent.

Hover fuselage attitude

The attitude that the fuselage makes with the rotor (and the earth) will depend upon the position of the longitudinal, lateral and vertical position of the centre of gravity. To simplify matters, a rotor disc will always be horizontal with respect to the earth in a zero airspeed hover, and the fuselage will take up an attitude beneath it to balance all the forces. Fig. 1.45 shows a helicopter in the hover with a neutral CG (diagram (a)), and an aft CG (diagram (b)).

Tail rotor and translating tendency

An additional problem is caused by the requirement to balance the forces in the horizontal axis. The main rotor torque moment attempts to rotate the fuselage in the opposite direction to the main rotor by virtue of Newton's Third Law; this rotation is prevented by the tail rotor. The tail rotor produces thrust to balance the main rotor, but this thrust also then attempts to translate the helicopter laterally. This translating tendency is prevented by
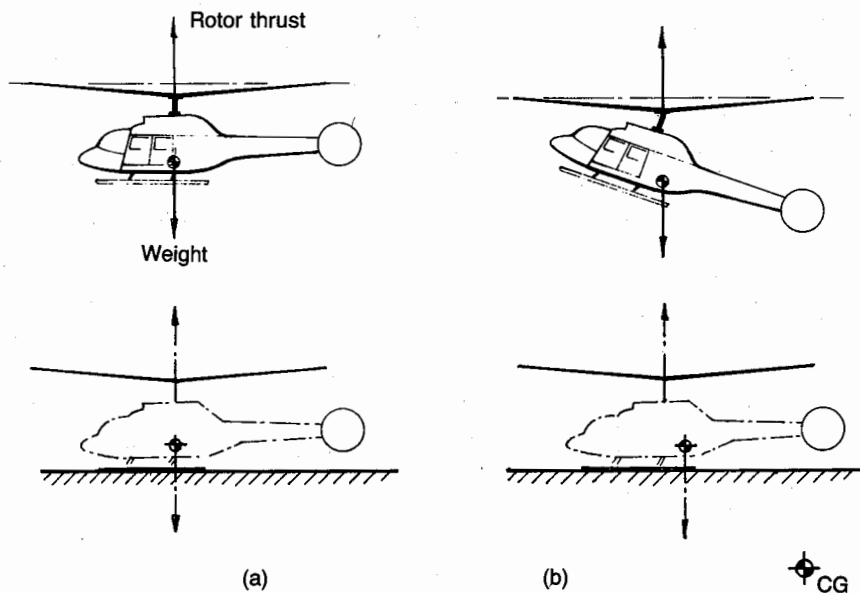
Fig. 1.45 Hovering.

a slight tilt of the main rotor thrust, either by tilting the fuselage or by a permanent offset of the transmission system. Changing the thrust at the main rotor, i.e. changing the collective position, will change the balance of forces, and the thrust of the tail rotor will have to be adjusted, together with an adjustment of the lateral tilt of the thrust vector. This is shown in fig. 1.46.

The vertical position of the tail rotor with respect to the height of the main rotor hub will have an effect on the amount of lateral tilt that is required to balance the overall moments. If the tail rotor is mounted in-line with the main rotor hub, there is little, if any, rolling moment produced. For this reason the tail rotor is often mounted on a pylon.

In summary, in a zero airspeed hover, the main rotor thrust will be vertical when viewed from the side, but tilted slightly when viewed from the rear.

Forward flight

A helicopter moves forward by tilting the thrust vector in the direction of desired movement, and this is accomplished by use of the cyclic control system shown in fig. 1.47. The exact mechanism which tilts the disc is complex, requiring the blade pitch to be changed in advance of the location
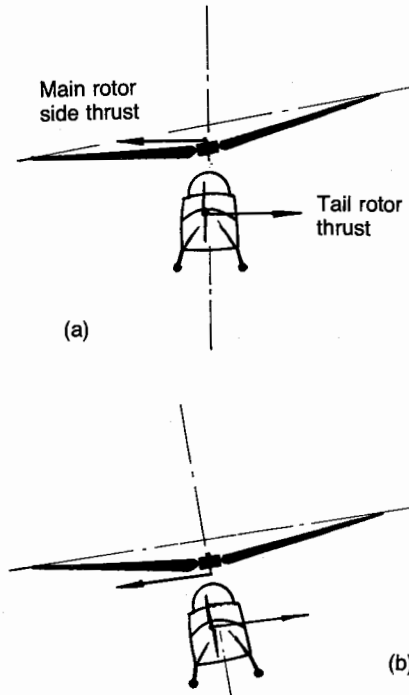
Fig. 1.46 Tail rotor drift and roll (translating) tendency.

where the blade position should be. For example, to tilt the disc (tip path plane) forward, the blade pitch must be reduced 90° before the position where the tip path plane will be lowest, and must be at the maximum blade pitch, 90°, before the tip path plane would be at its highest. The net effect is to tilt the tip path plane in the desired direction.

In steady forward flight, the tip path plane will be tilted forward with respect to the fuselage. The balance of forces (weight, thrust, drag, lift from the horizontal stabiliser, horizontal and vertical components of rotor thrust) about the CG must still be maintained, as shown in fig. 1.48.

Obviously, the higher the forward speed, the more forward the tilt of the rotor disc, and the greater the change in fuselage attitude. This may be unacceptably nose down for reasons of ride comfort, field of view from the cockpit, etc. In order to keep the fuselage near level, various methods are used. In some cases, the fuselage attitude in the hover is quite nose up, and in forward flight is near level. Another method is to use a horizontal stabiliser mounted at the rear of the fuselage, such that its cambered aerofoil section produces downward lift, to keep the fuselage level with increasing speed.
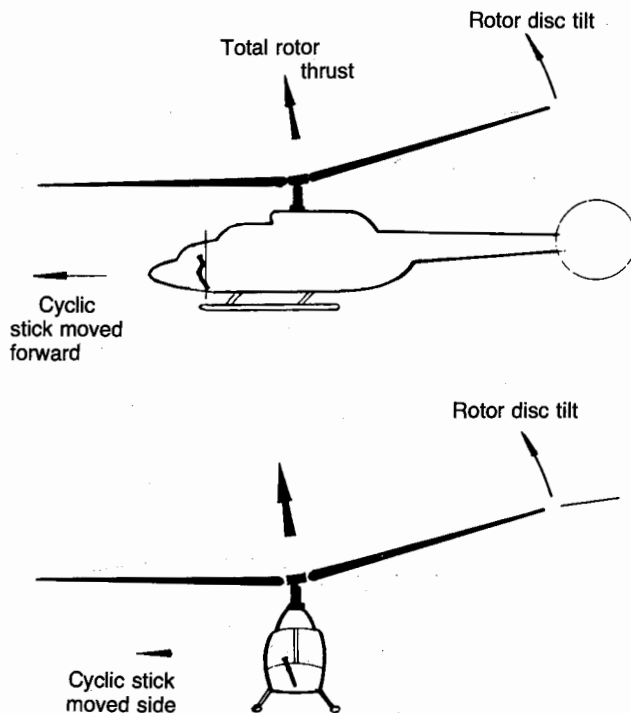
Fig. 1.47 Cyclic pitch.

Tail rotor and inherent sideslip

In forward flight the tail rotor will still produce thrust, and the main rotor and fuselage must counteract this to prevent the helicopter from turning. With the 'wings' level and the slip indicator ball in the middle, the pilot would not expect the helicopter to turn. The slip indicator ball is indicating sideforce. The net effect is that the helicopter is seldom, if ever, at zero sideslip. The angle between the fuselage and the sideslip in a steady flight condition is called the inherent sideslip, and changes with airspeed, CG, power and weight. Typical values for most helicopters in the cruise at mid-weight and CG are in the order of 3–4°.

Stability and control aspects of the helicopter

General

The main element to be considered in any discussion of stability and control is the main rotor and how it is affected by changes in airspeed, angle

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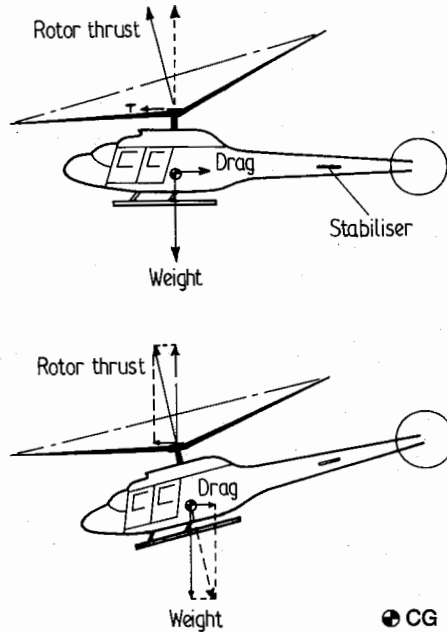
Fig. 1.48 Forces in horizontal forward flight.

of attack and relative airflow direction (sideslip) from a particular condition. This subject, which is very complex may be summarised as follows:

Changes in airspeed
The rotor will tend to provide a positive moment to return towards the trimmed airspeed; this is roughly equivalent to a fixed-wing aircraft's static stability.

Change in relative direction
The rotor will tend to move away from a change in sideslip, that is, if the sideslip is changed, it will continue to change. Note that this is for the rotor only; the fuselage will also have a large effect on the overall result.

Change in angle of attack
The rotor will continue to change angle of attack away from the trim condition if disturbed. This is unstable, and is roughly the same as a fixed-wing aircraft's manoeuvre stability. Again, the fuselage will have a large effect on the overall result.

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Longitudinal stability

The main longitudinal mode of motion of a helicopter is the 'long term', equivalent to the phugoid mode of a fixed-wing aircraft. For a helicopter this is usually a divergent oscillation, which means the pilot must make corrections to prevent the oscillation from becoming too large.

Lateral directional stability

Rotor contributions – Change in relative direction of airflow

The rotor will tend to move away from a change in sideslip – that is, if the sideslip is changed, it will continue to change. Note that this is for the rotor only.

The main contribution to lateral directional stability is the fuselage, and in particular the vertical stabiliser, which attempts to keep the fuselage aligned with the relative airflow. Obviously, this is only of significance if there is airflow across the vertical stabiliser. In the hover, the lateral directional stability is generated solely by the main and tail rotors.

Hover

In the hover, the helicopter is not influenced greatly by changes in longitudinal airspeed, and changes in side wind (lateral airspeed) are corrected by changes in lateral cyclic control and tail rotor position. Changes in angle of attack are due to vertical speed changes, and this affects the overall balance of forces with the tail rotor. In a zero airspeed hover, the helicopter has no directional stability; i.e., it will settle on any direction.

Forward flight

The modes of motion of the helicopter are slightly different from those of a fixed-wing aircraft in that typically it has slightly positive to neutral longitudinal static stability, neutral to slightly positive manoeuvre stability, neutral to divergent long-term (or phugoid) positive lateral and positive directional stability.

The end result of the stability and control aspects referred to is that a helicopter is unstable in nearly all regimes of flight, and requires a relatively high pilot workload.

Trimming

The helicopter is said to be in trim when it is in a steady, unaccelerated flight condition, which can be level flight, a climb, descent, hover and so on. There are no unbalanced moments that are causing the helicopter to change from that condition. Unlike the fixed-wing aircraft where there

are auxiliary control surfaces like trim tabs to permit these small adjustments, all the trimming is done through the main and tail rotors. Rotor systems must therefore be capable of providing not only large movements for control, but also small adjustments to permit trimming out of small disturbances.

Control effectiveness and control margin

The rotor systems are effective at all airspeeds, but there is a limit to the amount the main rotor can tilt, and/or the amount of thrust that the tail rotor can produce. Typically, the flight envelope or CG envelope of the helicopter will be limited to permit the controls to have sufficient margin of movement to be effective.

One last area that must be considered in helicopter principles of flight is the power required vs airspeed curve and the implications this has on stability and control. The basic shape of this curve (also called a power polar) is similar to that of the fixed-wing aircraft. The major difference is that it extends not just to zero airspeed, but to negative airspeeds as well. The implication is that the helicopter is able to operate at any airspeed, but that below minimum power speed, the helicopter, like the fixed-wing aeroplane, is operating in the region of reversed effect. Slowing down below minimum power speed will require more power to remain in level flight in this condition.

Operations in the hover are normally conducted with reference to height above the ground, and small changes in airspeed (i.e. caused by the wind if stationary with respect to the ground) can cause large changes in the power required to maintain a constant altitude.

Miscellaneous comments

New concepts such as the fenestron tail rotor and NOTAR concept are not covered in this book, as they are variations on the theme of the main rotor/tail rotor helicopter. Tandem, co-axial and synchrocopter configurations are also very complex in their treatment of stability and control, and the method in which automatic flight control systems are installed and used on all of these systems is not significantly different from the classical type discussed here.

TEST QUESTIONS

1 Briefly describe how the lift force is generated by a wing.
2 In what manner does a change in angle of attack affect the pressure distribution over a wing?

3 Apart from angle of attack, what other factors can affect pressure distribution?

4 The centre of pressure is the point:
 (a) about which the pitching moment is constant regardless of angle of attack.
 (b) from which total lift force acts.
 (c) at which the boundary layer airflow tends to break away from the wing surface.

5 Name the two principal types of drag, and state the components which constitute them.

6 What do you understand by the terms 'positive' and 'neutral' stability?

7 How does aerodynamic damping contribute to the stability of an aircraft?

8 As far as influence on the behaviour of an aircraft is concerned, in what manner does directional stability differ from longitudinal stability?

9 The static margin is:
 (a) the distance between the CG of an aircraft and the centre of pressure.
 (b) the CG location which produces neutral static stability.
 (c) an indication of the degree of longitudinal stability.

10 In which type of flight control system would there be no difference between stick-fixed and stick-free stability?

11 What would be the effect of a short-period of oscillation on an aircraft having longitudinal static instability?

12 In connection with directional stability, what are the principal factors affecting the size of the stabilising yawing moment?

13 What do you understand by the term 'aerodynamic coupling'?

14 If an aircraft has directional static instability, displacements in yaw will cause:
 (a) spiral divergence.
 (b) directional divergence.
 (c) Dutch Roll.

15 What is meant by the effectiveness of a flight control system, and on what factors does it depend?

16 What are the primary differences between control systems which are power-assisted and power-operated?

17 Explain why rolling of an aircraft also results in turning.

18 Under the conditions referred to in Question 17, what other effect occurs, and how can it be minimised?

19 Why is a trim control system an integral part of the flight controls of an aircraft? State the principal methods commonly adopted.

20 Why is it necessary to provide lift augmentation systems?

21 The purpose of primary stops in a flight control system is to:
 (a) prevent damage to the system when the aircraft is parked in high wind conditions.

(b) restrict the movement of the controls to their correct range.

(c) act as a stop in the event of control flutter.

22 What is the purpose of slotted type Fowler flaps?

23 What is the primary difference in operation of leading edge flaps, and slats?

24 What is meant by the term 'direct lift control', and what methods are employed to achieve it?

25 Why are the blades of an articulated rotor allowed to flap and drag?

26 In terms of relative airflow and the lift produced, how do the rotor blades of a helicopter in forward flight, differ aerodynamically from a fixed wing?

27 The forces which give rise to the coning angle are:

(a) thrust and centrifugal.

(b) lift and centripetal.

(c) lift and centrifugal.

28 When the collective pitch lever is operated, the pitch of each blade of the main rotor is:

(a) either increased or decreased by the same amount.

(b) either increased or decreased so that each blade is at a different angle.

(c) increased by the same amount to provide lift only.

29 In what way does cyclic pitch differ from that of collective pitch?

30 What are the main functions of a tail rotor system, and how is it controlled by the pilot?

31 Briefly describe how a helicopter can be turned on to any desired heading in the hover.

32 What is the purpose of the lead-lag hinge?

33 What would be the effect of putting a very large horizontal stabiliser on the back of the tail boom?

34 What effect would putting an auxiliary engine to produce horizontal thrust have on the helicopter?

35 Describe the changes in power that a helicopter would have to make as it translates into forward flight at a constant altitude above the ground.

2
Servomechanisms and Automatic Control Fundamentals

In manually controlled flight, the pilot and the flight control system of his aircraft together comprise what may be termed a man/machine closed-loop servo-system as shown in fig. 2.1. Let us consider the simple case of an aircraft which after flying on a constant heading at a particular altitude is required to continue its flight on the same constant heading, but at a lower altitude. The pilot will move the control column forward to apply downward movement to the elevators, thus causing a nose-down attitude of the aircraft and initiation of the descent. Since the descent must be made at a certain angle and rate of change the pilot will also monitor those of his primary flight instruments which detect and indicate attitude changes; namely, gyro horizon, vertical speed indicator, altimeter and airspeed indicator, and then start returning the elevators to their neutral position by pulling back on the control column. In order to level out at the new altitude, the control column will firstly be pulled further back, thereby applying upward movement of the elevators to produce a nose-up attitude of the aircraft, and then will be moved forward again to position the elevators in neutral to fly into the new level flight attitude.

In the case of a helicopter, the pilot establishes the descent by lowering the collective pitch lever, adjusting the throttle to maintain engine speed, and operating the yaw pedals to maintain balance. The cyclic control stick would also have to be adjusted to maintain descent attitude and airspeed. To level off from the descent the pilot starts raising the collective pitch lever such that he leads the desired level-off altitude by an amount dependent on the rate of descent. At the same time, adjustments are made to the pedals and the cyclic pitch, to obtain cruising speed and balance level-flight attitude as the desired altitude is reached.

From the foregoing, and much simplified, explanation of how an attitude change is effected, the particular point to be noted is that a pilot must always 'follow-up' his initial control inputs by applying secondary opposing inputs, thereby progressively removing control so that the attitude changes will be made as smoothly and as accurately as possible, and without exceeding those changes commanded by the input. Such a closed-loop
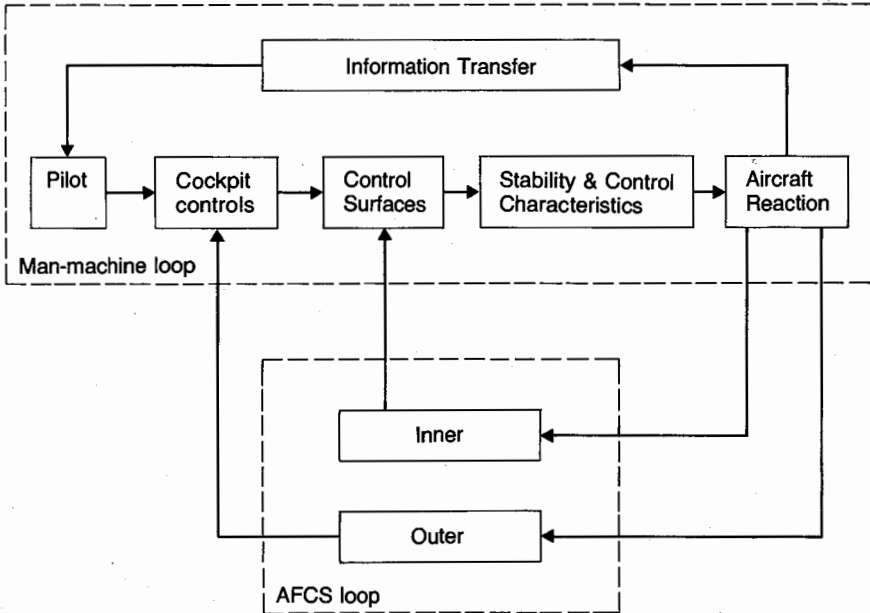
Fig. 2.1 Closed loop handling.

servomechanism technique is applied to an automatic flight control system, the 'follow-up' action in this connection being referred to as 'feedback'.

The role of an AFCS

The role of the AFCS is generally to:

(i) Overcome a stability and control deficiency, e.g. preventing a divergent phugoid or Dutch Roll mode;

(ii) Improve the handling or ride qualities, in general, or in some specific operating mode, e.g. holding altitude or airspeed, turn to and capture of a new track, and hover augmentation for the helicopter;

(iii) Carry out a manoeuvre that the pilot is unable to perform either due to the accuracy required, or the length of time over which it is necessary to carry out the task, or the lack of visual cues etc. Some examples of this might be: precise altitude holding at cruising heights for long periods, ILS-coupled approaches in extremely low visibility, and hover at night or over water.

The response of an AFCS is much more rapid than that of the human pilot, and prevents disturbances reaching sizeable proportions. As a comparison,
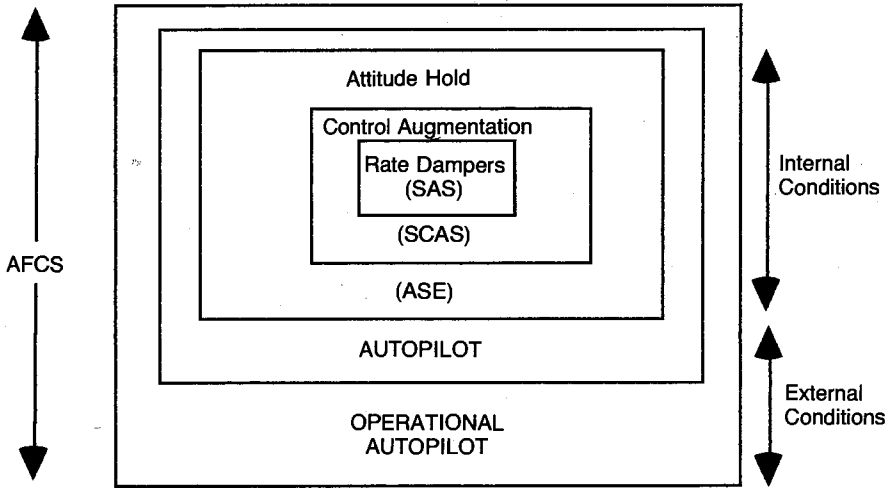
Fig. 2.2 Hierarchy of AFCS.

the human pilot can detect a change of pitch attitude of 1° (from the attitude indicator) 0.3 seconds after it occurs. There is a further delay of a 0.5 s while he decides on the amount of control correction to be applied and implements it. On the other hand, an autostabiliser or autopilot can detect a smaller disturbance (0.1°) more quickly (0.05 s) and apply an input to overcome the disturbance within 0.1 s. The combined effect is that less disturbance is seen by the airframe, which will prevent effects related to that disturbance from coupling to other areas, and so on.

The normal closed loop of man/machine control is still valid, the only difference being that the AFCS inner and outer loops modify the aircraft responses, as also indicated in fig. 2.1.

Definitions

By itself AFCS is generally understood to mean any flight control augmentation that assists the pilot in flying an aircraft. Knowing exactly what terms mean, and the various contexts in which they are used (and misused) will help immensely in understanding any AFCS. It should be noted that the definitions used in this book may differ from those used by manufacturers.

An example of confusion is in the term 'yaw damper', which commonly means an AFCS related to yaw, but there are aircraft which have mechanisms to prevent rapid movement of the pedals, which are also called yaw dampers.
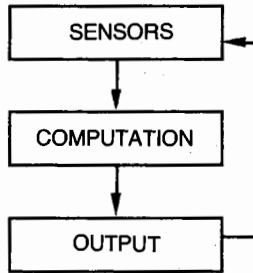
Fig. 2.3 Main components of an AFCS.

Hierarchy of an AFCS

Figure 2.2 illustrates the various 'layers' which can exist within an AFCS.

Internal versus external conditions

Internal conditions are those derived from sensors within an AFCS, and relate to pitch, roll and yaw attitudes, and their associated rates and accelerations. External conditions relate to airspeed, altitude, track, and other navigational information derived from sensors external to, but integrated with, the AFCS. The 'inner' loop of the AFCS typically handles internal conditions, that is, items that are independent of outside events, while the 'outer loop' may handle external conditions. Autopilots are typically outer loop devices.

Components of an AFCS

In general the components of any AFCS can be sub-divided into three distinct groups, and are arranged as shown in fig. 2.3.

- **Sensors.** These measure the relevant parameters and transmit the information in the computation group.
- **Computers.** These convert the information from the sensors into the signals which are fed to the system output devices.
- **Output Devices.** These convert the computed signals into a form which will result in the necessary aircraft control surface movements.

Sensors

Typical parameters that are used in an AFCS and the sensors used to measure these parameters are given in Table 2.1.

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Table 2.1 AFCS sensors

Parameter	Sensor	Alternate sensor
Pitch ⎫ Roll ⎬ Rate Yaw ⎭	Rate gyroscopes	Attitude gyroscopes (differentiated)
Pitch ⎫ Roll ⎬ Attitudes	Vertical referenced gyroscope	Rate gyroscopes (integrated)
Heading	Compass, flux valve, inertial platform	Attitude and heading reference system
Normal ⎫ (vertical) ⎬ Lateral ⎪ Accelerations Yaw ⎭	Accelerometer	Differentiated rate
Airspeed	ASI/Mach	Air data system
Altitude	Barometric altitude	Inertial platform
Groundspeed (along/across)	Doppler	Inertial or computed
Height (above ground)	Radar altimeter	Doppler
Sideslip	Yaw vane	Differential pressure system
Angle of attack	Vane	Differential pressure system
Controls/actuator positions	Position sensors	Linear variable displacement transducer (LVDT)
Flight path and navigation data	ILS, MLS, Inertial navigation systems	GPS, Loran

Computers

The computation performed in this part of an AFCS may be very simple or extremely complex, depending on the type of system and the functions to be performed. Electronic circuits are the most commonly used, but mechanical and fluidic systems also exist. Some of the more common functions performed by an AFCS computer are given in Table 2.2.

Derived signals
While it is possible to derive signals from other sensors on the aircraft, this approach is not without problems. For example, deriving rate from an attitude gyro is not difficult, but adds a computational delay to the circuit.

Table 2.2 Computer functions

Amplification	Increases the signal level from the sensor to a high enough level to be effective as an output. In the simplest form this equates to gearing in a mechanical system. Also known as gain.
Integration	Integrators can be used to derive pseudo-information. (i.e. rate information can be integrated to produce attitude data). There are problems integrating small (i.e. near zero) signals.
Differentiation	Differentiation is used in a similar but opposite fashion to integrators (i.e. attitude data can be differentiated to produce rate information). They cannot detect small changes in rate.
Limiting	Used to restrict the effect of parameter changes to certain limits. For example, may not permit values greater than $x°/s$ of rate to be used.
Shaping	Signal shapers are used to adapt the computer output, to produce the desired handling characteristics or flight path of an aircraft. May be considered non-linear amplifiers.
Programming	Programs are used to produce outputs that will allow an aircraft to fly predetermined manoeuvres.

It may also add 'noise' to the signals because small changes in pitch attitude (e.g. caused by a propeller out of balance) may be sensed as a rate and used in computation. It is worth noting that every computation will cause a time delay in signal processing, which can be critical in high-gain tasks.

Specific aircraft problems can also arise. More than one helicopter has had a self-induced vibration problem which resulted from the AFCS gyros vibrating and feeding this signal into the actuators. In this case, turning off the AFCS reduced the vibration level.

Output devices

The output signals from the computer must be converted to produce physical movement of the control surfaces; this is normally effected through some form of actuator, utilising either electro-mechanical, electro-hydraulic, or pneumatic principles. Not only are there different types of actuators, but they can also be used in different configurations, i.e. in series with the flying controls, parallel to them or a combination of both. Whatever type of actuator or configuration is used, the function is to produce a movement of the control surfaces.

The design of an actuator will always be a compromise between:

- Authority or percentage of control surface movement effected versus rate of travel in a failure;
- Gain or rate of normal movement;
- Size and accuracy of movement for stability and/or control.

Actuator feedback signals

In order for the computer to know the position of the actuator, it is necessary to feed back a signal to the computer. While this is simple, there is a problem if this signal is lost, which may tell the computer that the actuator has not moved, or is in the incorrect position. This can cause problems, and is often solved by duplication of feedback signals. An additional use of the feedback signal is to provide an indication to the crew of the status of the actuator, and, if suitable information is available, permit them to compare demand and actual positions.

Additionally, the response of the actuator in the event of a loss of signal, larger than expected signal or loss of feedback signal must be considered. For example, if the actuator freezes in position, the effect of any failure may be minimal, but if it suddenly moves to the extreme of travel, there will be a large effect.

These subjects are covered in more detail in the appropriate chapters on fixed-wing aircraft and helicopter systems as there are significant differences in the method of use.

Types of AFCS

Rate damping systems

These systems are the most basic building blocks, and normally are concerned mostly with stability, as opposed to control. Rate damping is important for two reasons:

(i) It will stop unwanted rates of motion from developing; and
(ii) It may make rates commanded by the pilot more predictable.

It should be clearly understood that a rate damping system will not maintain a specific attitude. For example, while in cruise with the 'wings level,' a gust produces a roll rate of 2°/s to the right. The roll rate sensor will detect this shortly after it happens, and the computer will apply a correction through the actuator to the flight control system. This will stop the roll rate, but since the corrective action happens a short time after the initial roll rate, the gust will have had some time to act, leaving a change in roll attitude (bank angle) that the pilot must correct. A rate damping system by itself will not return the aircraft to 'wings level.'

While these systems do improve the stability of the aircraft, they also

cause some problems. An inherent problem is a tendency to allow the nose to drop during turns. In a turn, the nose is moving across the horizon, and the sensors, operating in the plane of the airframe, detect this component as a pitch rate. The pitch rate sensor detects a component of turn rate and applies a nose-down control input to stop the pitch rate. This causes the nose to drop, which is not desired.

Control augmentation systems

For various reasons, a simple rate damper may not incorporate methods to tell the system that the rate it is sensing has been commanded by the pilot. The AFCS will damp out all rates that occur, and the pilot will sense this as sluggish handling as the system stops his commands from taking place. For this reason, some rate damping systems (particularly those fitted to helicopters) incorporate ways of feeding forward the commanded control input prior to the rate damping sensor detecting and damping the rate.

Autopilots

Autopilots hold the aircraft to an external condition, such as airspeed, altitude and track. This is a simple academic distinction, and the definitions get very blurred in distinguishing between autopilots and attitude hold systems when, for example, an airspeed hold is an integral part of the AFC system.

The function of the autopilot is to provide control of the aircraft as opposed to assisting with stability. With the autopilot engaged, the pilot no longer directly controls the aircraft but selects the flight conditions that he wants the autopilot to maintain; he then monitors the functioning of the autopilot.

As the autopilot is a device which controls the aircraft, it still depends, like the pilot, on the aircraft possessing adequate stability. If the aircraft is deficient in stability, automatic and/or manual flight to the desired accuracy is difficult to attain. Stability augmentation is still needed to provide the necessary short-term damping.

Stability augmentation may be physically incorporated with the autopilot; however, as it performs a separate function, it should always be recognisable as a distinct sub-system. Normally provision is made to allow stability augmentation functions to be used separately, even if fully automatic control is not required or not available due to an autopilot malfunction.

Operational autopilot

These systems perform a manoeuvre or series of manoeuvres, such as an automatic ILS approach, flare and landing in low visibility conditions or an automatic transition to the hover.

System protection

With any automatic system it is necessary to protect against malfunctions – in particular, runaways. This is normally achieved either by limiting the authority of the actuator or the rate at which the actuator can travel. In this way the pilot should always be able to override the effects of a malfunction. Some solutions to the problem of system protection are given below.

Comparators

In a simple comparator system, outputs from the sensors and actuators are monitored and compared. The comparator will disconnect the system if the attitude change sensed is in the same sense as the actuator movement, e.g. the nose is pitching down while the AFCS is commanding a nose-down input. This can be made to work even on a single lane system.

Rate trigger system

Thresholds are introduced that will trip off the system if the rate exceeds a predetermined value. By knowing the characteristics of an aircraft during a system runaway (these are normally very marked and distinguishable from normal flight), the value of the threshold switch can be set so it should not trip the system even during flight in turbulence.

Duplex system

In a duplex system two 'lanes' are used. This means not only two sensors and computers, but also two actuators, hydraulics, electrical systems and switches. A duplex system, shown in fig. 2.4 may be configured such that the outputs from the two actuators are averaged before the demand is passed to the control surfaces. If a runaway occurs in one lane, with its actuator being driven 'hard over' (rapidly travels to the end of travel) in one direction, the other lane will sense a disturbance and apply a corrective output to its actuator, and the resulting average actuator displacement should be near zero. This type of system will not totally eliminate the effects of a single-lane runaway, since a disturbance must take place before the 'good' lane can respond and counteract the disturbance. However, more complex systems not only average the outputs of the actuators but also monitor their positions. A comparator is then used to examine the actuator position signals, and if they differ by a pre-set amount the channel is disengaged. In this way it becomes quicker and easier to detect runaways and other system malfunctions.
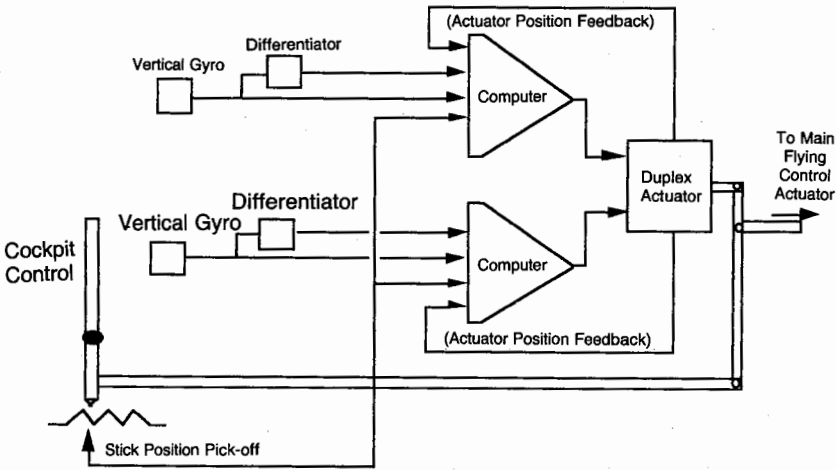
Fig. 2.4 Schematic diagram of a duplex system.

Cross-coupled feedback

Obviously, the ability of the system to sense failures would be improved if the computers could compare not only their own actuator feedback, but also the actuator in the other lane.

If a large difference existed between the two actuator positions and signals fed to the computers, it would be easy to stop the whole system in that axis. This is shown in fig. 2.5.

Triplex system

A triplex system is an extension of the duplex system except that each channel of the system has three lanes instead of two. In such a system, malfunctions are detected via a system of comparators and voting takes place to decide which lane is at fault. Such a system is said to have single failure survivability as a complete lane can be lost without jeopardising the integrity of the system. This would permit a flight or mission to be accomplished following a single failure.

Quadruplex system

This is merely an extension of the triplex system to include four lanes per channel. Its advantage is that it adds extra redundancy to the system in that two lanes can be lost before the system integrity is jeopardised.
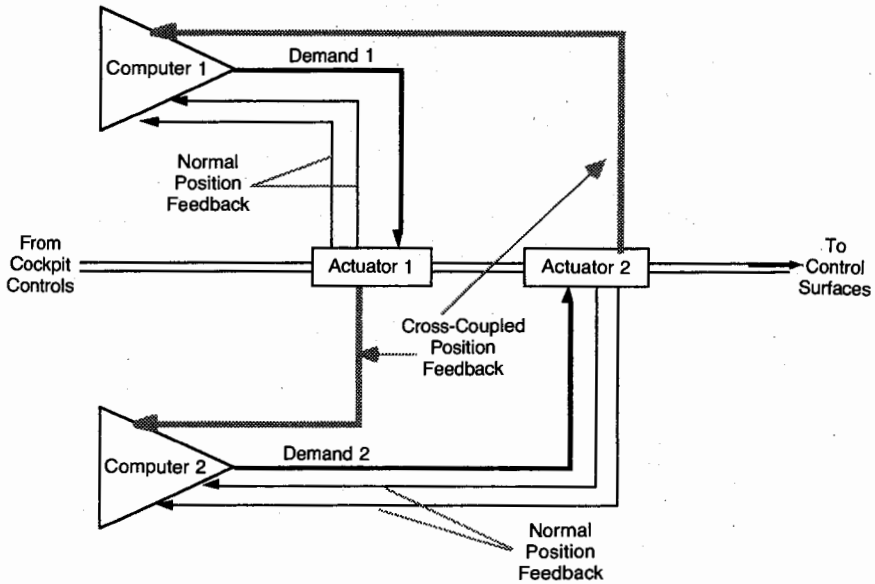
Fig. 2.5 Schematic of cross-coupled actuator feedback.

Model following system

A model following system is basically a software system that has the flight characteristics programmed into the AFCS computer, and works on the principle of comparing the predicted aircraft responses, with those measured by the system sensors. Such a system is not used on its own but may be added to a duplex system to make it a pseudo-triplex system.

Automatic change of gain

In most multiple channel systems (i.e. more than one pitch or roll channel), if one lane fails for any reason, the other lane(s) may change their gain or speed of operation to compensate. This is to leave the pilot with the same 'feel' for the aircraft. In one such system, the speed of movement of the actuator was increased to move the control surface the same amount as when both systems were in operation. The only drawback to this is that there is now increased effect of a runaway in that channel, without the protection of the second system. One helicopter type has a limitation in airspeed if one system fails because of the effect of a runaway in the remaining system, yet if both systems are off, the restriction is removed because the pilot must be 'hands on' to fly it.

Servomechanisms

A servomechanism may be broadly defined as a closed-loop control system in which a small power input controls a much larger power output in a strictly proportionate manner. In applying such a mechanism to the automatic control of an aircraft, the system must be capable of continuous operation and have the ability to (i) detect the difference between an input and an output (error detection); (ii) amplify the error signals; and (iii) control the closing of the servo loop by providing the feedback.

There are two main classes of servomechanism: (i) position control and (ii) speed control; both classes may be independently applied to automatic flight control systems depending respectively on whether they are of the displacement type or the rate sensing type. In some control systems they may also be used in conjunction with each other (see also page 149).

Position control servomechanism

A block schematic diagram of a position control servomechanism is illustrated in fig. 2.6, and from this it will be noted that it is one in which a load has to be rotated through an output angle θ_o corresponding to an input angle θ_i of a controlling shaft. The controlling shaft is, in this example, mechanically coupled to the wiper arm of a potentiometer, the signal output of which is fed to a servomotor via an amplifier. The output angle of the load is measured by a second potentiometer whose wiper arm is mechanically coupled to an output shaft. The potentiometers are electrically connected such that when their wiper arms occupy corresponding angular positions the servomechanism is in a 'null' or zero signal condition. When it is required to move the load to a particular angular position (θ_o) the controlling shaft is rotated through the appropriate number of degrees; thus the mechanism is no longer at 'null' and an error signal corresponding to angle θ_i is produced and fed to the amplifier. The amplifier has an amplification factor of K, and therefore the input to the servomotor is increased to $K\theta_i$. As the motor positions the load, the output shaft rotates the wiper arm of the second potentiometer to produce a signal corre-
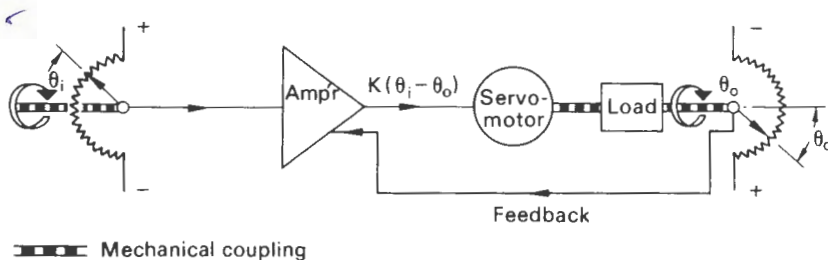
Fig. 2.6 Position control servomechanism.

sponding to an angle θ_o. This signal is fed back to the amplifier thereby reducing the input error signal to the amplifier so that the real output from this unit to the servomoter is $K(\theta_i - \theta_o)$. When the load finally reaches the position required, the servomechanism will then be at a new 'null' condition.

Speed control servomechanism

A speed control servomechanism is one in which error signals are produced as a result of a difference between voltages corresponding to input and output speeds, such signals being used to control the speed of the servo-motor and load. Referring to fig. 2.7, it will be noted that the system differs from that used for position control in that the servomotor also drives a device known as a tachogenerator. When it is required to operate the load, the servomotor is driven by an amplified input error voltage, V_i, and the motor accelerates the load towards the required speed. At the same time, the motor drives the tachogenerator which produces an output voltage, V_o, in proportion to its speed of rotation. The output voltage is fed back to the amplifier thereby reducing the input error voltage and so producing a real output from the amplifier equal to $K(V_i - V_o)$. The servomotor in this class of servomechanism (sometimes called a velodyne) is therefore controlled by differences in voltages, and will speed up or slow down until the difference is zero.

Response of servomechanism

The response to servomechanism is the pattern of behaviour of the load when a change is made to the input condition, the most important factors being the form which the input change takes and the various restraints, friction, etc., which act on the output. There are two types of input change to be considered and these are referred to as step input, and ramp input, the names being derived from the shape of the curves of input against time as shown in fig. 2.8.

A *step input* is one whereby the input (e.g. the controlling shaft of the
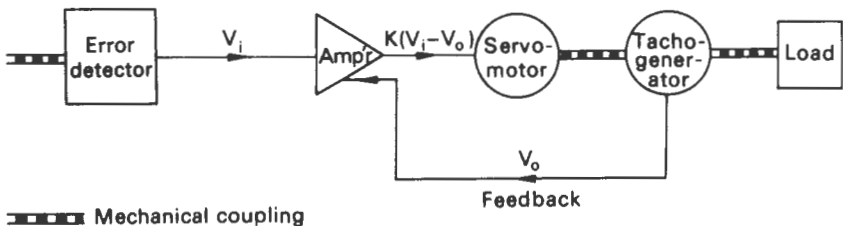
Fig. 2.7 Speed control servomechanism.
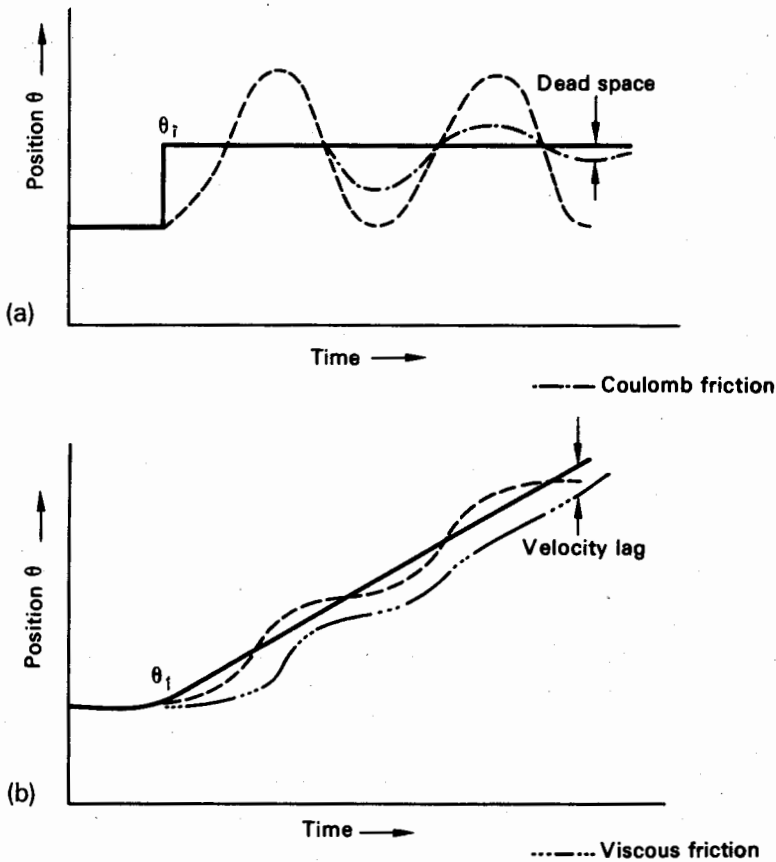
Fig. 2.8 Response of servomechanisms: (a) step input (b) ramp input.

system shown in fig. 2.6) is suddenly changed to a new angular position θ_i, from a null position. Because of the inertia of the load an angular change at the servomechanism output will not be able to follow exactly that at the input, with the result that a large error signal is produced initially. This causes the load to be accelerated to its required position, and thereby reduces the error to zero. At this point however, and although the acceleration is zero, the load has reached a steady rate of change, and so it overshoots resulting in an increase of error in the opposite sense to decelerate the load until it comes to rest in the opposite direction. By this time the error signal is equal to the original error signal but of opposite polarity, and so the load is accelerated back towards the required position and produces another overshoot. If the frictional losses in the system are negligible, a continuous oscillation is produced.

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A *ramp input* is one whereby the input is suddenly moved at a constant speed. In the early stages of the input, and while the error signal is small, the load accelerates slowly and lags behind the input. The signal increases as the lag increases, thereby building up the acceleration. Eventually the input and load speeds are equal, but since a substantial position error exists, the load continues to accelerate, the acceleration is reduced, and the load attains a constant speed at zero position error with no error signal. Thus, as in the case of a step input, a continuous oscillation is produced.

Damping

Oscillatory or transient responses of a servomechanism from whatever cause are obviously undesirable, and so it is necessary to provide some form of damping by which a load can be brought to rest in its required position with the minimum of overshoot.

Varying degrees of damping can be applied (see fig. 2.9). Using only inherent friction *light damping* is achieved. If there is too much extra viscous friction, the system is *heavily damped* and a very sluggish response is produced. The degree of damping which just prevents any overshoot is known as *critical damping*. Damping which allows one small overshoot and gives the smallest settling time is known as *optimum damping*.

Servomechanisms possess various inherent factors which, together, have the general effect of reducing the amplitude of each successive oscillation; such factors include static friction, kinetic friction, eddy current, lubricant viscosity, etc., and while contributing to damping requirements they do have certain detrimental effects, e.g. power is wasted, and errors can be introduced with the servomechanism operating in the steady state. The effects are partly due to a small force of constant magnitude known as *coulomb friction*, and to *viscous friction* which increases with speed.

Coulomb friction relates particularly to the response to a step input and this is illustrated in fig. 2.8 *a*. It has the tendency to downgrade the sensitivity of a servomechanism, since the torque required to overcome the friction must be generated before any movement of the load takes place. To provide this torque the load error must reach some finite size and any errors less than this will not be corrected. The load comes to rest somewhere within a band of error (the dead space), the width of the band depending on the amount of coulomb friction. The friction is, however, very small in most current types of servomechanism so that its effect can be neglected.

Viscous friction (fig. 2.8 *b*) produces a similar dead space effect, but as the friction varies with speed, the effect is associated with a ramp input. In the steady state the load moves with constant speed, and is therefore resisted by viscous friction. An error signal must be produced to overcome this and so a steady state error must exist, the error being known as *velocity*
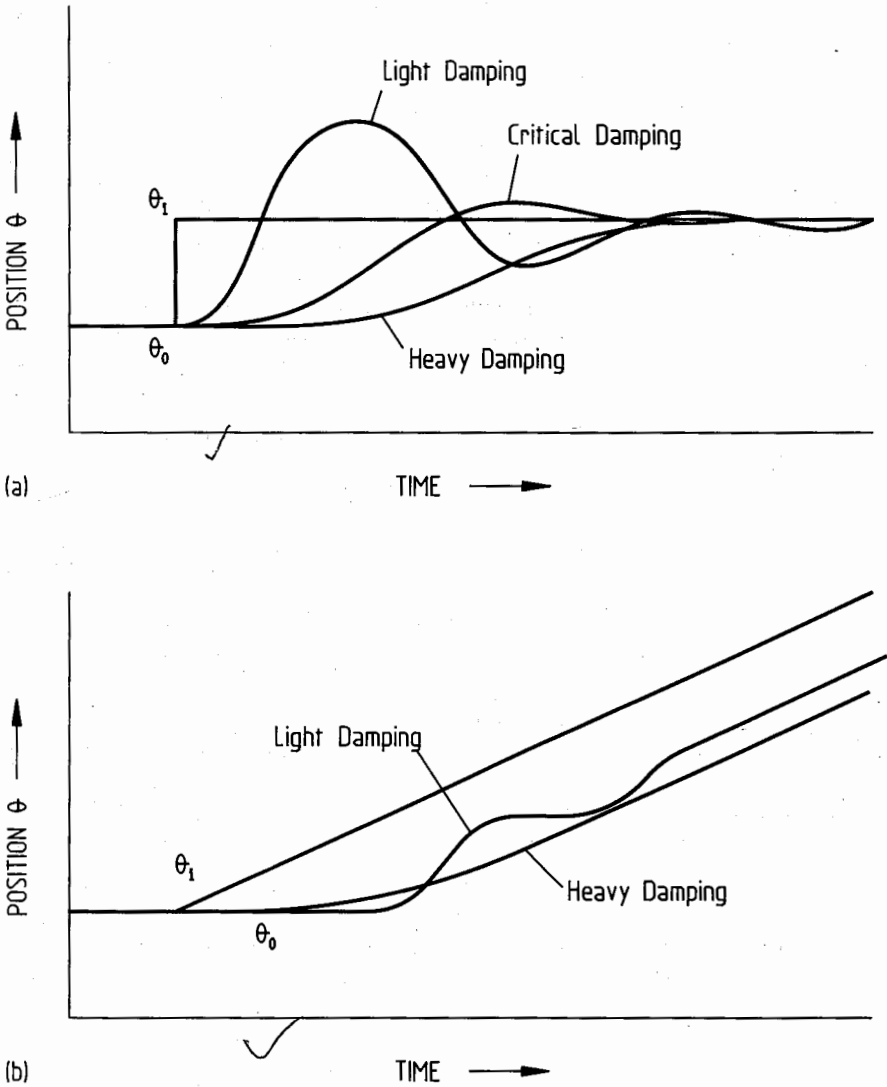
Fig. 2.9 Degrees of damping: (a) step input (b) ramp input.

lag. Coulomb friction also contributes to velocity lag, but is considered small in comparison with viscous friction.

The transient responses just described are generally adequate in applications requiring the use of small position servomechanisms, but when large loads are involved it is desirable to further reduce the number of oscillations, and the response time. Two methods commonly employed are *viscous damping* and *velocity feedback* damping.

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Viscous damping

A device commonly used for viscous damping is one in which a disc is free to rotate between the pole faces of an electromagnet. The disc is coupled to the servomechanism output shaft so that, as it rotates, eddy currents are induced in the disc. The eddy currents are of a magnitude proportional to the field strength, and to the disc velocity, and they establish magnetic fields and forces which oppose rotation of the disc and output shaft. The damping effect is produced by absorption of the servomotor torque in the desired proportion.

The response achieved by additional viscous damping can be made adequate for the particular servomechanism function, but since it absorbs servomotor torque it has the disadvantage of wasting energy; furthermore, where a ramp input is concerned the velocity lag is increased.

Velocity feedback damping

Velocity feedback damping overcomes the wasting of energy by feeding back a voltage from a tachogenerator (see fig. 2.7), the voltage being proportional to the load velocity and in opposition to the error signal applied to the amplifier unit of the servomechanism. Thus, the net input to the amplifier is the error signal voltage (i.e. the difference between input and output voltages) minus the velocity feedback voltage and since the overall effect will result in a lowering of the amplifier output, the servomotor torque will also be lowered so that less energy will be expended. Velocity feedback also increases velocity lag in response to ramp input, but for a different physical reason. In this case, the steady state velocity of the load imposes a signal at the amplifier input which must be cancelled in some way if the steady state velocity is to be maintained. The cancellation can only be made by an equal error signal, which means that an error must exist. By suitably adjusting the feedback voltage it can be arranged that the error signal is reduced to zero and then reversed before the load reaches its new position. In this manner the momentum of the load acting against the reversed servomotor torque will bring the load to rest just as it reaches its new position, thereby reducing overshoots and subsequent instability. The principle of this form of damping is shown in fig. 2.10.

Error-rate damping

As already noted, in velocity control servomechanisms (ramp input) employing velocity feedback damping the transient response is improved but velocity lag is increased. This can be tolerated in certain applications, but where requirements for rotating a load at constant speed are to be met the lag must be reduced to zero in the steady condition. This may be achieved by adopting either of two methods which in each case produce the same result, i.e. cancelling the velocity feedback signal when the input and output velocities are equal. One method (see fig. 2.11 *a*) is to fit a second tachogenerator at the input so that it feeds a signal forward into the

(a) Input θ_i

Time ⟶

(b) Output θ_0

No Overshoot

(c) Error $e = \theta_i - \theta_0$

$c = a-b$

(d) Load Velocity

(e) Net Input to Amplifier

Load Accelerating

$e = c-d$

Load Decelerating
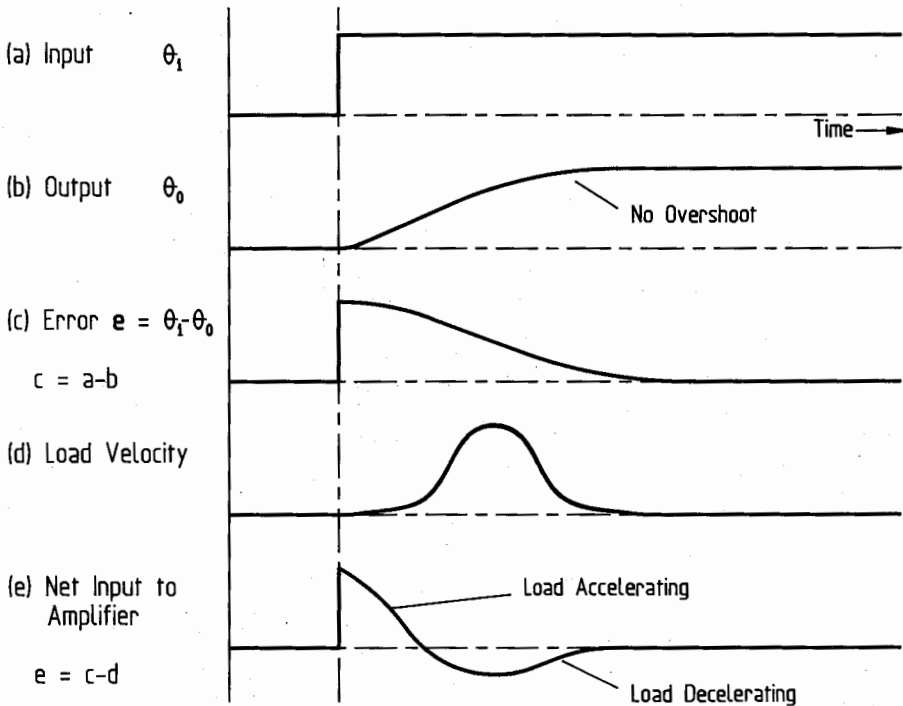
Fig. 2.10 Principle of velocity feedback damping.

amplifier, thus making the net input an error voltage plus a voltage pro-
portional to input shaft speed minus the velocity feedback voltage. During a
ramp input a steady state is eventually reached in which the tachogenerators
apply equal and opposite voltages to the amplifier; the net input is therefore
zero. If any velocity lag exists at this stage, the position error signal will
establish torques at the servomotor to reduce it.

The foregoing method, although eliminating velocity lag, presents the
difficulty of ensuring that the voltage outputs of both tachogenerators will
remain constant over a long period of time. Since the velocity of an error is
equal to its rate of change, with respect to time, i.e. the differential of the
error, then by combining a differential signal with the actual error signal at
the amplifier input, the same final result will be obtained as when using two
tachogenerators. In the second method, therefore, the tachogenerators are
dispensed with and are replaced by a resistance-capacitance differentiating
network as shown in fig. 2.11 b.

Transient velocity damping
This type of damping also known as acceleration feedback, utilises a dif-
ferentiating network connected in the velocity feedback signal line as shown

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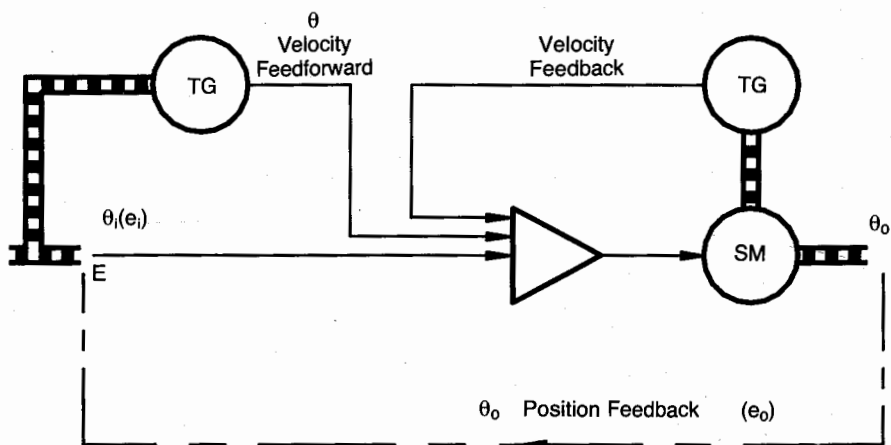
Error Voltage $E = (e_i) - (e_o)$

SM   Servomotor
TG   Tachogenerator

    Mechanical Drive

(a)
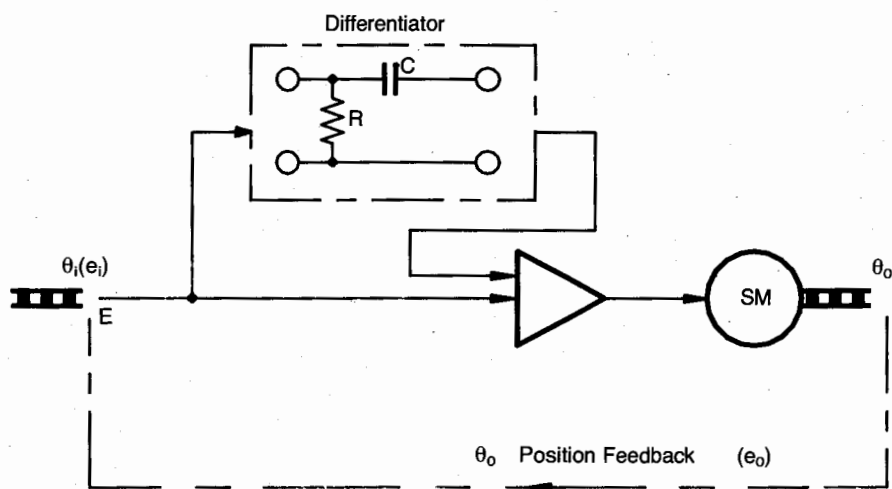
(b)

Fig. 2.11 Error-rate damping: (a) feedforward of input velocity (b) feedforward of error rate.

Fig. 2.12 Transient velocity damping.

in fig. 2.12. Thus, only the derivative of the load velocity reaches the amplifier, with the result that damping is effective only during the transient response period, i.e. when a rate of change of load velocity exists. Once the steady state is reached there is no further rate of change, the derivative is zero, and the feedback ceases thereby reducing velocity lag.

Phase advance damping
A suitable transient response in a remote position control system and a good steady state response in a velocity system can be obtained by inserting a resistance-capacitance network in the input to the amplifier, as shown in fig. 2.13. With this arrangement, the output signal is θ degrees in advance of the input signal.

When a position control system is subjected to a step function input, the error rises immediately to its maximum value because of the inertia of the system. Initially, therefore, since the capacitor 'C' cannot charge instantaneously (due to its time constant) the full error voltage is developed across R_2 and is applied to the amplifier, causing the motor to accelerate rapidly. As the capacitor is charged the voltage across it rises and the input to the amplifier falls, thus reducing the motor torque. As the load reaches the required position, the error voltage falls. However, if the values of the
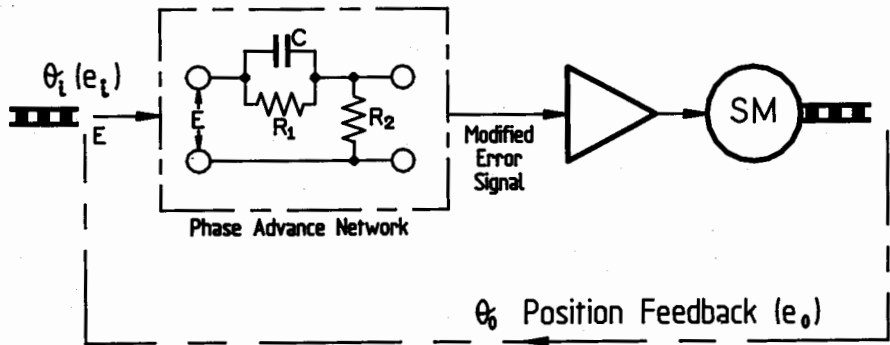
$$\text{Error Voltage } E = (e_i)-(e_o)$$

SM — Servomotor
■■■■ Mechanical Drive

Fig. 2.13 Phase advance damping.

components of the phase advance network have been carefully chosen, the charge acquired by the capacitor during the initial period will cause the voltage across it to exceed the error voltage. Thus the voltage applied to the amplifier is now negative despite the slightly positive error voltage. This means that a retarding torque is applied to the load before it reaches its required position; overshooting is therefore prevented and stability during the transient period is improved.

For a ramp function input the phase advance network gives an almost zero error in the steady state, i.e. it virtually eliminates velocity lag. In the steady state there is neither acceleration nor deceleration and zero torque is required. For this condition to be satisfied, the input to the amplifier must be zero, i.e. zero error.

Integral control
The methods so far described reduce velocity lag, but have no effect on lag and 'dead space' caused by inherent friction. A commonly used method of dealing with these residual steady state errors is known as integral control. The arrangement as used in conjunction with feedforward of error rate is shown in fig. 2.14.

The differentiator operates in the same manner as that used for error rate and transient velocity damping, but the conditions are modified by the inclusion of an integrator which feeds the time integral of the error signal into the amplifier. The effect on the transient response is negligible, but as the error settles to its steady state so its integral increases, superimposing on the amplifier a signal which provides additional torque at the load. The load is moved by this torque towards the correct position.
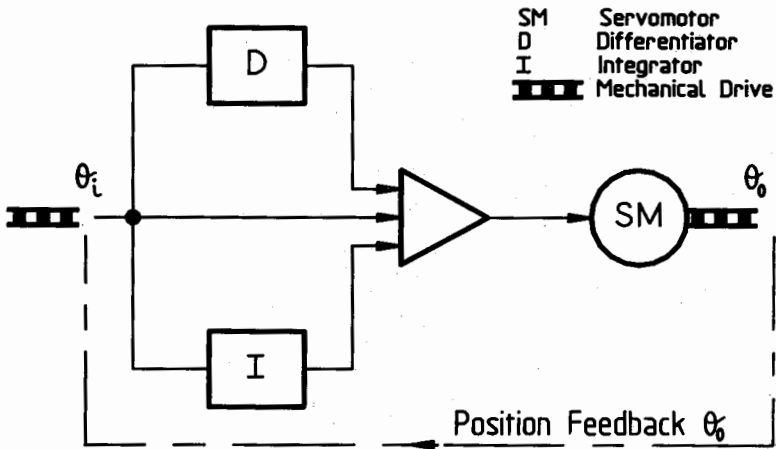
Fig. 2.14 Integral control.

Adjustment of the proportion of the integrator output can be made to ensure, when the error signal is zero, that the subsequent constant integrator output is just sufficient to counter the inherent friction. Thus, velocity lag is zero. For a step input the dead space error signal is integrated until large enough to zero the error, and adjustment of the damping differentiator output ensures stability.

Fundamentals of automatic control

Having briefly studied some of the fundamental operating principles of servomechanisms, we can now see how the closed-loop servo technique can be applied as a means of achieving automatic flight control of an aircraft. Figure 2.15 is a functional diagram of a closed-loop system which is basic to all classes of automatic flight control systems, and from this we note that there are four principal elements which together are allocated the task of coping with what is generally termed 'inner loop stabilisation'. The individual functions of the elements are as follows:

(i)   Sensing of attitude changes of the aircraft about its principal axes by means of stable reference devices; e.g. gyroscopes and/or accelerometers;

(ii) Sensing of attitude changes in terms of error signals and the transmission of such signals;

(iii) Processing of error signals and their conversion into a form suitable for operation of the servomotors forming the output stage;

(iv) Conversion of processed signals into movement of the aircraft flight control surfaces.
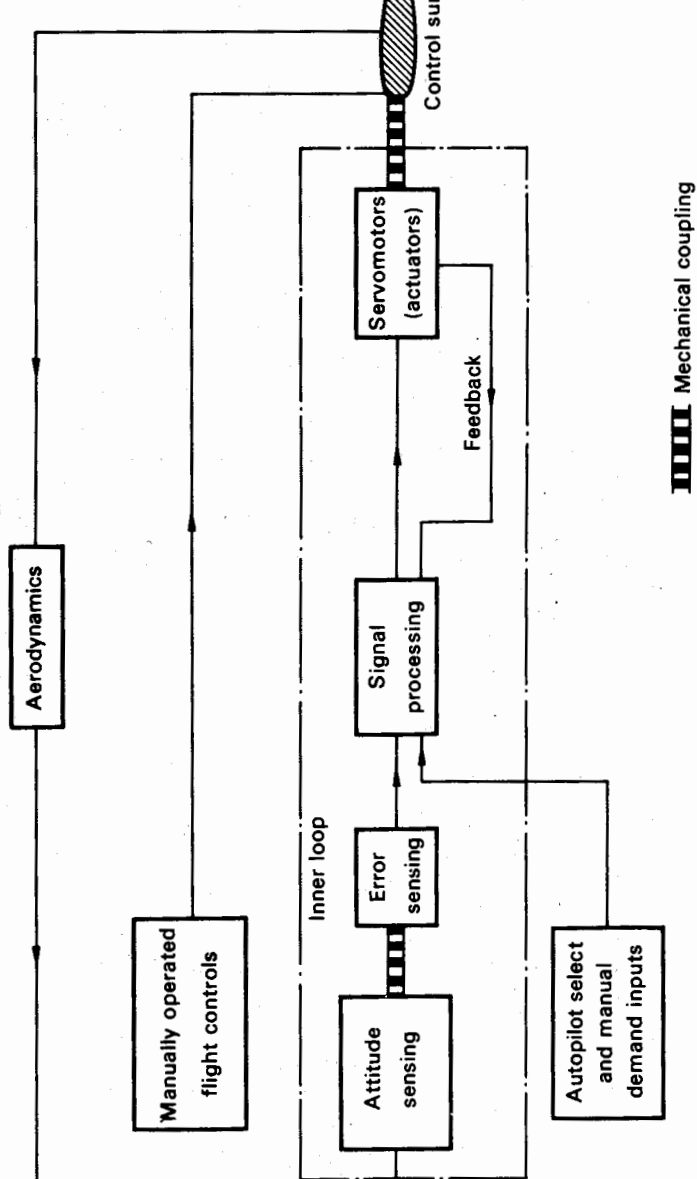
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Inner loop stabilisation.

The number of control loops, or channels, comprising an automatic control system is dependent on the number of axes about which control is to be effected.

Note: Although certain of the automatic control fundamentals and applications of the above elements are appropriate to both fixed-wing aircraft and helicopter systems, it has nonetheless been considered convenient for purpose of explanation, to treat them separately. Thus, from this point on and through chapters 3 to 7, descriptions relate to fixed-wing aircraft systems, while helicopter systems and comparable features form the whole of chapter 8.

Classifications of systems

Based on the number of axes about which control is effected, it is usual to classify systems in the following manner:

Single-axis in which attitude control is normally about the roll axis only; the control surfaces forming part of the one and only control loop are, therefore, the ailerons. Such a control system is the most basic in concept, and it is used in a number of types of small fixed-wing aircraft for lateral stabilisation, or wing-levelling as it is frequently termed. The pilot can inject command signals into the control loop thereby enabling him to turn the aircraft automatically. In some cases, signals from a compass system and from radio navigation equipment are also injected into the loop so that magnetic headings, and tracking capability can be automatically maintained; such operating modes are known as heading-hold and radio-coupling respectively, and form part of the outer loop control described in chapter 6.

Two-axis in which attitude control is, in most cases, about the roll and pitch axes; the control surfaces forming part of the two loops are, therefore, the ailerons and elevators. Manual turn control, heading-hold and radio-coupling facilities are normally standard features in any one design with, in some cases, an additional facility for selecting and holding a specific altitude.

Three-axis in which attitude control about all three axes is carried out by specifically related control channels of an automatic flight control system.

Control

In its basic mode of operation, the function of an AFCS is to hold an aircraft on a desired flight path, by detecting and correcting any departure from that path; in other words, it functions as a stability augmentation system (SAS). When a system is initially engaged via appropriate interlock

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circuits, clutch mechanisms are operated to provide mechanical coupling between the appropriate servomotors and the aircraft's primary flight control system, and the manually trimmed attitude existing at the moment of engagement is maintained.

Note: Many types of aircraft use hydraulically-powered flight control systems. In such cases, therefore, separate servo-actuators are not required since during its engagement, an AFCS controls the flow of hydraulic fluid in power control units (PCUs). Having noted this, however, several current aircraft types do use AFCS servo-actuators which directly control the PCUs (see page 209).

If the attitude of the aircraft changes as a result of, say, an air disturbance or an out-of-trim condition of the aircraft, the attitude sensing elements will detect this change and their associated command signal sensor elements will translate the change into an error signal. The signal is fed to the relevant control channel amplifier in the signal processing element and after amplification it is supplied to the servo-actuator so that it can apply corrective control. For example, if the error signal is caused by a change about the pitch axis, nose down as shown in fig. 2.16, the signal is processed by the pitch control channel and fed to the pitch servo-actuator which repositions the elevators to correct the attitude change. The servo-actuator also repositions a feedback or follow-up element (see also page 158 and 222) the purpose of which is to reduce an input error signal and thereby limit the control applied. Thus, the feedback element produces a signal equal and opposite to that of the input error, thereby limiting servo-actuator operation until it stops with the elevators in the angular position required to return the aircraft to the level flight attitude. As the aircraft returns, the error signal decreases and the feedback signal, via the amplifier, now causes the servomotor to reduce the angular position of the elevators towards neutral. A similar sequence takes place when an error signal is caused by a change about either the roll or yaw axes.

In addition to the foregoing role of inner loop stabilisation, an AFCS must have the capability of assisting the pilot in manoeuvring the aircraft so that its attitude can be changed in order to comply with required in-flight procedures. Such changes would, for example, be a turn on to a new heading, or an altitude change. Any attempt, however, by the pilot to make these changes by applying forces to the control wheel or rudder pedals in the conventional manner would (with the exception of a control wheel steering mode, see page 199) be resisted by the automatic control system since in performing its primary stabilising function it would interpret the changes as disturbances. It is, therefore, necessary to provide a flight control panel through which the pilot can inject command signals into the appropriate control channels and thereby initiate servo-control. The flight control panel may thus be considered as the primary element in the outer
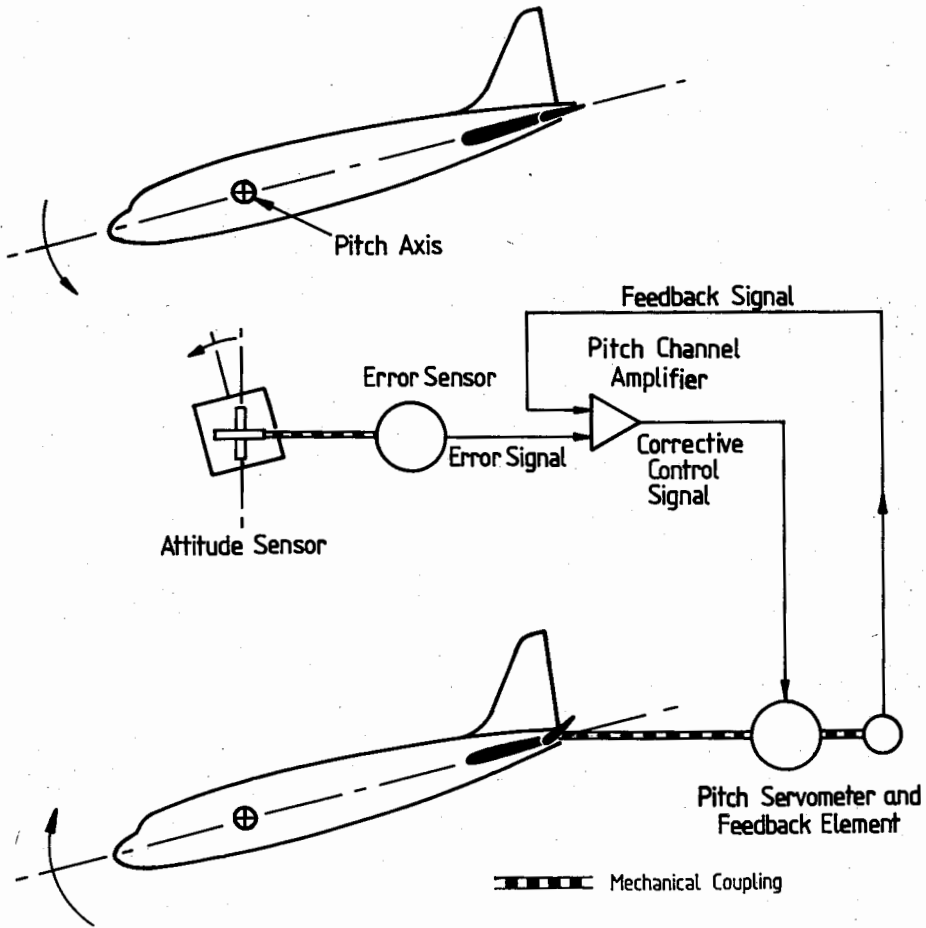
Fig. 2.16 Sensing and correction of an attitude change.

control loop (see chapter 6) and the control facilities it provides depend on whether the control system as a whole is a simple single-axis 'wing leveller' or whether it forms part of a more complex flight guidance system. In the basic form, however, control panels have two main control facilities and these are (i) a turn control, and (ii) a pitch control. There are exceptions to this, and they apply to systems adopting the control wheel steering mode (see page 199).

The turn control consists of a control knob which can be displaced to the left or to the right of a neutral position commonly referred to as the 'centre detent' position. When the knob is in this position, the control circuit provides an interlock for pre-selected heading data signals from a compass system. In the displaced position, the interlock circuit is interrupted, and

since the control knob is connected to either a potentiometer or a synchro transmitter rotor, depending on the type of control system, a command signal circuit is completed to the roll control channel of the control system and thence to the aileron servo-actuator. Thus, the aircraft is rolled into a turn at an angle proportional to the amount of displacement of the control knob. The resulting feedback signal opposes the command signal until both signals are equal. In order to maintain the commanded roll angle and turn, the aileron control must be progressively 'taken off'. This is accomplished by the appropriate attitude sensor which, in sensing the roll command as a disturbance, will cause its associated command signal sensor to transmit a signal which is ultimately applied to the aileron servo-actuator causing it to return the ailerons to their neutral positions. The aircraft, therefore, is flown at the required roll angle until the turn control knob is returned to the detent position. When this is done, the roll command signals and the attitude sensor signals will be in the opposite sense, the ailerons being returned to neutral as the aircraft returns to the level flight attitude. Since the heading data system continuously operates in a follow-up mode, the return of the control knob to detent also permits the signals appropriate to the new heading to be applied to the roll control channel via the re-engaged interlock circuit.

The angle of roll must be limited to angles compatible with the aircraft's control characteristics, and this is effected by passing the command signals through a limiter network. The limiter output is also supplied through a cross-feed circuit to the rudder control channel so that the rudder is displaced by an amount sufficient to provide co-ordination of the turn (see page 121).

The pitch control comprises a control wheel which can be rotated in a vertical plane, and in the same sense required to establish a nose-down or nose-up displacement from a neutral position. The control wheel is also connected to either a potentiometer, or a synchro transmitter rotor, but in this case, the appropriate command signals are applied to the pitch control channel and elevator servomotor of the control system. Thus, the method of applying control, and of returning the elevators to their neutral position, is similar in operation to that applied to turn control. In systems which incorporate altitude-hold and glide path-coupling facilities (see chapter 6) the pitch control also includes an interlock circuit which disconnects these facilities whenever a pitch attitude change is selected, thereby preventing any opposition to the command signals.

In addition to turn and pitch control facilities, a control panel also incorporates a switch for engaging the servomotors to the aircraft's primary flight control system; and examples are shown in fig. 2.17.

In the majority of cases, and as already noted, it is the practice to integrate an AFCS with a flight director system (FDS) to provide more precise automatic flight guidance. As the operating mode requirements for attitude, radio navigation and certain manometric data, e.g. airspeed and
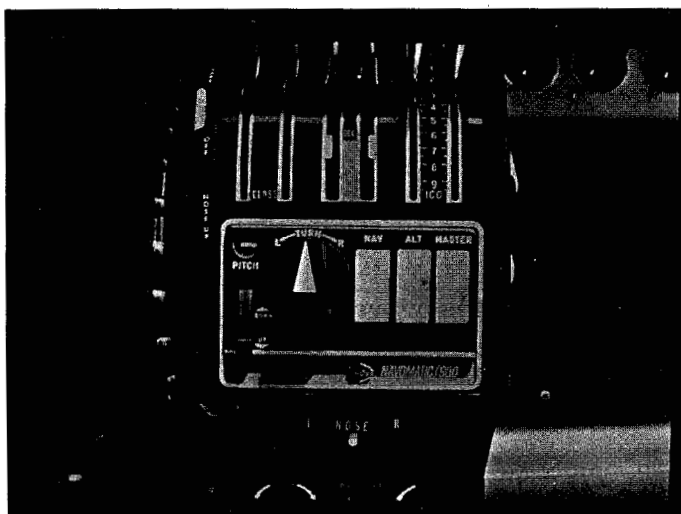
Control panel installed in a small twin-engined aircraft
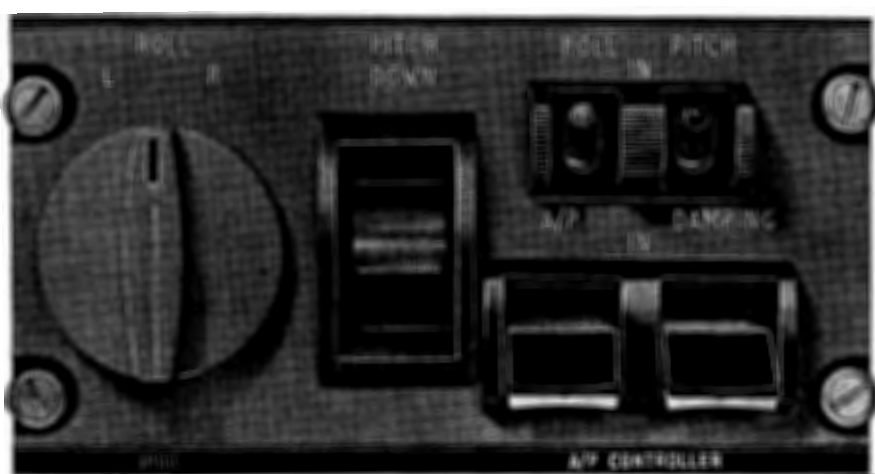
Fig. 2.17 Control panels.

altitude, are common to both systems, then as may be seen from the example shown in fig. 2.18, the relevant control panels can be grouped together. In this example, the left-hand and right-hand FDS indicators respectively are coupled to the AFCS by operating the upper and lower coupling switches on the control panel.

There are, of course, many variations in the 'make-up' of panels depending on the operational requirements to be met by specific aircraft/
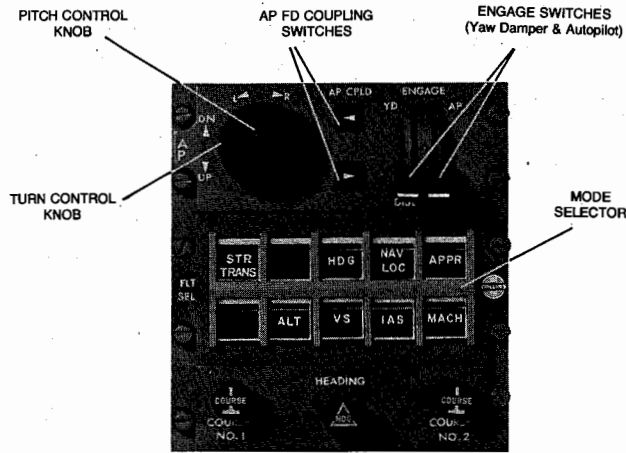
Fig. 2.18 Control and mode selector panels (autopilot/flight director system).

AFCS/FDS combinations. The panel shown in fig. 2.19 is one applied to the Boeing 767 combination and contains all the switches and logic circuits for mode selection and control of both automatic control system and FDS channels. In addition to this panel, a separately located flight controller panel is provided and contains the pitch control wheel and turn control knob.

In some aircraft, the AFC and FD systems, although utilising common data sources, are in fact controlled from separate control panels. The Boeing 737 is one example and the configurations of the panels used are illustrated in fig. 2.20. A further point of interest to note is that the automatic control system panel does not include pitch and turn controls, the reason for this being that in this type of aircraft the concept of control wheel steering (see page 199) is adopted.

Interlocks

Before an automatic control system can be engaged with an aircraft's flight controls, certain preliminary operating requirements must be fulfilled to ensure that the system is in a condition whereby it may safely take control of the aircraft. The principal requirements are that the connections between system power supplies, the elements comprising the system, and the appropriate signal and engage circuits are electrically complete. It is the practice, therefore, to incorporate within any automatic control system a series of switches and/or relays, known as interlocks, which operate in a specific sequence to ensure satisfactory engagement, and the coupling of input signals from outer loop control elements (see chapter 6).
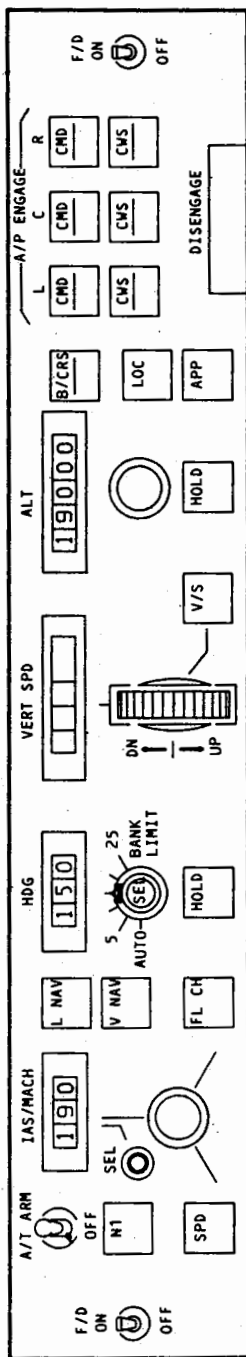
Fig. 2.19 Mode control panel.
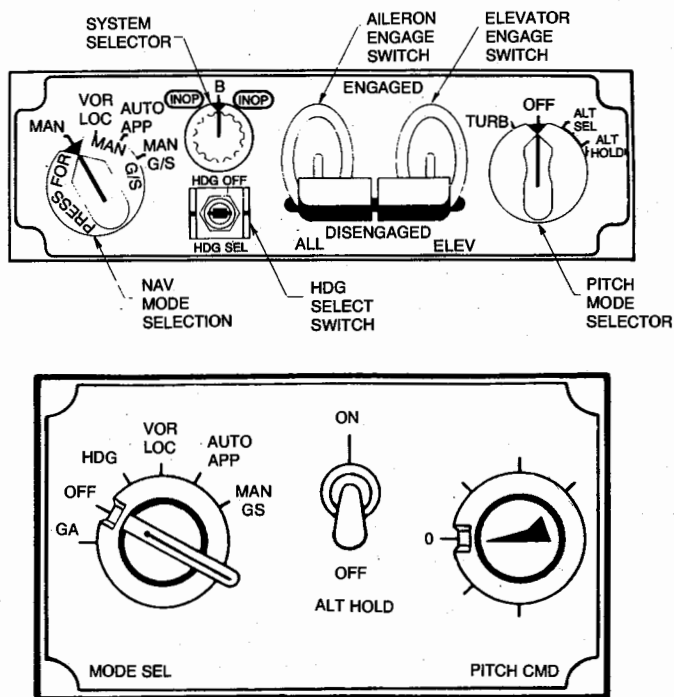
Fig. 2.20 Autopilot/flight director system control.

The number of interlocks incorporated in any one system varies considerably according to the control capability of that system (see page 95) and space does not permit detailed description of the various functions and circuit combinations. However, the fundamentals of interlock operation in general, may be understood from fig. 2.21 which is a much simplified presentation of an engagement circuit based on a system utilising electric motor-driven servo-actuators. It will be noted that the contacts of switches and relays, in the main, form a series-connected circuit, i.e. each must be closed in order to complete the power supply circuits to the servomotor clutches via the engage switch.

When the aircraft's power supply is on, it is automatically supplied to the attitude sensing elements and signal processing elements of the control system, so that they are in a stand-by condition when the engage switch is in the 'off' position. Certain of the relays in the engage circuit are also automatically energised when the power supplies are on, namely, the d.c. and a.c. power monitoring relays and the Mach trim coupler relays. The vertical-axis gyroscope relay also energises automatically but only after the gyroscope has run up to operating speed, and the fast erection cycle has been completed. In order to complete the circuit to the engage switch, the

Vertical gyro

Turn control switch

Auto trim cut-out switch

Mode selector switch

Man.

Rudder servo clutch

Aileron servo clutch

Engage relay

Elevator servo clutch

Mach trim coupler relays

Power monitoring relays

a.c.

d.c.

Engage switch

Damper

Off

Autopilot

Disengage button switches

Co-pilot's control wheel

Captain's control wheel

Engage interlock relay
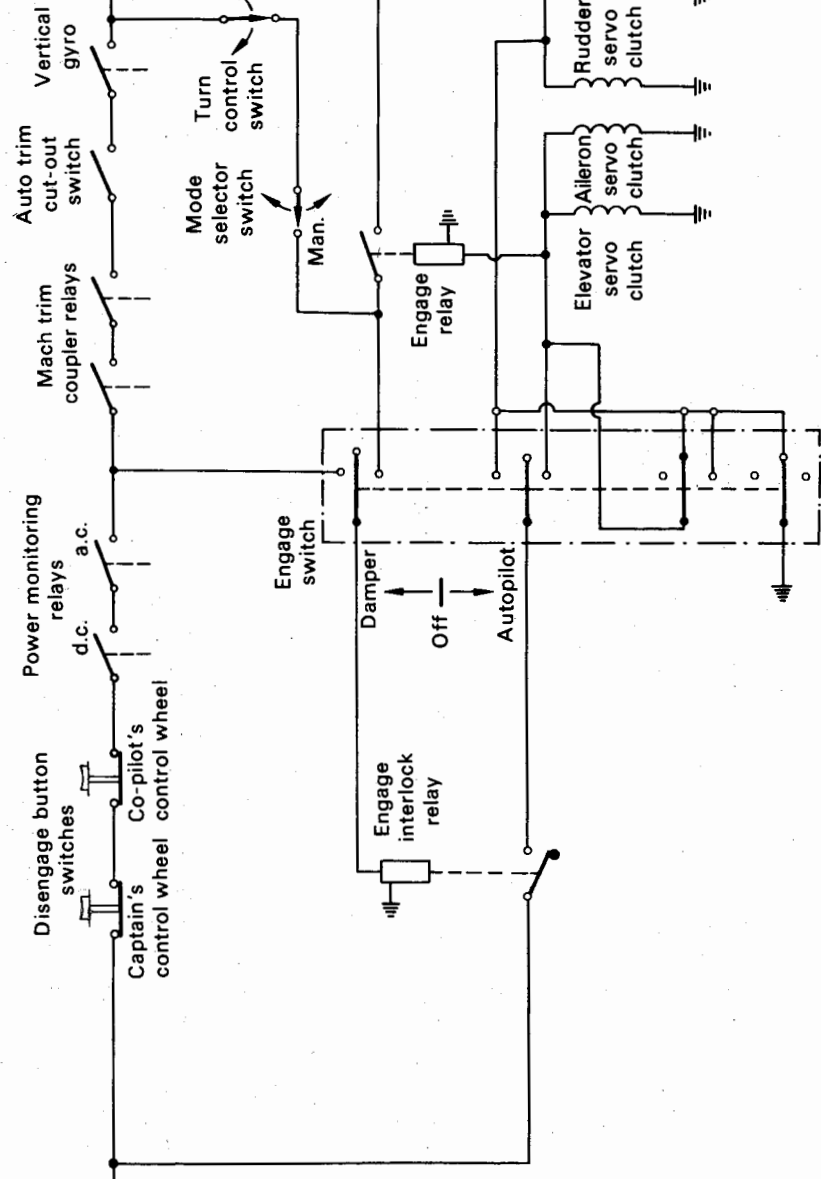
terlocks.

turn control knob must be at its centre detent position, the mode selector knob at the 'manual' position, and as in the example shown, the automatic trim cut-out switch must also be on.

When the engage switch is placed in the 'Autopilot' position, direct current is supplied from the mode selector switch to the engage interlock relay, which on being energised completes the power supply circuit to the clutches of the elevator and aileron servo-actuator motors, and to the coil of an engage relay, which then completes a circuit to the engage switch, the circuit being in parallel with the turn control and mode selector switches. At the same time, of course, the rudder servo-actuator motor clutch is energised but in this particular example the circuit is completed via another set of contacts within the engage switch. This also applies to a coil circuit within the engage switch, the purpose of the coil being to hold the switch at the 'Autopilot' engaged position.

As noted earlier (page 97), when it is necessary to apply a turn command to the flight control system, the turn control knob is positioned to the left or right of its detent, and this action corresponds to an 'open interlock' condition. However, as reference to fig. 2.21 will show, this will not cause disengagement of the servo-actuator motor clutches, because the engage interlock relay coil then remains energised via the d.c. supply circuit in parallel with the turn control switch. The circuit functions in a similar manner whenever the mode selector switch is positioned out of the 'manual' position in order to apply appropriate outer loop control commands to the flight control system. The circuits relevant to the outer loop elements also incorporate interlocks, and their functions are described in the appropriate sections of this book.

The 'damper' position of the engage switch permits the selection of a yaw damping facility (page 222) which in the example shown in fig. 2.21, is provided by the rudder servomotor. When selected, the circuits through the turn control and mode selector, together with the aileron and elevator servo-actuator motor clutch circuits, are isolated from the engage switch, and the d.c. supply for the rudder servo-actuator motor clutch, and engage switch hold-in coil, is fed via the closed contacts of the engage interlock relay.

Disengagement of the automatic control system is normally effected by pressing one or other of the button switches on the control wheels. This action de-energises the engage relay and the engage interlock relay, and also interrupts the engage switch hold-in coil circuit thereby releasing the switch to the 'off' position. Similarly, disengagement will take place whenever any one of the other interlock switches or relays is opened.

Trimming and synchronisation

In addition to the pre-engage requirements that automatic control system circuits are electrically complete, it must also be ensured that on engage-

ment the 'take-over' is effected smoothly and without 'snatching' of the aircraft's control system. In other words, the aircraft must be trimmed for the desired flight attitude before engagement, and the automatic control system must be synchronised to maintain that attitude on engagement.

When power is applied to the automatic control system, the attitude sensing elements are in operation so that they will always detect the aircraft's attitude, and therefore, supply any necessary control command signals to the servomotors. At the same time, any signals will be supplied to the appropriate channels of a trim indicating system, or out-of-trim light system (page 228). For example if, before control system engagement, the aircraft is in a climb, or has been trimmed to fly in a nose-up attitude, the pitch attitude sensing element will detect this, and will supply a signal to the elevator servo-actuator commanding it to move 'elevator down', such as would be shown on the trim indicator. Because the signal in this case is a standing one, assuming for the moment that it has no opposition, the servo-actuator will continue to operate, and if at any one moment it was engaged with the elevator control system, the elevators would be snatched from their trimmed position and so cause a nose-down attitude change.

The aerodynamic load acting on the elevators would be felt by the servo-actuator, thereby helping to retard its rotation. As soon as the sensing element of the pitch attitude detector responds to the attitude change, the opposing signal produced would then eventually stop the actuator and cause it to apply control in the opposite direction. Thus, control would be of an oscillatory nature and the aircraft would take up the pitch attitude determined by the attitude detector and not that which it was desired the control system should maintain, i.e. in the example considered a climb or a nose-up trim condition. It is, therefore, necessary to oppose the standing signal and reduce it to zero before engaging the control system, so as to stop the servo-actuator in a position which is synchronised with the datum attitude detected by the sensing element, such position being indicated by the return of the trim indicator pointer to its central position.

The manner in which synchronising is effected depends on the type of AFCS and the signal processing circuit arrangements adopted; some representative examples are described in chapter 5.

TEST QUESTIONS

1 What are the main requirements of a closed-loop control system?
2 Describe the operating principle of a typical servomechanism.
3 What is meant by the term 'response' of a servomechanism?
4 What are the main differences between a step input and a ramp input?
5 State the effects which coulomb friction and viscous friction have on servomechanism response, and the methods adopted to minimise them.
6 What is the difference between critical damping and optimum damping?

7 Describe one method adopted for reducing velocity lag.

8 What method is adopted to reduce the steady state error referred to as 'dead space'?

9 In applying the closed-loop servomechanism principle to an automatic flight control system, what main elements are required?

10 Briefly describe the operation of an automatic flight control system when correcting a disturbance about an aircraft's pitch axis.

11 Which interlock circuits are associated with the turn and pitch controls of a pilot's control panel, and when are they interrupted?

12 Why is it necessary for interlocks to be incorporated within a control system?

13 What would be the effect if a control system were engaged with 'standing' attitude command signals present, and how is it overcome?

3

Sensing of Attitude Changes

Under automatically controlled flight conditions, the sensing of all changes in aircraft attitude is accomplished by referencing them against some form of stabilised device comprising the primary element of inner loop stabilisation. The device most commonly adopted for this purpose is the gyroscope which, in some applications, is 'backed up' by a pendulous device that senses short-term attitude changes brought about by the effects of accelerations, vertical speed changes, and sideslip.

In the majority of current automatic control systems, gyroscopic and pendulous devices are contained in individual units that form part of a system. However, in aircraft utilising an inertial reference system for purposes of navigation (see page 190), attitude reference data for automatic flight control is derived from sensors on the stabilised inertial platform, and so individual gyroscopic or pendulous units for the AFCS itself are not required.

The gyroscope and its properties

A gyroscope is basically a mechanical device, the essential element of which is a heavy metal wheel, or rotor, spinning at high angular velocity about a spin axis. In order to provide spinning freedom, the rotor shaft is pivoted in a ring which is, in turn, pivoted in a second ring; the rings are known as the inner and outer gimbal rings respectively. The whole assembly comprises what is known as the gimbal system of a *free or space gyroscope* in that the spin axis, XX, remains fixed in space. The gimbal system is pivot-mounted in a frame as shown in fig. 3.1. The foregoing pivoting arrangements of the gimbal rings permit the gyroscope to have two degrees of freedom: (i) tilting freedom of the inner ring and rotor about the horizontal axis YY_1, and (ii) veering freedom of the complete gimbal system about the vertical axis ZZ_1.

In operation, the gyroscope possesses two important fundamental properties: gyroscopic inertia or rigidity, and precession. Both these properties depend on the principle of conservation of angular momentum, i.e. the

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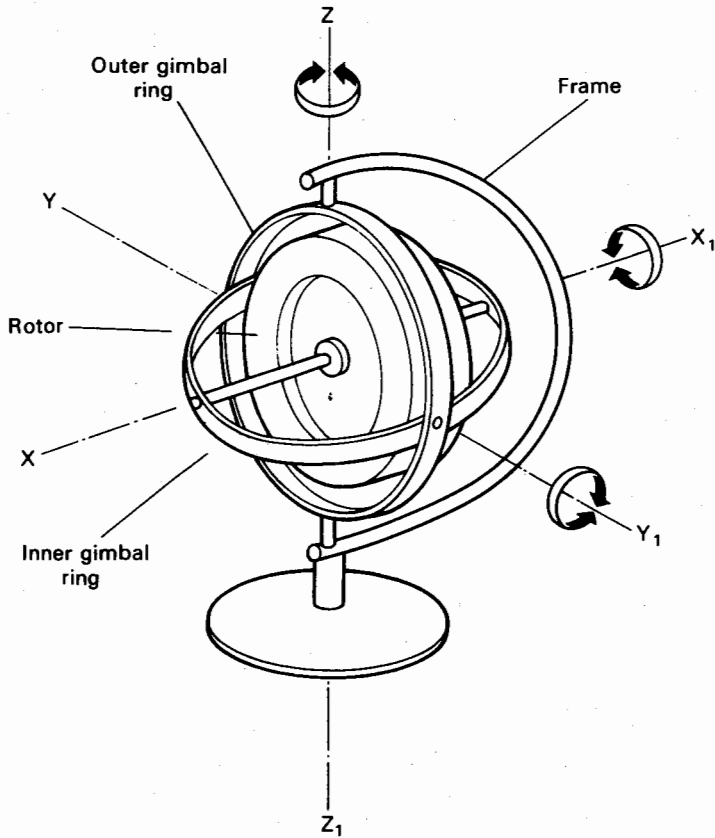
Fig. 3.1 Free gyroscope.

angular momentum of a body about a given point remains constant unless some force is applied to change it; the properties form the subjects of the first and second laws of gyrodynamics respectively.

Rigidity

This property resists any force which tends to change the plane of rotation of the rotor of a gyroscope, and is dependent on three factors: (i) the mass of the rotor, (ii) the angular momentum of the rotor, and (iii) the distance at which the mass acts from the axis of rotation, i.e. the radius of gyration.

Precession

Precession is the angular rate of change in the direction of the plane of rotation under the influence of an applied force. The rate of change is
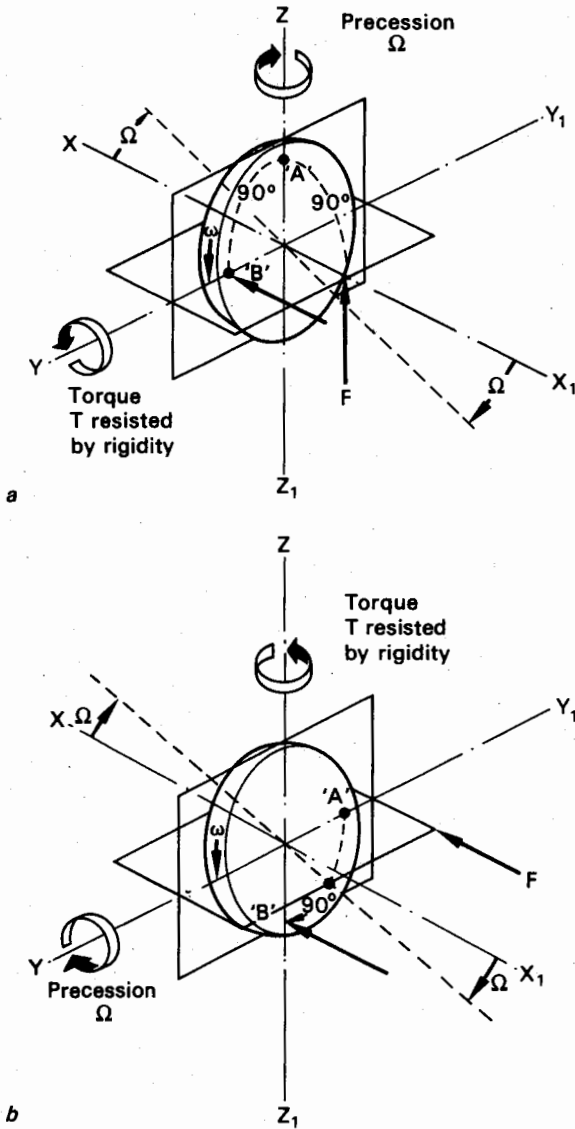
Fig. 3.2 Gyroscopic precession.

proportional to the strength of the applied force, and inversely proportional to the moment of inertia of the rotor and the angular momentum of the rotor.

Figure 3.2 *a* illustrates a free gyroscope, the rotor of which is assumed to be spinning, with constant angular momentum ω in an anti-clockwise

direction. If a force F is applied upward at the inner ring then, as with any body which may be moved about some axis, the force will produce a torque T (equal to the produce of the force and the distance at which it acts from the centre of the rotor) about the axis YY_1. Since a gyroscope possesses rigidity, any tendency for the torque to produce tilting of the inner ring and rotor about axis YY_1, is resisted. The complete gryroscope will, however, precess at a constant angular velocity about the axis ZZ_1, and in a direction at right-angles to the applied force.

A simple rule of thumb method of determining the direction in which a gyroscope will precess is to consider that the applied force acts at right-angles to the plane of spin, and at a point directly on the rotor rim (point 'A' in fig. 3.2 *a*). If the point is carried around the rotor and through 90° in the direction of rotation (point 'B') that will be the point at which the force and resulting torque are apparently acting.

The direction in which precession of a gyroscope takes place is dependent on the direction of rotation of its rotor, and on the direction of the applied force. This may be noted by comparing fig. 3.2 *a* with fig. 3.2 *b*. As soon as the force and torque are withdrawn precession ceases, but if they are continuously applied precession will continue until the plane of spin of the rotor is aligned with the plane of the applied force.

In connection with gyroscopic attitude sensing devices, it is usual to mutually associate the axes YY_1 and ZZ_1 with the spin axis and to relate them to input and output functions. Thus, the axis about which a torque is applied is termed the input axis and the one about which precession takes place is termed the output axis.

Limitations of a free gyroscope

The free or space gyroscope we have thus far considered would serve no, useful purpose as an aircraft attitude sensing device, since an aircraft in flight is still very much an 'earth-bound' vehicle. Thus, all attitude references must be with respect to the earth. It is required therefore, that a free gyroscope be corrected in some way to take into account the effects of such factors as the earth's rotation and the transport of the gyroscope from one point on the earth to another.

Effects of the earth's rotation

Since the earth rotates about its axis 360° every day its rate of rotation is 15° per hour, and in association with gyrodynamics, this is termed the earth rate (ω_e). Depending on the orientation of the spin axis and input axis (as defined earlier) of a free or space gyroscope, it will sense various components of the earth rate as an angular input, and to an observer on the earth the gyroscope would appear to veer, or drift as it is normally termed in this context.

Let us consider first a gyroscope positioned at a latitude λ ('A' in fig. 3.3) so that its spin axis is horizontal, and its input axis (ZZ_1 in this case) is
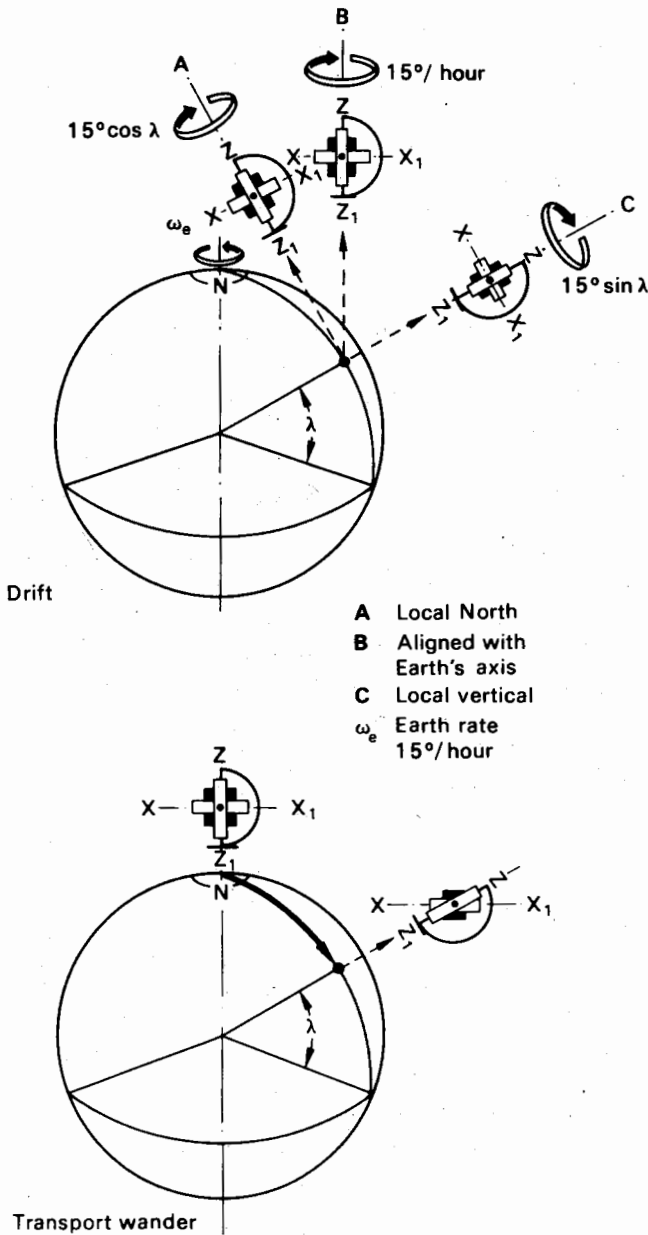
Fig. 3.3 Drift and transport wander.

aligned with the local N–S component of the earth's rate at this latitude. To an observer at the gyroscope's position, the gyroscope rotor and gimbal system would appear to drift in a horizontal plane relative to the frame; the drift rate would be equal to $15°\cos\lambda$. Drift would also be apparent at

position 'B', but since in this case the input axis is aligned with the earth's axis, the drift rate would be equal to the earth's rate, i.e. 15° per hour. When the input axis is aligned with the local vertical component of the earth's rate (position 'C'), the apparent drift would be equal to $15° \sin \lambda$. If the input axis were aligned with the local E–W component of the earth's rate, there would be no apparent drift since rotor rotation and earth rotation are both in the same plane.

Effects of transporting a gyroscope over the earth's surface

Let us again consider the effects on a horizontal axis gyroscope which is set up initially at the North Pole, with its input axis aligned with that of the earth then, as we have already observed, it would exhibit earth rate drift. Assume now that the gyroscope is transported to a lower latitude and with its input axis aligned with the local vertical component of the earth's rate. During the time of transport, it will have appeared to an observer on the earth that the spin axis of the gyroscope has tilted in a vertical plane, until at the new latitude λ it appears to be in the position shown. Apparent tilt, or transport wander as it is called, would also be observed if, during transport, the input axis were aligned with either a local N–S component or a local E–W component of the earth's rate. Since the transported gyroscope is also rotating with the earth the tilt and earth rate drift are simultaneous effects which, during one complete rotation of the earth, would make the gyroscope appear to trace out a conical path. A free gyroscope having a vertical spin axis would, on the other hand, only exhibit apparent tilt.

Earth or 'tied' gyroscope

In order to utilise a free gyroscope as a practical attitude sensing element, compensations must be made which will minimise earth rate drift and transport wander to the fullest extent and, by so doing, will maintain the axis in the required mutual positions relative to the earth. Thus, the gyroscope must be converted into what is commonly termed an earth or 'tied' gyroscope.

Various compensation methods are adopted for this purpose, and their application is governed by sensing element design. Fundamentally, however, they are based on gravity sensing and the utilisation of the property of precession.

Attitude references established by gyroscopes

Gyroscopes establish reference data against which changes in an aircraft's pitch, roll and yaw (or direction) attitudes may be sensed and for this purpose two types of earth gyroscope are adopted: the displacement gyroscope and the rate gyroscope.
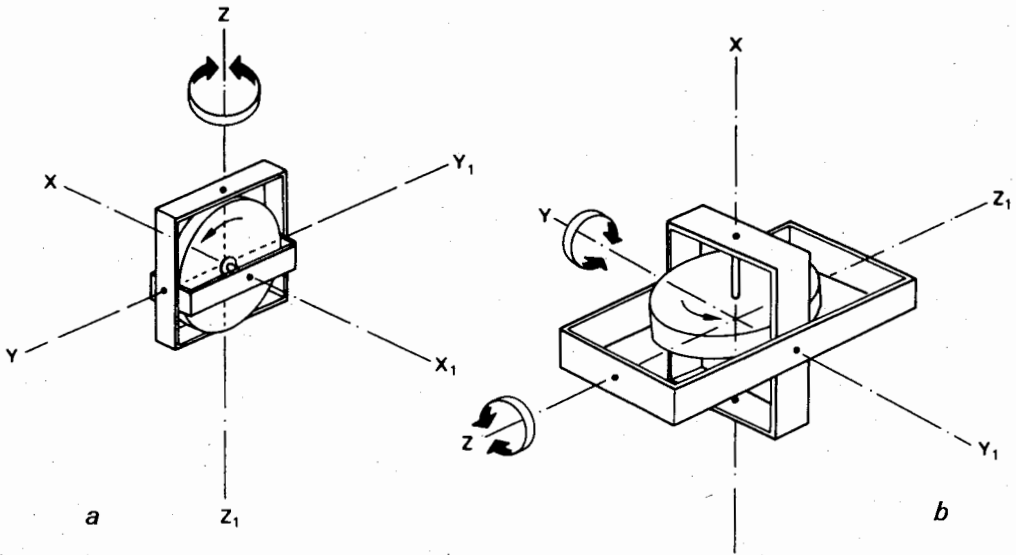
Fig. 3.4 Attitude references.
(a) Directional reference about axis ZZ_1
(b) Pitch attitude reference about axis YY_1. Roll attitude reference about ZZ_1

Displacement gyroscope

A displacement gyroscope is one which has two degrees of freedom, and can be used for detecting angular displacements about the fixed datum established by the property of rigidity. For the establishment of a directional reference, a gyroscope having its spin axis horizontal is employed (fig. 3.4 a). Changes in aircraft heading relative to the gyroscope are detected by an inductive type of sensor mounted on the axis ZZ_1.

A displacement gyroscope is also used for establishing pitch and roll attitude references, but as will be noted from fig. 3.4 b the gyroscope spin axis is, in this case, vertical. Changes in pitch-and-roll attitude relative to this gyroscope are detected by inductive type sensors mounted on the axes YY_1 and ZZ_1 respectively.

Rate gyroscope

Unlike a displacement gyroscope, a rate gyroscope (fig. 3.5) is constrained to only one degree of freedom, and it detects the angular rate at which displacements about a selected input axis take place. It utilises the property of precession for this purpose, the precession occurring in proportion to the rate at which displacements occur, equilibrium being established between precession and some form of controlled restraint (e.g. a spring). It will

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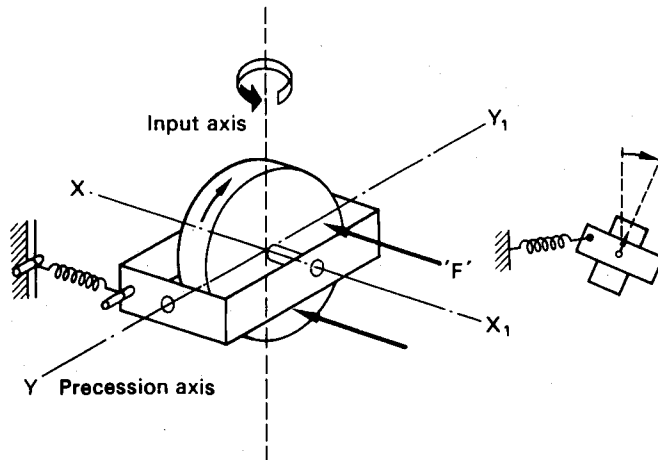
Fig. 3.5 Rate gyroscope principle.

be noted from fig. 3.5 that freedom of movement about the output or precession axis of a rate gyroscope is restrained, in this case, by a spring connected between the gimbal ring and casing. The spring, which may be of the linear or torsional type, has a characteristic which is calibrated to balance out precession produced at the appropriate rates of torque input. An alternative method of restraint adopted in some designs is one employing a force-balance, or force-feedback system, (see also page 115) whereby a torque is generated electrically to balance the precession, and by means of a torque motor the gimbal ring is restrained at a null position. The current flowing through the torque motor thus becomes a measure of the rate of torque input.

When a rate gyroscope is in its normal operating position the rotor spin axis will always be horizontal due to the spring restraint, and the moving element of the signal pick-off will be at the zero datum. With the rotor spinning, its rigidity will further ensure that the zero datum is maintained. When, however, the gryroscope is turned about a vertical input axis, the rigidity of the gyroscope resists the turning, but it tilts in one direction or the other about the precession axis YY_1 depending on the direction of the turn; the effects of a turn to the left are indicated in fig. 3.5. Precession continues until it is balanced by the restraining force created by the stretching of the spring. Since the precession of a rate gyroscope is equal to the product of angular momentum of the rotor, and the rate of turn, the spring force is then a measure of the rate of turn.

In practical applications, the gimbal ring deflection is generally limited by stops to not more that $2°$, the reason for this being to reduce the cross-coupling error due to the rate-of-turn component not being at right angles to the spin axis during gimbal ring deflection.

It should be noted that a rate gyroscope requires no erecting device or correction for random drift, for the simple reason that it is always centred at its zero datum position by the appropriate system of spring restraint. Since precession plays such an important part in the operation of rate gyroscopes, it is essential for the rotor speed to be maintained constant.

As an attitude change sensing device, a rate gyroscope has several advantages over a displacement gyroscope, the principal of which are (i) more rapid detection of small attitude changes since the time interval of the angular velocity is not involved, and (ii) smoother corrective action without 'overshoot' tendencies, because control signals are provided at a rate proportional to that at which attitude changes occur (called 'rate/rate control').

In the design of rate gyroscopes it is also necessary to incorporate a system of damping restraint. Such a system should be as small as possible to ensure that the gyroscope will instantly respond to rate-of-turn changes, and at the same time, provide sufficient restraint to damp out and oscillations. The damping methods adopted vary between manufacturers, but in general, they fall into four principal classes (i) eddy current, (ii) fluid, (iii) air dashpot, and (iv) shaped feedback networks.

Force-balance rate gyroscope

A force-balance, or force feedback, rate gyroscope operates on the same fundamental principle as its conventional counterpart, but instead of utilising spring balancing force as a measure of the rate of turn or torque input, balancing is achieved electrically by a torque motor acting on the gimbal ring. The torque motor is connected to the pick-off sensing element via a high-gain amplifier; thus, when the gyroscope is turned about its input axis, precession of the gimbal ring is detected by the pick-off element, and a signal is transmitted to the amplifier. The amplifier signal is then transmitted to the torque motor in which an electromagnetic field is generated to produce a torque equal and opposite to the input torque so that the gimbal ring is restrained at its null position. In other words, the pick-off element serves only as a null detector, and the torque motor system functions as a stiff electrical spring. Since the torque is precisely related to the current flowing through the torque motor, the current may be used directly as a rate signal output to the appropriate control channel. Damping is achieved by incorporating suitable shaping networks in the feedback amplifier, thereby eliminating the need for any of the methods adopted of conventional spring-restrained gyroscopes.

In some types of force-balance gyroscope, the pick-off element is of the optical type consisting of a light emitting diode, two photo-electric cells, and a vane attached to the gimbal ring. The diode generates infra-red radiation which is sensed by the two cells. The vane is positioned between the diode and the cells and it differentially obscures the radiation when the

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gimbal ring is displaced. The outputs from the photo-electric cells are combined and amplified and then supplied to a moving-coil type of torque motor which drives the gyroscope and pick-off element to the balance position.

Rate sensing about three axes

From the description given earlier of the operating principle of the rate gyroscope, we have observed that sensing of a turn is a direct result of precession caused by movement about a vertical input axis or, what is the same thing, movement of the gyroscope in the yawing plane. It should, therefore, be apparent that if two additonal rate gyroscopes can be positioned in an aircraft such that one can respond to movement in the pitching plane, and the other to movement in the rolling plane, then the resulting precession and spring balancing force in each case will be a measure of 'turn rates' in these two planes. We thus have the basis of a combined system for sensing the rates at which attitude changes occur in relation to all three axes of an aircraft, the disposition of the gyroscopes being arranged as shown in fig. 3.6.

Single-axis rate sensing

For certain types of small aircraft, automatic control systems are available for the purpose of providing simple wing-levelling or roll control, and yaw damping, via the ailerons only. In such systems, sensing is by means of a single rate gyroscope.

As noted earlier, the sensing attitude changes about the roll and yaw axes of an aircraft can be effected by the relative positioning of the input and output axes of two rate gyroscopes. However, since the spin axes of both gyroscopes are in the same horizontal direction (see fig. 3.6), then if we take either one of the gyroscopes and mount it so that its gimbal ring and precession axis is tilted at a preset angle (30° is a typical value) from the horizontal, it alone will sense both roll and yaw attitude changes. Examples of this sensing method are to be found in instruments known as turn co-ordinators, and as turn and bank stabiliser/trackers, the latter forming an optional feature in certain of the Edo-Aire Mitchell 'Century' series of control system for the provision of turn rate commands, and VOR radial and ILS localiser beam tracking.

Combined use of displacement and rate gyroscopes

A number of automatic flight control systems currently in service make use of error information, based not only on the magnitude of displacements of

YAW AXIS

Yaw gyro

ROLL AXIS

Pitch Gyro

PITCH AXIS

Roll gyro

Precession

X ——— ·—· X₁ Spin axes
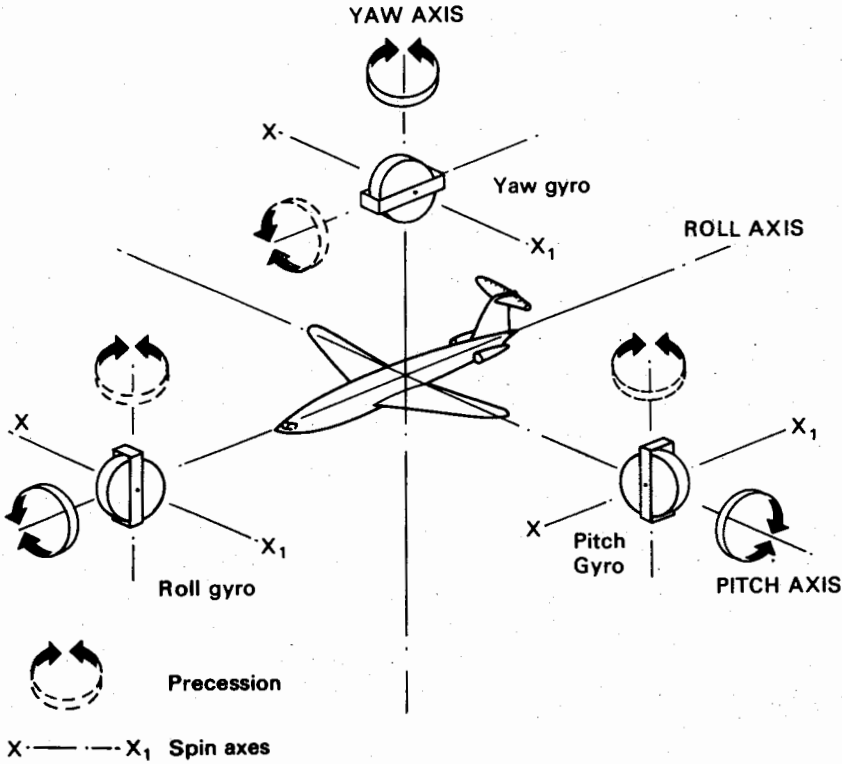
Fig. 3.6 Attitude sensing by rate gyroscopes.

an aircraft and its flight control surfaces but also on the rate of change of these displacements. In these systems, a vertical-axis gyroscope unit detects displacements as a primary attitude change sensor, and applies appropriate corrective control, but as there is always a tendency for a displacement sensing device to cause 'overshoot' the displacement corrections are damped by rate signals from rate gyroscopes. When the displacement error is no longer changing, the rate signals fall to zero and the error existing at that instant is acted upon by control forces that are proportional only to the magnitude of the error.

Thus, by combining the signals from the attitude sensing elements coupled to each type of gyroscope, the possibility of large deviations from reference conditions is greatly reduced and smoother control application and stability of operation is obtained. Furthermore, very small displacement errors and signals which would otherwise not cause corrective action, if left to a vertical-axis gyroscope alone, are detected more quickly by rate gyroscopes and can be built up into stronger corrective signals.

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Levelling and compensation systems

As we learned earlier in this chapter, a free gyroscope must be 'tied' to the earth by gravity sensing, before it can be utilised as a practical aircraft attitude sensing element; this applies in particular to horizontal-axis and vertical-axis gyroscopes. It is necessary, therefore, to provide systems which will level erecting gyroscopes to their appropriate reference datums, and compensate for any tendency to depart from them.

The design of systems varies, but in all cases they operate by applying torques which result in gyroscopic precession.

An erection system which is commonly adopted is shown schematically in fig. 3.7. It consists of two electrolytic switches of the variable resistance type; one positioned so as to detect movement of the whole gimbal system about the roll axis, and the other to detect movement of the inner gimbal ring about the pitch axis. Both switches are mounted on the inner gimbal ring which encases the gyro motor. The *roll* erection switch is connected to a torque motor mounted parallel to the *pitch axis* such that the stator is fixed to the outer gimbal ring and its rotor is fixed to the inner gimbal ring. The *pitch* erection switch which is mounted parallel to the *roll* axis is connected to a similar torque motor having its rotor connected to the outer gimbal ring, and its stator fixed to the frame of the unit. The stators are two-phase wound; one phase, known as the reference phase is continuously energised from an a.c. supply (115 volts or 26 volts in some cases) while the other phase (the control phase) is double wound with one winding for each direction of rotation. The rotors are of the squirrel-cage type with no windings.

Each switch consists of a sealed glass tube, or envelope, partially filled with an electrolytic solution, and containing three electrodes: one connected to the a.c. supply and each of the other two connected to the respective sections of the control phase windings of the appropriate torque motors.

When the gyro rotor axis is stabilised in the vertical position, both switches are horizontal and as the end electrodes are immersed in equal amounts of electrolyte, the current flowing in the two sections of the control phase winding and electromagnetic effect in the rotor will be equally opposed and so no torques are applied.

Assume that the gimbal system is tilted, say to the right, about the roll axis, then the tilting of the roll erection switch will cause a change in the amount of surface area of electrolyte in contact with the electrodes. This in turn causes an imbalance in the electrical resistance, e.g. the resistance will be lower at the 'low' end electrode than at the 'high' end electrode. As a result, more current will flow in that half of the torque motor stator control phase winding connected to the 'low' end electrode, and the electromagnetic effect induced in the squirrel-cage rotor will be increased. The tendency for the rotor to rotate is resisted by the gyro and so a reactive torque is applied to the inner gimbal ring, causing the whole gimbal system to precess about its roll axis and so take up its stabilised vertical position once more.
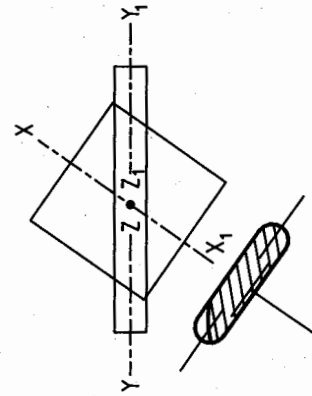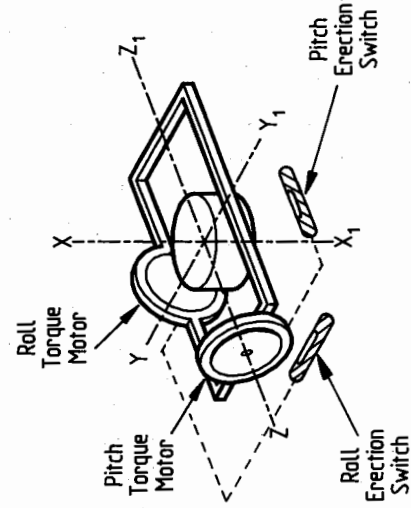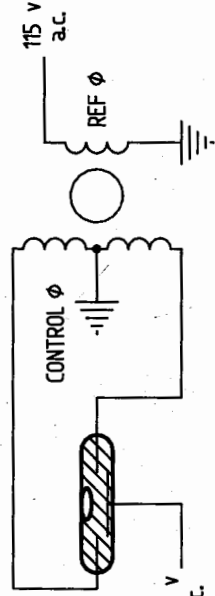
Fig. 3.7 Torque motor and levelling switch system.

If the gyro rotor axis is tilted either forward or rearward about the pitch axis of the gimbal system, then by action similar to that just described, the pitch torque motor will apply a torque to the outer gimbal ring in order to precess the gyro back to the vertical position.

Electrolytic switches, although quite satisfactory for the purpose of maintaining gyroscope axes at their respective reference datums, are at a disadvantage when exposed to the forces developed under accelerating, decelerating and turning conditions. For example, accelerating or decelerating forces will cause the electrolyte of the pitch erection switch to be displaced and the torque produced by the pitch torque motor will precess the gyro rotor axis forward or rearward as the case may be. In other words, the gyro is made to take up a false vertical position. Similarly, during a turn the roll erection switch electrolyte will be displaced by centrifugal force, and the rotor axis will be precessed to a false vertical position left or right about the roll axis. It is therefore necessary to compensate for these effects, otherwise the synchros on the respective axes of the gimbal system will transmit false attitude command signals to the automatic contol system.

The method generally adopted is one which employs additional electrolytic switches positioned on the pitch and roll axes of the gyro, and connected to the torque motors in such a way that under the influence of forces they isolate the control phase windings from the pitch and roll erection switches. The operating principle may be understood from fig. 3.8, which illustrates the circuit arrangement appropriate to pitch erection switch isolation or 'cut-out'.

The cut-out switches are arranged as a pair set at an angle to each other so that there is differentiation between acceleration and deceleration forces. If we consider the effects of an acceleration in the direction shown, the electrolyte in the pitch erection switch will be displaced in the opposite direction and the unbalanced current condition produced in the torque
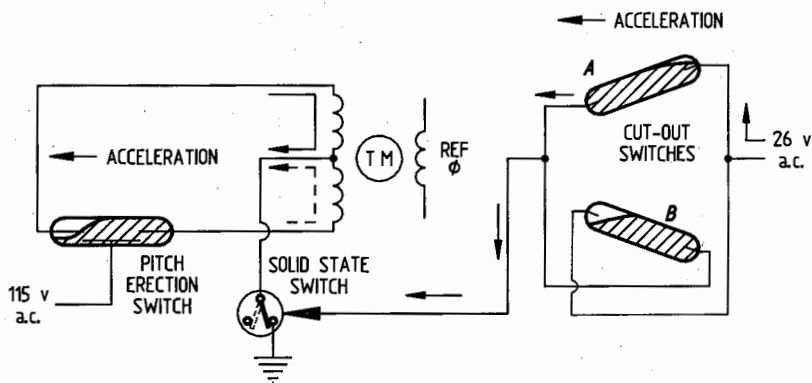
Fig. 3.8 Erection cut-out switch operation.

motor control phase winding will tend to precess the gyro away from the vertical position. At the same time however, the acceleration force displaces the electrolyte in the cut-out switch 'A', and as will be noted, it completes a circuit to a solid-state switch which is then operated to interrupt the control phase winding circuit to ground, thereby preventing torque motor operation. Switch 'B' performs a similar function which is only effective under the influence of a decelerating force.

Roll erection switch 'cut-out' is accomplished by an arrangement ident-ical to that shown, except that the cut-out switches are angled with respect to each other about the roll axis to differentiate between the centrifugal forces produced when turning either left or right.

Sideslip sensing and turn co-ordination

As discussed earlier (see page 40), the displacement of the ailerons for initiating a turn, causes a difference between the lift force of each wing and a related difference in drag, with the result that the aircraft will yaw in a direction that will oppose the rolling moment required for turning. The yaw, the extent of which depends on the roll angle and on the airspeed, also produces sideslip in a direction opposite to that of the yaw (see page 21). Another effect produced by turning is loss of lift resulting in a nose-down displacement of the aircraft. Turning is therefore a rather complex manoeuvre requiring counteraction of the effects of several forces, and the application of simultaneous counteracting displacements of all control surfaces in order to achieve a co-ordinated turn.

The primary gyroscopic-type attitude sensing elements of an automatic flight control system, are somewhat limited in their capability of di-rectly establishing the requisite control signals for turn co-ordination. For example, under sideslip conditions the movement of the aircraft causes bodily displacements of the stabilised axes of the elements, so that aircraft movement is no longer relative to the axes. It is therefore necessary to provide 'back-up' methods of detecting yaw and sideslip effects, and of producing co-ordination signals for integration with those of the main control channels. Various methods of achieving this are adopted and some representative examples are described in the following paragraphs; some methods of counteracting nose-down displacements will be described under 'Lift compensation' on page 144.

Figure 3.9 illustrates a method of achieving turn co-ordination and is one in which the required corrective control signal is established as a result of any difference between a desired true yaw rate signal, and an aircraft yaw rate signal derived from two accelerometer type sensors. One sensor is mounted at the CG position of the aircraft, while the other is mounted at some forward position in the fuselage; their signal outputs are summed at junction 2.
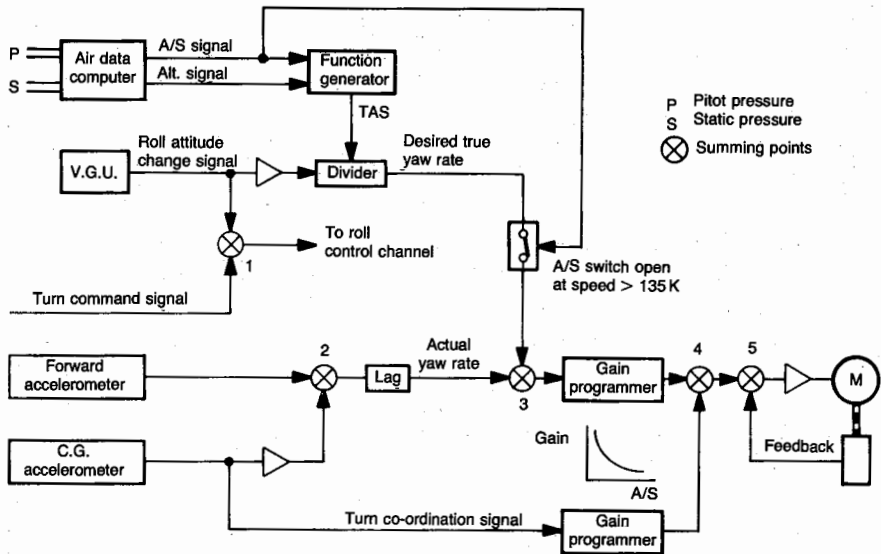
Fig. 3.9 Turn co-ordination.

When a turn command is applied to the system, the signal produced is supplied to the roll control channel via summing junction 1, and after processing, the roll servomotor displaces the ailerons in the appropriate direction and the turn is initiated. The change in attitude about the roll axis is sensed by the vertical-axis gyroscope unit, and so its synchro detector element will develop a roll attitude change signal which is summed with the turn command signal at junction 1. The resultant signal therefore limits aileron displacements to ensure that the aircraft is flown at the roll angle required for the turn.

Since an opposing yawing motion occurs, it is necessary for the rudder to be displaced in a direction required to keep the aircraft in the turn; thus, a signal must be supplied to the yaw control channel and the rudder actuator. In the system illustrated, this is achieved by feeding the roll attitude change signal from the vertical-axis gyroscope unit to a circuit network which divides the signal by a true airspeed signal calculated by a function generator, the latter being supplied with airspeed and altitude signals from an air data computer. The output signal from the divider circuit therefore establishes what is termed the desired true yaw rate of the aircraft at the particular roll angle and airspeed, and it is supplied to the yaw control channel from summing junction 3, for processing and operation of the rudder actuator.

Ideally, the desired true yaw rate signal only should be sufficient for achieving turn co-ordination, but variations in an aircraft's aerodynamic and control characteristics require additional monitoring of yawing motion

and the derivation of a signal corresponding to the actual yaw rate of the aircraft. The yawing motion exerts a force on the mass within the forward accelerometer unit, causing it to be displaced in the appropriate direction, and thereby induces a signal which is applied to summing junction 2. Similarly, a signal will be induced by the accelerometer unit at the CG position as a result of sideslip force causing displacement of its mass, and since this signal is also applied to summing junction 2, then the sum of the signals from both accelerometer units corresponds to actual yaw rate and is applied to summing junction 3. If, at this moment there is a difference between actual yaw rate and the desired true yaw rate, the remaining signal is gain programmed (see also page 154) as a function of indicated airspeed and fed to the yaw control channel and rudder actuator which repositions the rudder to provide further damping of the yawing motion, and co-ordination of the turn. It will be noted from the diagram that gain is reduced as airspeed increases, since smaller displacements of the rudder are then required.

The actual yaw rate and the true yaw rate signals are essentially of a short-term nature, and so if it is necessary to continually co-ordinate an extended turn, a signal capable of being sustained for longer periods is required. Reference to fig. 3.9 shows that this co-ordination signal is derived by 'tapping off' the output from the accelerometer unit at the CG position, and applying it to summing junction 4 via a lag circuit element and a second gain programmer. The signal to the rudder actuator therefore produces a steady state displacement of the rudder to oppose sideslip and to maintain co-ordination.

In order to ensure compatibility of an AFCS with the flight handling characteristics of a type of aircraft in which it is installed, it is sometimes required that at airspeeds above or below certain values, the input of computed yaw rate signals should either be isolated completely, or their gain reduced. For example, the aerodynamic and flight control charac-teristics of an aircraft may be such that during turns above a specified airspeed, co-ordination is an inherent characteristic, and any further application of a computed yaw rate signal could induce an undesirable condition of spiral divergence (see page 26). A method of overcoming this, is to pass signals through an airspeed-monitored switch or switches in some cases. In the system shown in fig. 3.9 a switch is inserted in the desired true yaw rate signal line, and when the airspeed sensed by the air data computer exceeds a specific value (135 knots in this case) the switch is opened. Thus, yaw rate control is then related solely to the actual yaw rate of the aircraft as sensed by the accelerometer units.

Another example of producing turn co-ordination signals is shown in fig. 3.10. In this case, initial rudder displacement is provided by a command signal which is derived by cross-feeding the roll angle signal from a vertical gyroscope unit to a turn co-ordination high-pass filter. The filtered signal is then a yaw function of rate of change of roll attitude, and producing
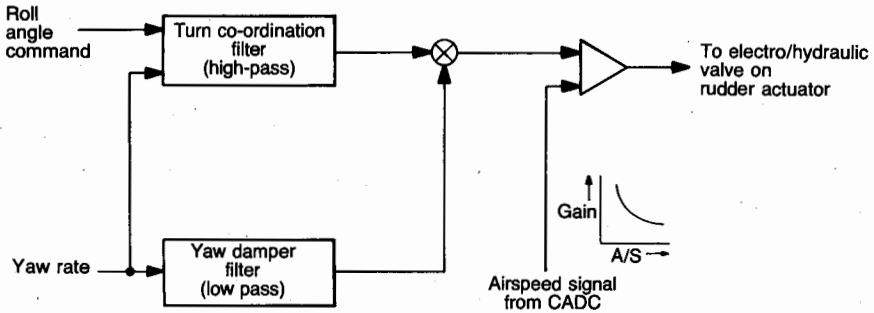
Fig. 3.10 Turn co-ordination.

proportional rudder displacement on a short-term basis. Long term rudder displacement is attained by applying the signals from a yaw rate gyro to a yaw damper low-pass filter which passes only the steady-state yaw rate signals present during a sustained turn. The summated turn co-ordination signal is modified within the servo amplifier as a function of airspeed such that the amplifier gain is programmed to compensate for the effectiveness of the rudder as airspeed is varied. Thus for example, low rudder displacement is required when airspeed is increased and therefore, the gain is reduced.

Inertial attitude references

In aircraft utilising inertial (IN) or inertial reference (IR) systems, it is no longer necessary for the associated automatic control and flight director systems to be dependent on their own individual gyroscopic sensing elements (see also page 190). The reason for this is, quite simply, that an IN or IR system depends for its operation on inertia sensing elements which must always be stabilised horizontally with respect to the aircraft's axes, and since gyroscopes are used for this purpose, then together with certain position transmitters they can also establish the attitude references and steering commands required for automatic flight control.

The method of integrating all three systems is schematically illustrated in fig. 3.11. The inertia sensing elements (not shown in the diagram) are accelerometers producing signal outputs which, after processing by the IN computer, provide data essential for en-route navigation and guidance of an aircraft.

The gyroscopes used in this example of IN system are of the rate integrating type (i.e. on that provides an output displacement proportional to the integral of the input rate) and as will be noted there is one for each axis and they are mounted on two platforms within a gimbal system. Their input and output axes are arranged so that they sense attitude changes in

INERTIAL NAVIGATION UNIT

AUTOMATIC FLIGHT CONTROL SYSTEM

COMPASS COUPLER

YAW DAMPER COMPUTER

ROLL COMPUTER

PITCH COMPUTER

Platform Heading

I.A   Input Axis
O.A   Output Axis

Yaw Position Transmitter

Yaw Gyro

Roll Gimbal Position Transmitter

Roll Attitude

Roll Gyro

Pitch Attitude

Pitch Gimbal Position Transmitter

PITCH AXIS

YAW AXIS

ROLL AXIS

Pitch Gyro
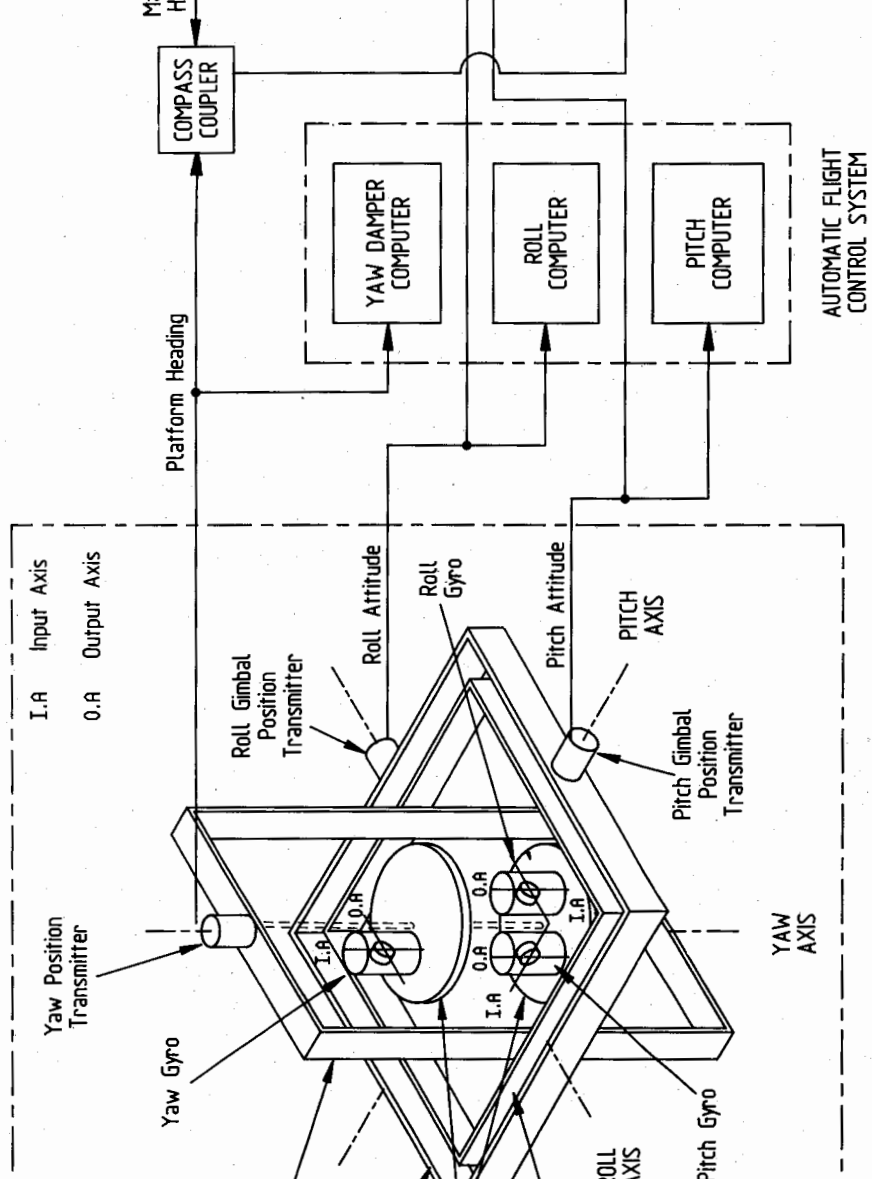
references from an inertial platform.

an identical manner to the three-axis rate transmitter already described (see page 116). The gimbal system of each gyroscope is mechanically coupled to a pick-off element, and each of these is electrically connected via an amplifier to individual torque motors. The motors are mounted on the gimbal rings of the IN unit such that they are disposed about the three principal axes.

Whenever there is a change in attitude of the aircraft about either of its axes, the gimbal system and platforms of the IN unit are tilted. This is sensed by the appropriate rate gyroscope which precesses about its output axis so that its pick-off will develop an output signal proportional to the number of degrees of displacement, enabling the torque motor to drive its associated gimbal ring of the IN unit to a position at which the platforms are again level.

Let us consider as an example a displacement of the aircraft occurring about the roll axis. Since the IN unit gimbal system and casing are 'tied' to the aircraft, the platforms will also be displaced. This displacement will, however, be sensed by the roll rate gyroscope whose pick-off transmits a signal via an amplifier to a torque motor on the inner gimbal ring. The torque motor therefore drives the inner gimbal ring about its roll axis to restore the platforms to the level position. At the same time, a signal is produced by a resolver synchro, the rotor of which is positioned by movement of the inner gimbal ring. The output signal from the synchro is applied to a torque motor on the outer gimbal ring, so as to drive this ring about the roll axis on which it is supported within the unit casing. The outer gimbal ring also rotates the pitch gimbal ring about the roll axis. The reason for having this second gimbal ring is to prevent 'gimbal lock' which would otherwise occur between the inner gimbal ring and pitch gimbal ring.

In order to transmit the degree of displacement to the automatic flight control and flight director systems, synchros are mounted on the IN unit gimbal rings to detect their positions; there is one synchro for each principal axis. Thus, for the roll displacement we have considered, as soon as the torque motors start driving the inner and outer roll gimbal rings to the 'platform level' position, the rotor of a control synchro is rotated within its stator by the outer roll gimbal ring. The signal produced in the stator corresponds to the angular difference between the outer roll gimbal ring and the aircraft's position about the roll axis. In other words, it is a roll attitude error signal and is supplied to the roll control channel of the AFCS computer as a command to restore the aircraft to its normal attitude. As will be noted from fig. 3.11, the signal is also supplied to the flight director system attitude director indicator which then displays the changes taking place in aircraft attitude.

Levelling of the platforms and production of output signals to an automatic flight control and flight director system, as a result of aircraft displacements about the pitch axis, are implemented in a similar manner.

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There is, however, a difference in the case of displacements about the yaw axis. The yaw rate gyroscope, in detecting the change in azimuth, causes its pick-off to develop an output signal which is supplied to a torque motor on the yaw axis of the inner gimbal ring. Unlike the pitch and roll torque motors, the azimuth motor cannot rotate the gimbal ring system, and so instead, it rotates the levelled platform on which it is mounted, in the opposite direction to aircraft displacement, thereby maintaining it in the same position relative to the gimbal system. The platform rotates the rotor of an azimuth control synchro to produce a platform heading signal that is supplied as a command to the yaw damper channel of the automatic flight control computer, and also to the flight director system horizontal situation indicator for display of actual heading changes.

Location of attitude sensing elements

The locations of attitude sensing elements, and in particular those which are remote from the flight crew compartment of an aircraft, have to be carefully chosen otherwise the inherent flexibility of the aircraft structure will be interpreted by sensing elements as attitude changes, and thereby result in a condition of unstable coupling between the aircraft and its automatic flight control system. Locations must, therefore, be in areas where flexural frequency sensing is at a minimum.

TEST QUESTIONS

1 Define the two fundamental properties of a gyroscope and state the factors on which they depend.
2 What corrections must be applied to a basic gyroscope in order to ensure its suitability as an attitude sensor?
3 What is meant by the term 'earth rate'?
4 What is meant by 'transport wander', and how does it relate to horizontal-axis and vertical-axis gyroscopes?
5 Explain the operating principles of a rate gyroscope, and also how the principle can be applied to the sensing of aircraft attitude changes about all three axes.
6 How can a single rate gyroscope be adapted to sense both roll and yaw attitude changes?
7 For what purpose are displacement gyroscope and rate gyroscope elements employed in combination with each other?
8 Why is it necessary to provide additional detecting elements to ensure automatic turn co-ordination?

9 Describe the operation of a typical method of achieving turn co-ordination.

10 How are electrolytic switches and torque motors used for the levelling of a vertical-axis gyroscope, positioned in relation to the axes of the gimbal system?

11 Describe how a levelling switch and a torque motor operate when the gyroscope is tilted about the pitch axis.

12 What effects can acceleration forces have on a levelling system, and how are they compensated?

13 How is the stabilising system of an inertial platform used for providing attitude references to an automatic control system?

4

Command Signal Detection

The primary attitude sensing elements employed in an AFCS form the most vital part of the servomechanism loop since they detect attitude changes in terms of errors which demand monitoring and correction. Both these functions, however, cannot be performed by the elements alone, and in this respect they are not unlike the primary flight instruments in that they demand some action on the part of the pilot in his role as the error monitoring and correcting element in the control loop. In other words, it is necessary to provide error signal sensors which by suitable coupling to the attitude sensors will automatically detect the phase and magnitude of the errors, and transmit this intelligence to the output section of the servo-mechanism. Signal sensors, or transducers as they are often called, are normally of the inductive type, although in some basic types of autopilot the principle of variable capacitance is adopted: some typical examples are considered in this chapter.

Inductive elements

The fundamental operating principle of inductive elements is based on that of the conventional transformer, i.e. for a particular ratio between the number of windings of a primary coil and a secondary coil, and for a certain fixed value of voltage input to the primary coil, some higher or lower fixed value of voltage output can be obtained from the secondary coil. In addition, however, the elements must have the ability to change the phases of their output in order to establish the direction of the correcting control signals. In applying this principle to error sensing, it is necessary for the secondary coil output to be of a variable nature, and in order to derive this the basic iron core structure is arranged in two parts, one part being fixed while the position of the other part is variable. Some examples of the methods commonly adopted are the 'E' and 'I' method; the moving vane method; and the method of synchronous transmission.

E and I bar sensors

An example of this type of sensor is illustrated in fig. 4.1. The centre limb of the fixed E-shaped core is wound with a primary coil supplied with alternating current at the required level, while the two outer limbs are wound with secondary coils connected in series opposition. The I bar is pivoted at its centre, and is rotated by the attitude sensing element, in this case of the gyroscopic type, so that the bar varies the air gaps between it and the outer limbs of the E-shaped core. When the I bar is in its neutral position, the air gaps are equal, an equal magnetic flux flows in each limb of the E-shaped core, thereby inducing equal and opposite voltages in the secondary coils; the output from the coils is therefore zero.

When the attitude sensing element detects a change in attitude (either as a result of disturbance of the aircraft or by injection of a required attitude change command signal) the I bar is deflected through a small angle such that the air gaps will no longer be equal, and the reluctance of the magnetic circuit will change. The flux in one outer limb of the E core will increase, while the flux in the other decreases; thus, there will be corresponding changes in the voltages induced in the secondary coils, and an output signal voltage will be produced. The greater the I bar deflection, the greater will be the signal voltages, and since the secondary coils are wound in opposition the voltage will be either in-phase or out-of-phase with respect to the input voltage to the primary coil, depending on the direction in which the I bar is deflected. The phase and magnitude of the signal are directly proportional to the direction and magnitude of the detected attitude change; thus, by amplifying the signal and feeding it to a servomotor, an appropriate flight control surface can be moved either to correct a disturbance or to manoeuvre the aircraft in response to an attitude change command. To ensure that the servomotor provides the required amount of control surface
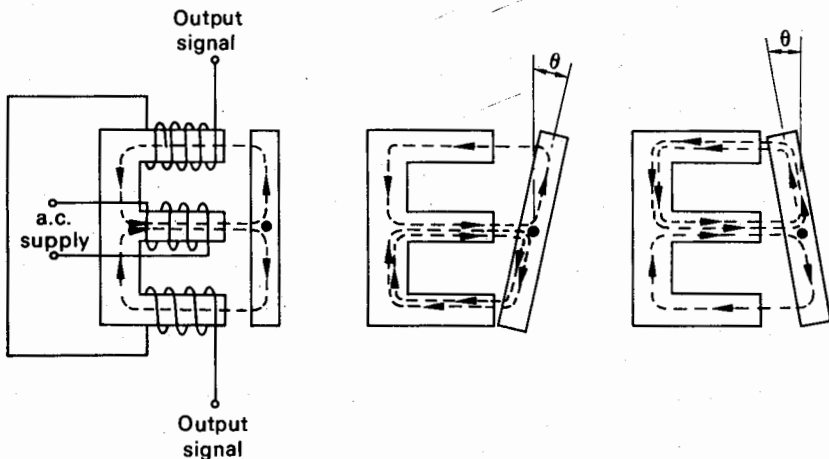
Fig. 4.1 E and I bar sensor.

displacement, or runs at a rate proportional to that of a disturbance as the case may be, a feedback signal is applied to the control loop to oppose the error voltage signal; (some typical methods adopted for the generation of feedback are described in chapters 5 and 7).

In some applications of this type of sensor, the I bar is deflected linearly to produce output voltage signals in the same manner as that described.

Moving vane method

This is adopted in some types of flight control system in which the inductive transducers are actuated by the vertical gyroscope of a steering horizon indicator, an example being one of the flight director/autopilot systems manufactured by Edo-Aire Mitchell.

Two transducers are employed, and they each consist of a moving vane and a fixed coil assembly as shown schematically in fig. 4.2. One transducer has its vane mounted on the inner gimbal ring of the gyroscope, and therefore senses pitch attitude changes, while the vane of the second is mounted on the outer gimbal ring, and senses changes in roll attitude.

A square-wave alternating voltage is applied to coils 1 and 3 of each fixed coil assembly, the voltages being 180° out-of-phase. In the level flight attitude (fig. 4.2 *a*) the vanes of both transducers are centred over the number 2 coils, and since in this position there is equal electromagnetic coupling between coils 1 and 3, no voltage is induced in coil 2 and no command signal will be applied to the computer/amplifier.

When a change in aircraft attitude occurs about one or other of the vertical gyroscope axes, there will be relative movement between the vane and coil assembly of the relevant transducer. Thus, assuming that movement causes the vane to be centred between coils 2 and 3 (fig. 4.2 *b*) the inductance will be increased and a square-wave voltage will be induced in coil 2, and which is in phase with the input to coil 3. A command signal will therefore be applied to the computer/amplifier. Movement of the vane towards coil 1 (fig. 4.2 *c*) results in a similar response except that the voltage induced in coil 2 is in phase with the inputs to coil 1.

Synchronous transmission systems

In many automatic flight control systems, the detection of error signals is accomplished by self-synchronous inductive transducers which are usually classified under the generic term *synchro*. They are divided into four main groups according to their function: (i) torque synchros, (ii) control synchros, (iii) differential synchros and (iv) resolver synchros.

Torque synchros

These are the simplest form of synchro and are used for the transmission of angular position information by means of induced signals, and for the
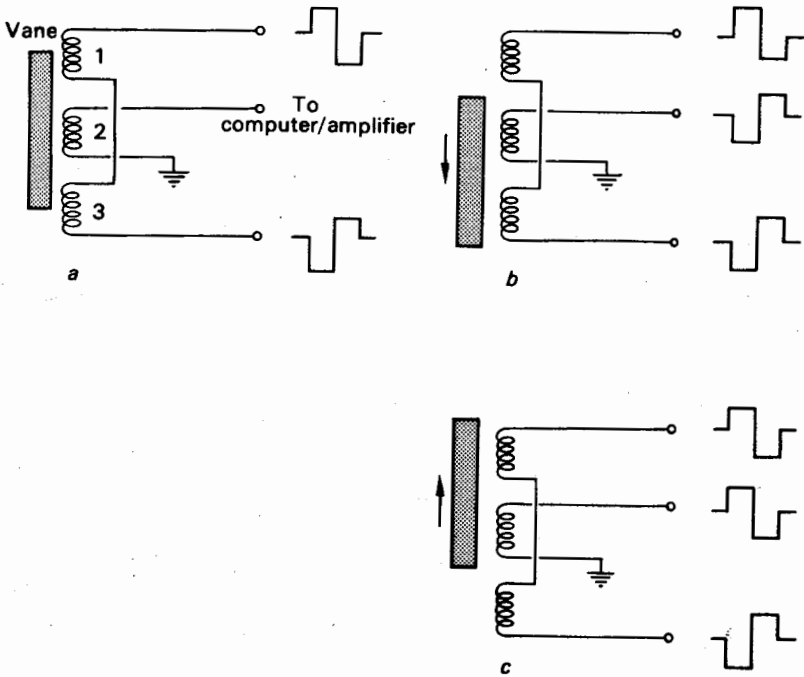
Fig. 4.2 Moving-vane pick-off element.

reproduction of this information by the position of a shaft at an output or receiver element. A typical application of torque synchros is in flight instrument systems.

A torque synchro system is comprised of two electrically similar units interconnected as shown in fig. 4.3, and by convention one is designated the transmitter (TX) and the other the receiver (TR). Each unit consists of a rotor carrying a winding, and concentrically mounted in a stator carrying three windings the axes of which are 120° apart. The principal physical differences between the TX and the TR are that the rotor of the TX is mechanically coupled to an input shaft, while the TR rotor is free to rotate. The rotor windings are connected to a source of single-phase alternating current supply, and the corresponding stator connections are joined together by transmission lines. A similarity between these connection arrangements and a conventional transformer may also be noted; the rotors corresponding to primary windings and the stators to secondary windings.

When the rotors are aligned with their respective stators in the position indicated they are said to be at 'electrical zero'; this refers to the reference angle standardised for synchros at which a given set of stator voltages will be produced, and by this convention enables replacement synchros to be matched to each other. Other positions are measured in degrees increasing
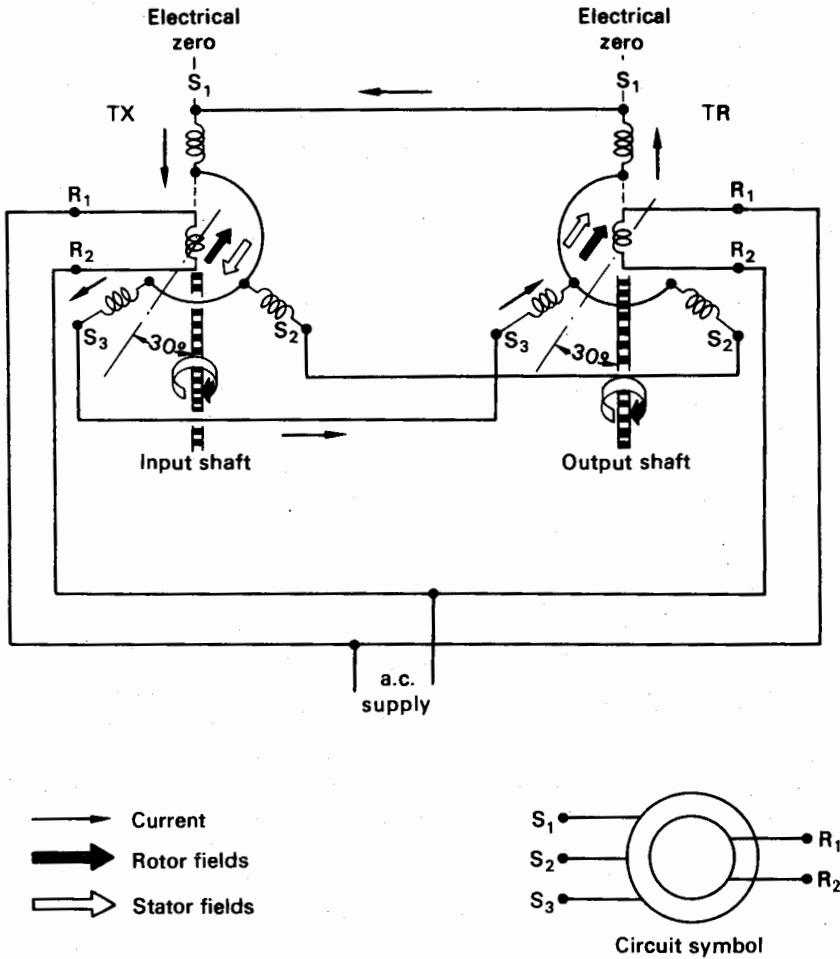
Fig. 4.3 Torque synchro system.

in an anti-clockwise direction when viewed from the output shaft end of the unit. With power applied to the rotors the alternating flux set up will, by transformer action, induce certain voltages in the stator coils, the voltage values being governed, as in any transformer, by the ratio of the number of turns of the rotor (primary) and stator (secondary) coils.

When the rotors are at 'null', the induced voltages will be equal and opposite; therefore, no current flows in the stator coils and so there are no magnetic fields produced to cause rotation of the rotor of TR. However, when the rotor of TX is turned, say, through an angle of 30° an imbalance occurs between the voltages induced in the stator coils, causing current to flow in the stator coils and transmission lines. The currents are greatest

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in the circuit where voltage imbalance is greatest, and their effect is to produce resultant magnetic fields which exert torques to turn the rotor of TR to the same position as that of TX. As the TR rotor continues to turn, the misalignment, voltage imbalance and currents decrease until the 30° position is reached and no further torque is exerted on the rotor.

Control synchros

Control synchros differ from torque synchros in that their function is to produce an error voltage signal in the receiving element, as opposed to the production of a rotor torque. Since this error signal is an indication of misalignment between an input and an output, control synchros are more widely used as attitude error detectors which, as we learned earlier, form part of closed-loop servo control systems.

The interconnection of the two elements of a control synchro system, as applied to a gyroscopic attitude sensing element, is shown in fig. 4.4. By convention, the transmitter is designated at CX, and the receiver designated as a control transformer CT. The CX is similar to a torque transmitter, and from the diagram it will be noted that the alternating current supply is connected to the CX rotor only. The CT rotor is not energised since it acts merely as an inductive winding for detecting the phase and magnitude of error signal voltages which are supplied to an amplifier. The amplified signals are then fed to a two-phase motor which is mechanically coupled to the CT rotor. Another difference to be noted is that a control synchro system is at electrical zero when the rotor of CT is at 90° with respect to the CX rotor.

In practical applications, the transmitters are located about the appropriate axes of the attitude sensing element, for example, the pitch and roll axes of a vertical-axis gyroscope reference unit, the rotors being secured to the gimbal rings. Thus, assuming that a disturbance about the pitch axis takes place, the stator of the pitch CX will rotate about the stabilised rotor through a certain angle and the resultant angle and the resultant flux in the CT stator will be displaced from its datum point by the same angle, and relative to the CT rotor position at that instant. An error voltage is therefore induced in the rotor, the phase and magnitude of the voltage depending on the direction of CX rotor rotation and on the degree of misalignment between it and the CT rotor. The error voltage is then amplified and fed to the control phase of the motor, the other phase being continuously supplied with alternating current. Since the control phase voltage of a two-phase motor can either lead or lag the reference phase voltage, the phase of the error voltage will determine the direction in which the motor will rotate, and its magnitude will determine its speed of rotation. As the motor rotates, it turns the rotor of the CT in the appropriate direction, thereby reducing its displacement relative to the CX rotor.
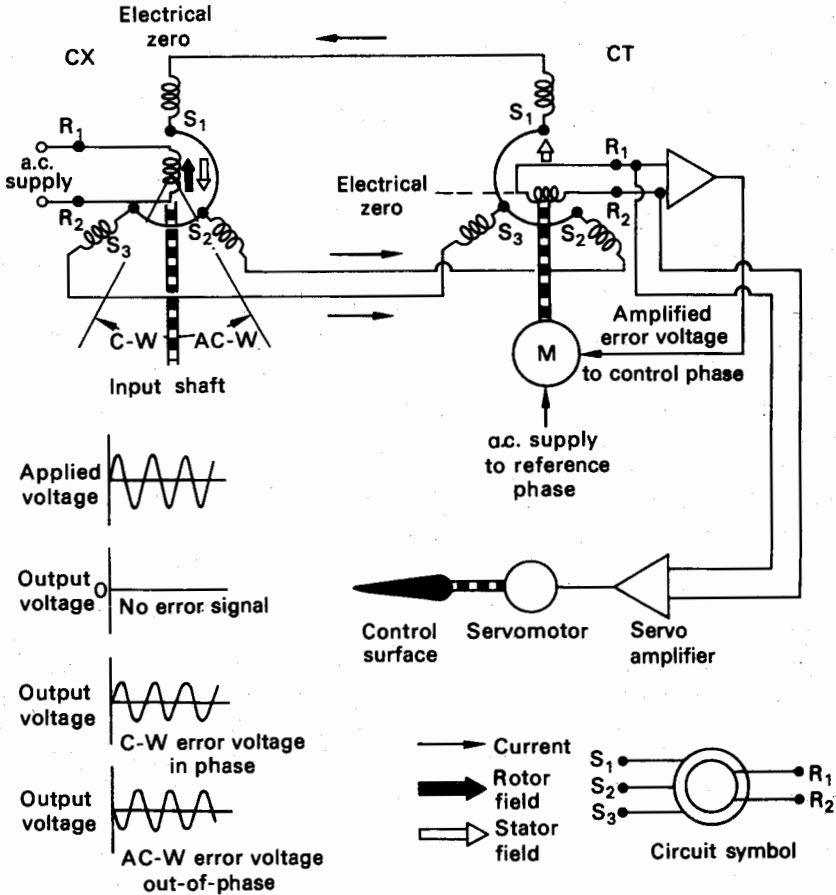
Fig. 4.4 Control synchro system.

Rotation continues until both rotors are in alignment (bearing in mind of course that the electrial zero points are at 90° from each other), at which position no further error voltage is induced.

Following a disturbance of the type considered in this example, it is necessary for the elevator to be displaced in order to return the aircraft to its normal flight attitude. This is accomplished by also applying the error voltage induced in the CT, to the power output section of the servo control channel loop. Thus, as can also be seen from fig. 4.4, the error voltage is applied to a servo amplifier which increases the magnitude of the voltage sufficient to drive the servomotor and so proportionately displace the elevators in the appropriate direction, until there is no further error. As the aircraft returns to its normal flight attitude, the change is again sensed

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by the vertical-axis gyroscope reference unit and the pitch CX, but the error voltage induced is now opposite in phase to that of the original displacement. Thus, the control synchro and pitch servo control systems are operated in the same manner as that described but in the opposite direction.

Differential synchros

In some cases, it is necessary to detect and transmit error signals representative of two angular positions, and in such a manner that the receiver element of a synchro system will indicate the difference or the sum of two angles. This is achieved by introducing a third synchro into either a torque or control system, and using it as a differential transmitter. Unlike TX or CX synchros, a differential transmitter (designated TDX or CDX) has an identically wound stator and rotor interconnected as shown in fig. 4.5.

As an illustration of the operation, we may consider the application of a differential transmitter to a torque synchro system at varying angular inputs as indicated in fig. 4.5. At a the TX rotor is shown rotated clockwise through 60° while the rotor of TDX remains at electrical zero; all the magnetic fields rotate, and the rotor of TR takes up the same angular position as the rotor of TX. If now the TX rotor remains at electrical zero, and the TDX rotor is rotated clockwise through 15°, say, the fields of both synchros remain in the electrical zero position because their position is determind by the orientation of the TX rotor (diagram b). However, a 15°
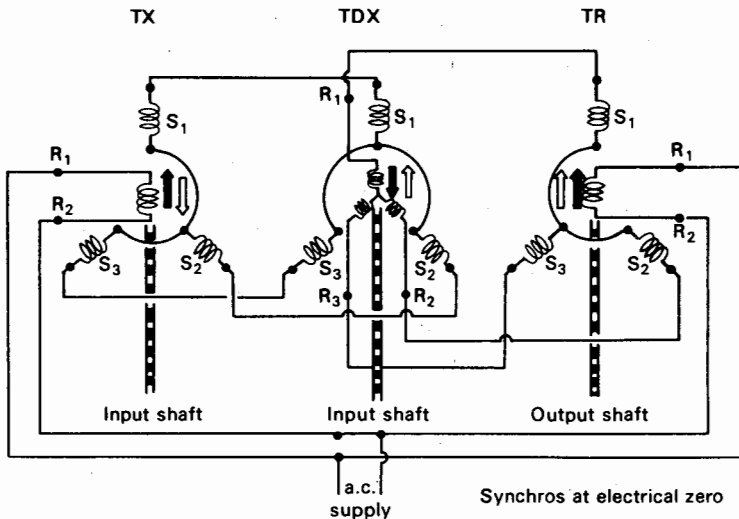
Fig. 4.5 Differential synchro in torque synchro system.

clockwise rotation of the TDX rotor without a change in the position of its field is equivalent to moving the rotor field 15° anti-clockwise while leaving the rotor at electrical zero. This relative angular change is duplicated in the stator of TR and so its rotor will align itself with the field, i.e. for a 15° clockwise rotation of the TDX rotor, the TR rotor will rotate 15° anti-clockwise.

Assume now that the TX rotor is rotated through 60° clockwise, and the TDX rotor through 15° clockwise, then because the TR rotor will rotate 15° anti-clockwise its final angular movement will be equal to the difference between the two input angles, i.e. it will turn through 45° (diagram c). The differential effect is of course reversed when the TDX rotor is rotated in the opposite direction to the TX rotor, so that the TR rotor rotates through an angle equal to the sum of the two input angles. By reversing pairs of leads either between TX and TDX, or between TDX and TR, any one of the
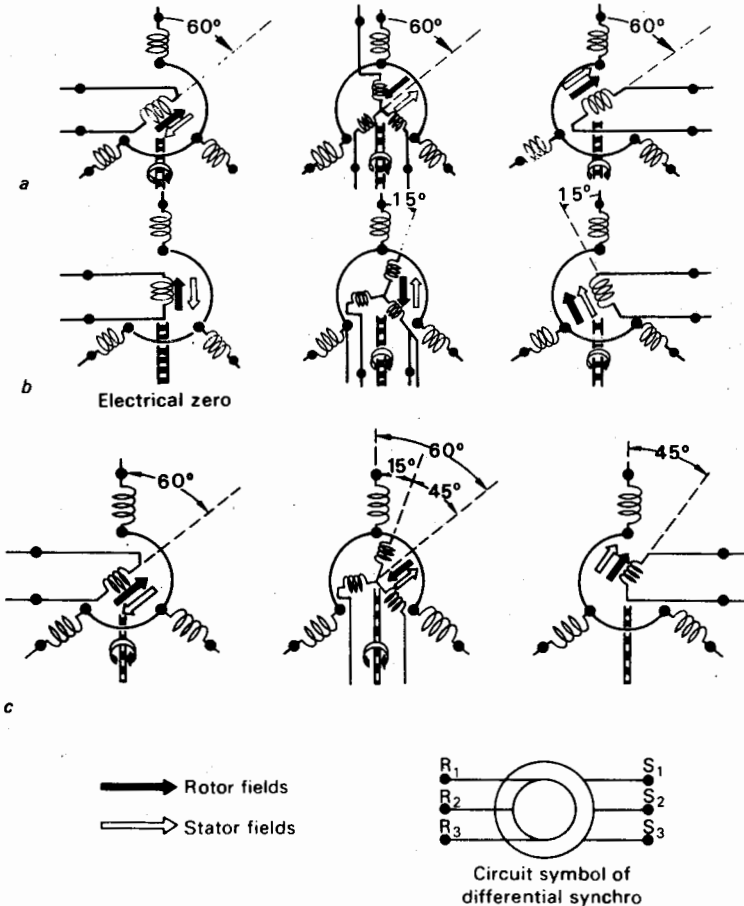
Fig. 4.5 (contd)

rotors can be made to assume a position equal to the sum or the difference of the angular positions of the other rotors.

In the same way that differential transmitter synchros can be used in torque synchro systems, they can be used in systems utilising control synchros to transmit control signal information on the sum or difference of two angles. The basic arrangement is shown in fig. 4.6.

Resolver synchros

The function of resolver synchros (designated RS) is to convert alternating voltages, which represent the cartesian co-ordinates of a point, into a shaft position, and a voltage, which together represent the polar co-ordinates of that point. They may also be used in the reverse manner for voltage conversion from polar to cartesian co-ordinates. A vector representing an alternating voltage can be defined in terms of the vector length (see fig. 4.7) and the angle it makes with the X axis; these are the polar co-ordinates of the vector. The vector r can also be defined in terms of x and y where $x = r \cos \theta$ and $y = r \sin \theta$; these expressions are the cartesian co-ordinates of the vector.

A typical arrangement of an RS for conversion from polar to cartesian co-ordinates is shown in fig. 4.8, and from this it will be noted that the stator and rotor each have two windings arranged in phase quadrature, thus providing an eight-terminal synchro. An alternating voltage is applied to the rotor winding $R_1 - R_2$, and the magnitude of this voltage, together with the angle through which the rotor is turned, represent the polar co-ordinates. In this application, the second winding is unused, and as is usual in such cases, it is short-circuited to improve the accuracy of the RS and to limit the spurious response.

In the position shown, the alternating flux produced by the current through rotor winding $R_1 - R_2$, links with both stator windings, but since the rotor winding is aligned only with $S_1 - S_2$, then maximum voltage will be induced in this winding. Winding $S_3 - S_4$ is in phase quadrature so no voltage is induced in it. When the rotor is rotated at a constant speed it will induce voltages in both stator windings, the voltages varying sinusoidally. The voltage across that stator winding which is aligned with the rotor at electrical zero will be a maximum at that position and will fall to zero after rotor displacement of 90°; this voltage is therefore a measure of the cosine of the displacement. The voltage is in phase with the voltage applied to $R_1 - R_2$ during the first 90° of displacement, and in anti-phase from 90° to 270°, finally rising from zero at 270° to maximum in-phase at 360°. Any angular displacement can therefore be identified by the amplitude and phase of the induced stator voltages. At electrical zero, stator winding $S_3 - S_4$, will have zero voltage induced in it, but at 90° displacement of rotor winding $R_1 - R_2$, maximum in-phase voltage will be induced and will vary sinusoidally
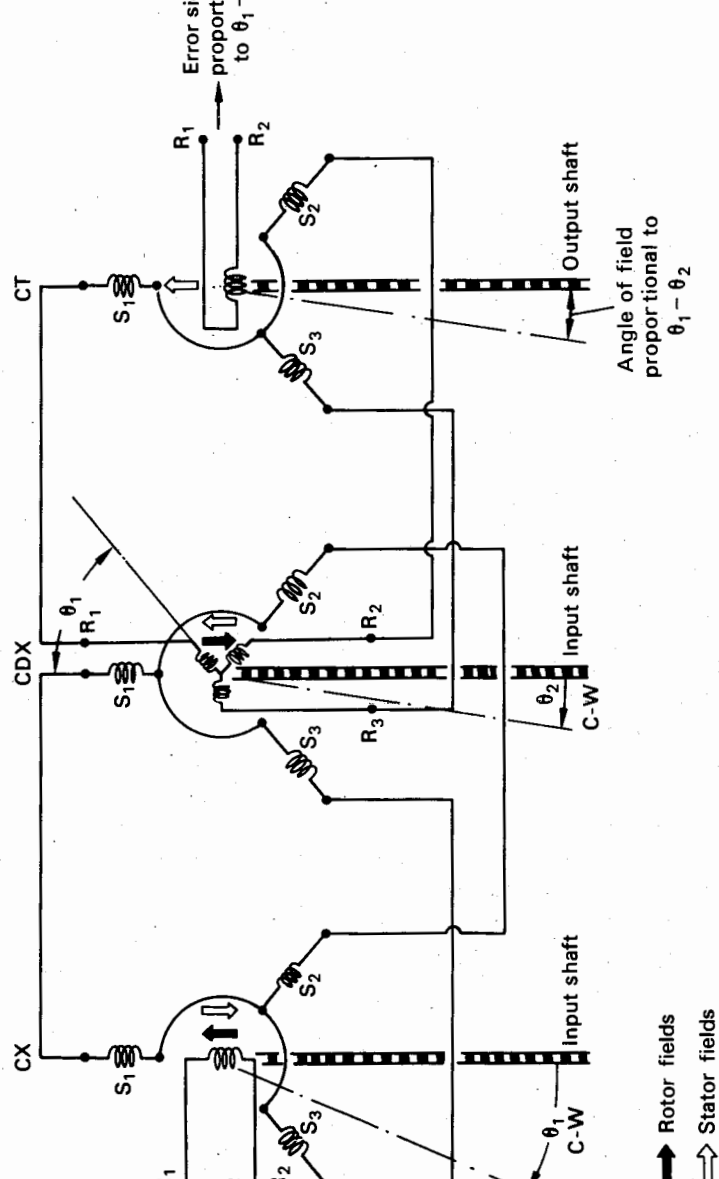
Differential synchro in control synchro system.

Rotor fields
Stator fields
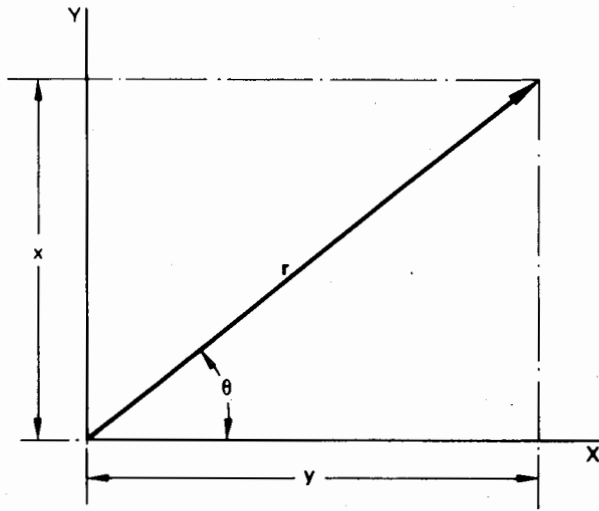
Fig. 4.7 Cartesian co-ordinates of a point.

throughout 360°; thus, the $S_3 - S_4$ voltage is directly proportional to the sine of the rotor displacement. The phase depends on the angle of displacement, any angle being identified by the amplitude and phase of the voltages induced in stator winding $S_3 - S_4$. The sum of the outputs from both stators, i.e. $r \cos \theta$ plus $r \sin \theta$, therefore defines in cartesian co-ordinates the input voltage and rotor rotation.

Figure 4.9 illustrates an arrangement whereby cartesian co-ordinates may be converted to polar co-ordinates. An alternating voltage $V_x = r \cos \theta$ is applied to the cosine stator winding $S_1 - S_2$, while a voltage $V_y = r \sin \theta$ is applied to the sine stator winding $S_3 - S_4$. An alternating flux representing cartesian co-ordinates is therefore produced inside the complete stator. One of the rotor windings, in this case $R_1 - R_2$, is connected to an amplifier and in the position shown it will have maximum voltage induced in it; this voltage will be applied to the amplifier. The output from the amplifier is applied to a servo-actuator which is mechanically coupled to a load and to the rotor. When the rotor is turned through 90° the induced voltage in winding $R_1 - R_2$, reduces to zero and the servo-actuator will stop. The rotor winding $R_3 - R_4$ will now be aligned with the stator flux, and a voltage will be induced in it which is proportional to the amplitude of the alternating flux as represented by the vector r, i.e. a voltage proportional to $\sqrt{(V_x^2 + V_y^2)}$. This voltage together with the angular position of the rotor therefore represents an output in terms of the polar co-ordinates.

A typical example of RS application to a control channel of an AFCS is shown in fig. 4.10. In this case, the control system is one employing rate gyroscopes and a vertical-axis gyroscope, for attitude sensing, and the

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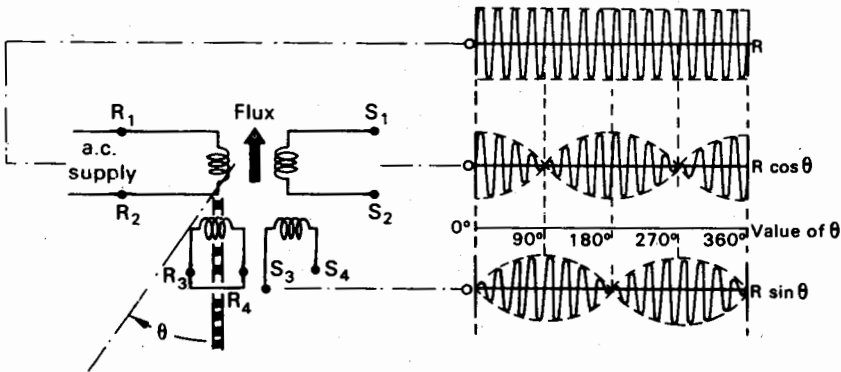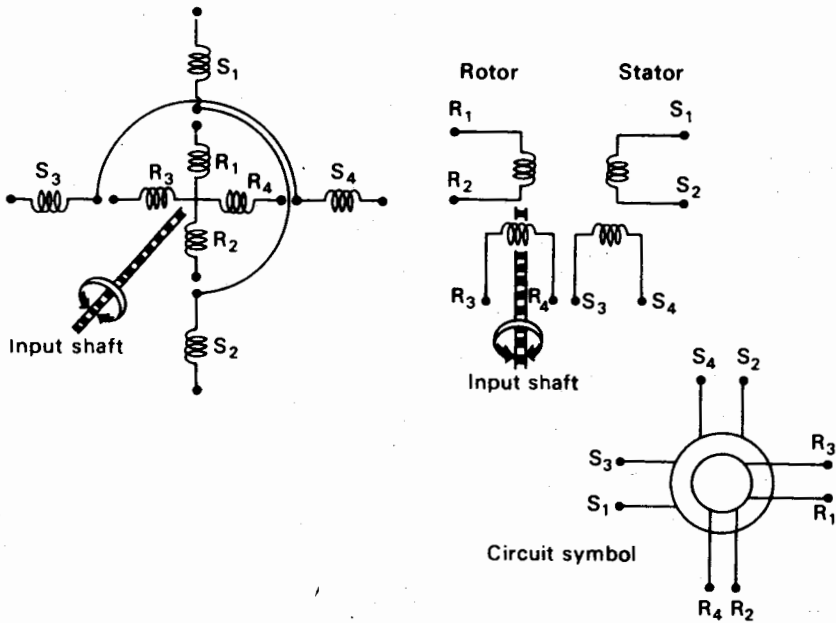
Fig. 4.8 Resolver synchro.

resolver performs the function of converting cartesian co-ordinates to polar co-ordinates in order to establish a true pitch rate (movement of an aircraft about a true horizontal axis) for pitch stabilisation. When the aircraft turns, the outputs from the pitch and roll rate gyroscopes contain a rate-of-change-of-heading component, and the output from the yaw rate gyroscope contains a rate-of-change-of-elevation component. The effect of the component in the pitch rate signal is to produce a continuous pitch-down signal; the elevation rate component is nulled by the yaw channel circuit.
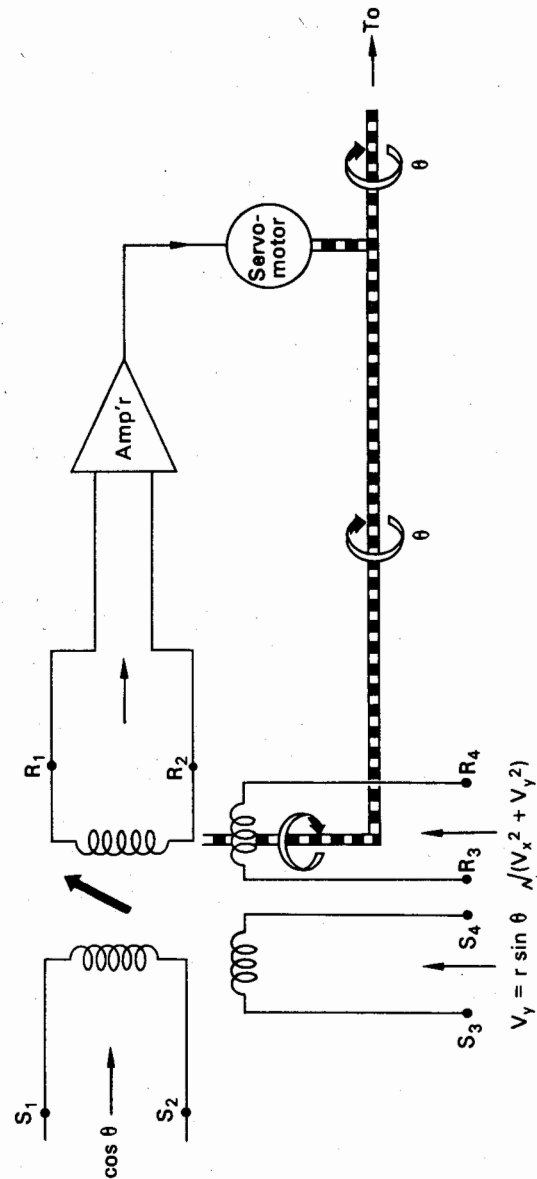
Conversion of cartesian co-ordinates to polar co-ordinates.

True pitch rate signal to servo amplifier and pitch servomotor

Yaw rate

O-V

Pitch rate

RS

Roll attitude signal

CT

Amp'r

Control φ

M
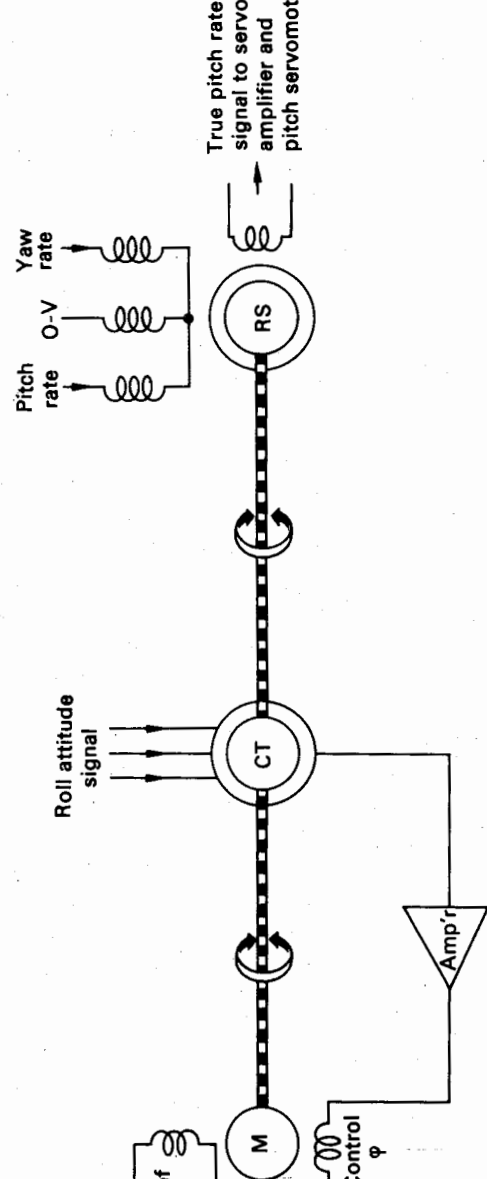
Resolver synchro application.

If, therefore, the pitch and yaw rate signals can be resolved with respect to the roll rate then the true pitch rate of the aircraft during turns can be determined.

The roll attitude signal is fed to the stator of a CT synchro and this establishes an error signal representing the roll angle, in the rotor. After amplification, the signal is applied to the control winding of a motor, the shaft of which is mechanically coupled to the CT rotor and to the RS rotor. The motor therefore rotates the CT rotor to reduce the error signal to zero (generally termed datum chasing) and also rotates the RS rotor to a position which mechanically represents the roll angle at that instant. The RS stator is electrically connected to the pitch rate signal and yaw rate signal amplifier, so that during turns, two rate signals are supplied to the stator which have a sine/cosine relationship, and produce a resultant error signal with respect to the rotor, the signal representing the true pitch rate.

Lift compensation

As already pointed out (see page 122) an aircraft has the tendency of being displaced nose-down during a turn. In order to correct for this, the command signal sections of an automatic flight control system therefore require the addition of what is termed a lift compensation circuit. One circuit arrangement is shown in fig. 4.11 and from this will be noted that it is essentially a synchronous transmission network, which generates a signal corresponding to the trigonometric function of an angle known as the versine. Signals from the pitch and roll attitude detector CX synchros of a vertical gyroscope unit are supplied to the stators of an RS, and as the rotor windings are in phase quadrature, the signals induced in them are the sine and cosine of the transmitted attitude signals. The pitch and roll sine signals are fed to the appropriate corrective control channels, while the cosine signals are fed to a network which places them in opposition. At zero pitch and roll angles, the cosine of each angle is unity and, being in opposition, the output to the pitch channel is zero. At some angle of roll the cosine of the angle decreases, while that of the pitch angle remains substantially the same, no movement in pitch having taken place. The opposing signals in this condition are, therefore, cosine of pitch angle equal to unity, and cosine of roll angle less than unity; thus, the resulting signal is the versine $1-\cos\theta$, where θ is the roll angle in degrees. This signal is fed to the pitch servo channel to provide the required nose-up correction during turns. Since the cosine of the roll angle decreases as the angle increases, the versine signal will also increase and so result in greater nose-up correction.

The resolver shown in fig. 4.10 also performs a similar function by supplying its true pitch rate signal output to the pitch channel servo amplifier.

A further example of versine signal generation is shown in fig. 4.12.

Bank attitude

Bank (sine)

$1-\cos\theta$

Pitch (sine)

Pitch attitude

Bank (cosine)

Pitch (cosine)

RS

RS

X

X

ation

ation

sine generator.

Versine signal generation.

Labels visible in figure: Cosine, Versine Output, A.C. Supply, RS, A.C. Ref Signal, Sine, Clutch, Gearbox, Mechanical Coupling, TG, M, Mode Controlled Rate Feedback, Roll Rate Command Limiter, Heading Navigation Signals, 1, 2, 3
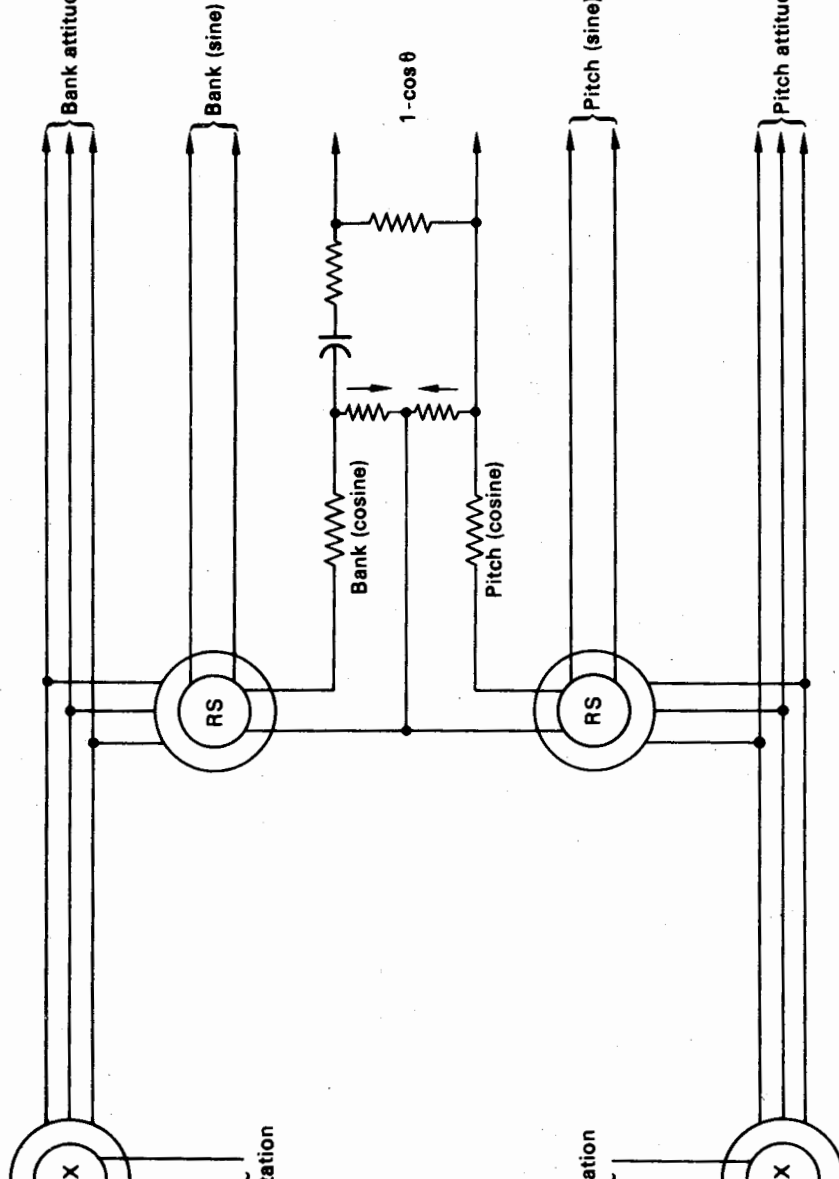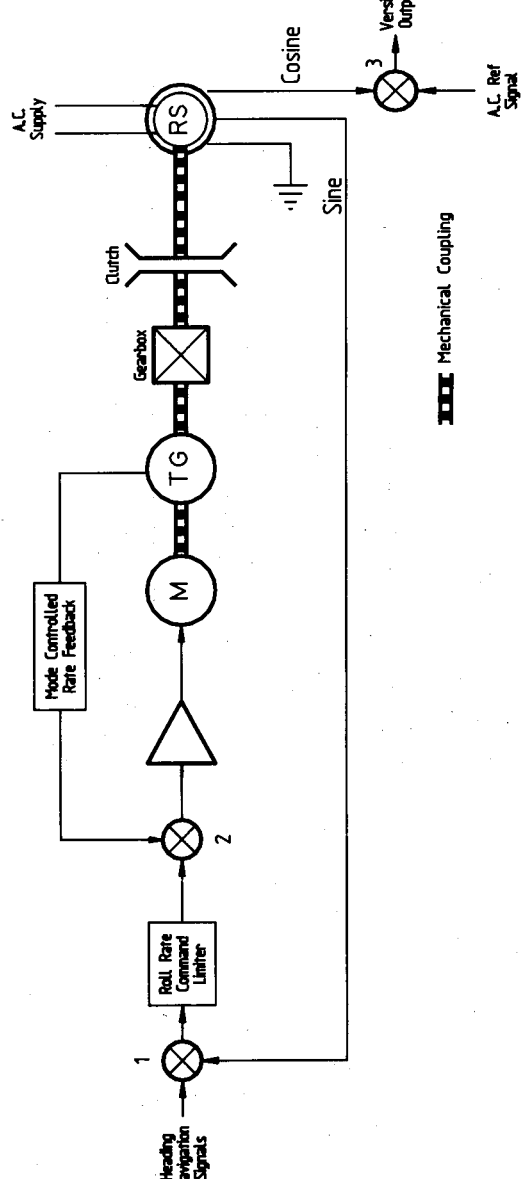
Whenever the motor is driven out of the zero bank angle position, as a result of steering commands, the resolver generates a sine signal output that is summed at point 1 with the steering command signals. At the same time, a cosine signal is produced which, on being summed at point 3 with an a.c. reference signal, forms the versine output to the pitch channel.

TEST QUESTIONS

1 What is the fundamental operating principle of an inductive type error signal sensor?

2 Explain the operation of an 'E' and 'I' bar type sensor when an attitude change occurs.

3 Describe how a torque synchro system operates.

4 What would be the effect on a receiver torque synchro response in the event that cross-connections had inadvertently been made between (a) transmitter and receiver power supplies, and (b) transmitter and receiver stators?

5 What is the function of a control synchro system, and how does its operation differ from that of a torque synchro system?

6 State the purpose of a differential synchro and how its internal circuit arrangement differs from that of either a torque or control synchro.

7 What do you understand by the terms 'polar' and 'cartesian' co-ordinates in relation to voltage conversion?

8 Explain the operation of a synchro which will produce the voltage conversion referred to in Question 7.

9 Why is it necessary to provide a lift compensation circuit in a command signal section of an automatic flight control system? Explain how such compensation is applied.

5
Command Signal Processing

The signals produced by error signal sensors, in whatever form the sensors may take, cannot be applied directly to their associated servo-actuators for the principal reasons that further computation of signals is necessary particularly when outer loop control is adopted (see chapter 6), and in terms of power capability they are not strong enough to cope with the aerodynamic loads acting on the control surfaces. Therefore, in any one flight control system it is necessary to incorporate within the corresponding servo control loops a signal processing system having some, or all, of the following functions:

(i)   Differentiating, e.g. deriving simulated rate information from a vertical-axis gyroscope controlled signal sensor.
(ii)   Demodulating, i.e. converting a.c. error signals into d.c. control signals which have the same phase relationship.
(iii)   Integrating to obtain simulated attitude information or to correct any sustained attitude errors.
(iv)   Amplifying to increase sensor signals to a level high enough to operate the servomotors.
(v)   Limiting to ensure that certain parameter changes are kept within prescribed limits.
(vi)   Gain adjustments that may be pre-set and/or automatically programmed to adapt system response to suit the handling qualities or flight path of an aircraft.
(vii)   Programming to produce a precise manoeuvre, e.g. when selecting a particular outer loop control mode (see chapter 6).
(viii) Applying feedback signals to ensure that corrective control is proportional to command signal inputs.

The methods adopted for performing the foregoing functions are varied and, in common with the other elements which make up any one complete control system, such methods are governed by system design and how it relates to the handling characteristics of various types of aircraft. Within the confines of this chapter, therefore, it is not possible to go into any great

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detail of the individual control channel configurations. There is, however, a basic similarity between the requirements for, and the applications of, signal processing elements, and the fundamentals of purpose and operation may be understood with the aid of the block diagram of fig. 5.1. Although the diagram is based on a roll control channel of a particular type of flight control system, it may nevertheless be considered generally representative.

When a turn has been commanded either through the medium of the magnetic heading reference system (MHRS), the turn controller, or by the VOR or Localiser navigation receivers, a corresponding roll attitude error signal is determined by summing the values of the commanded roll attitude signal and the signal corresponding to the exisiting attitude; the latter signal is sensed by the roll sensing element of the vertical reference unit. Summation takes place in this case at point 'A' and the error signal produced is fed into the servo-control loop at point 'B'. Also at this point, a signal from a roll rate gyroscope sensing element is fed in to establish a turn rate for the command, and to prevent over-controlling. This is an example of the use of displacement gyroscope and rate gyroscope in combination (see page 116). Since the commanded roll attitude requires the ailerons to be displaced from their existing position, a position comparison process is also necessary to compute the error signal for servo-actuator operation. This is effected at the summing point 'C' to which is supplied what may be termed an 'existing position' feedback signal. The purpose of this signal, which is supplied from a position synchro type transducer, is to correct for non-linearities between the servo-actuator and the aileron system. The resulting difference signal is then applied to a discriminator, or pre-amplifier module, which removes quadrature and harmonic voltages and also increases the gain of the servo-control loop, thereby improving its response to signals commanding only small changes in aileron position. At the output side of the discriminator, and at summing point 'D', a rate feedback signal from the servomotor tachogenerator, is injected to back-off the command signal and so provide damping and servo-loop stability. The combined signal is therefore the one required to drive the servo-actuator to the commanded position, but in order to do this the signal must pass through a further stage of amplification and, as in all types of flight control systems, this is accomplished by means of a servo-amplifier.

In the system considered, the servo-actuator is of the two-phase induction motor type, and the control signal is supplied to its variable-phase winding. The direction of motor rotation is governed by the phase angle between the current flowing in the variable-phase winding, and that in the fixed-phase winding, the motor rotating clockwise or anti-clockwise depending on whether the variable-phase lags or leads the fixed-phase which is supplied from the main power source. A capacitor is inserted across the input to the servo-actuator to provide phase adjustment to the variable-phase current and this will always maintain a phase angle (whether lagging or leading) relative to the fixed-phase, thus maintaining

Clutch

Servo
position
synchro

Servo
motor

Tacho-
generator

Servo
position
synchro

Position
feedback

Gain

Rate
feedback

Gain

Servo

Discriminator

Gain

'D'

'C'

Gain

Position
feedback

Synchroniser

Cross-
feed

Gain

Bank
angle
limiter

Bank demand
rate limiter

Integrator

Clutched heading

Pre-set heading

VOR

LOC

'B'

'A'

Gain

Gain

Mode
selector
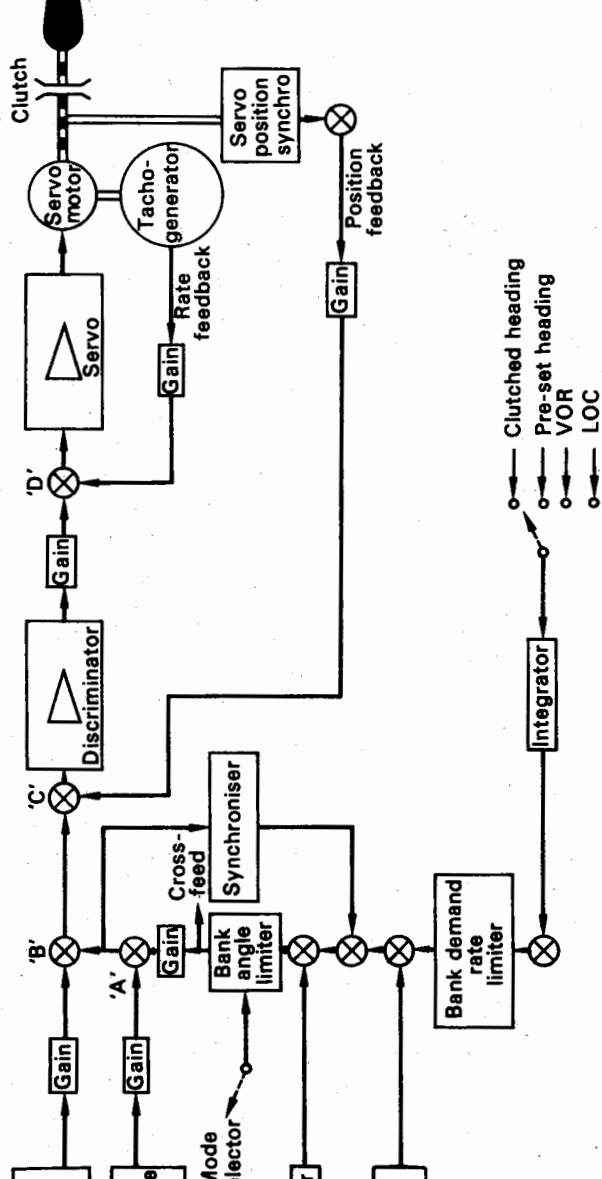
signal processing.
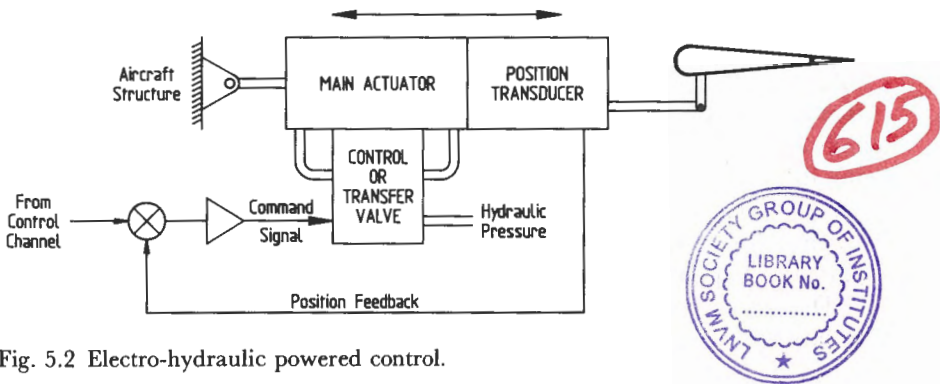
Fig. 5.2 Electro-hydraulic powered control.

rotation of the motor in the direction commanded by the original input signal.

When the difference between commanded and existing attitude is zero, there is then no further input to the servo-amplifier, the servomotor ceases to rotate, and the control surfaces (ailerons in this case) take up the position required to satisfy the commanded attitude change. Oscillations and 'hunting' of the servo-actuator motor about the zero signal point are prevented by supplying rate feedback signals from a tachogenerator (see page 158).

In an aircraft that utilises a powered flight control system, the command signal from a servo-amplifier is used for the positioning of a control or transfer valve in a power control unit, and as may be seen from fig. 5.2, this obviates the need for a separate motor-driven servo-actuator. The operation of this type of system is further described in chapter 7.

Referring to fig. 5.1 it will be noted that a servo-control channel is also dependent on other elements, the signals from which are utilised for the improvement of system response to commanded attitude changes and ultimate control capability. The functions of these elements, and brief details of their operating fundamentals are given in the following paragraphs.

Limiting

Under automatically controlled flight conditions, it is necessary to monitor what is generally termed the 'authority' of the control system (see also page 280) notably in respect of the roll and pitch channels; in other words, limits must be placed on commanded control signals to prevent excessive attitude changes and harsh manoeuvring.

In the example of roll control channel shown in fig. 5.1, it will be noted that there are two limiting elements in the signal processing chain: a roll command rate limiter, and a roll command limiter. The rate limiter limits

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the rate of change of command signals to some selected value, e.g. 5 degrees/second, so as to 'soften' aileron displacements and prevent harsh rolling of the aircraft. Signal processing through the limiter network is such that it imposes a specific time constant on the roll command signal. The roll command limiter controls the roll angle authority of the control system, the limit which the circuit is actually capable of being dependent on the control mode selected. Limits are accurately pre-set, and when the control channel is operating in the appropriate mode they are controlled by a d.c. bias signal applied to limiting diodes within the limiter module.

The roll command rate limiter consists of a limiting demodulator, a simple d.c. integrator, and a 400 Hz modulator stage with overall unity feedback. As already mentioned, it imposes lag on the roll command signal with a specific time constant; in this example the constant is about 10 seconds.

The output from the limiter module is summed with signals from the appropriate outer loop elements, and fed into the servo-control signal chain at summing points 'A' and 'B'. In addition, the limiter output is crossfed to the rudder control channel to provide co-ordination of the turn resulting from the displacement of the ailerons (see also page 121).

Synchronising

The necessity for synchronising has already been explained in chapter 2, (page 104) and as was also noted, the manner in which it is effected is dependent on the type of AFCS and the signal processing circuit arrangements adopted.

In the case of certain basic types of system, it is usual for the control panel pitch or roll controls, as appropriate, to be operated, thereby injecting a command signal into the appropriate amplifier channel, such that it will oppose any standing signal from an attitude signal transducer and reduce it to zero before engaging the control system. In the majority of systems however, it is effected by specifically designed synchronising circuits which automatically sense standing signals and oppose, or 'wash out' any resulting error signals.

An example of a circuit as applied to a roll control channel is shown in fig. 5.3 and is one which utilises the characteristics of both an operating amplifier in the integrating function, and of servomechanism feedback. Assuming that prior to engagement of the AFCS, the aircraft is at some angle of roll there will be a corresponding roll attitude signal output from the roll sensing transducer of the vertical-axis gyroscope unit, and this is supplied to summing point 2. All other command inputs normally supplied to summing junction 1 are at zero, since the control system is not at this stage coupled to any mode. The roll attitude signal is therefore, an error signal which flows out from summing junction 2 to summing junction 3 via
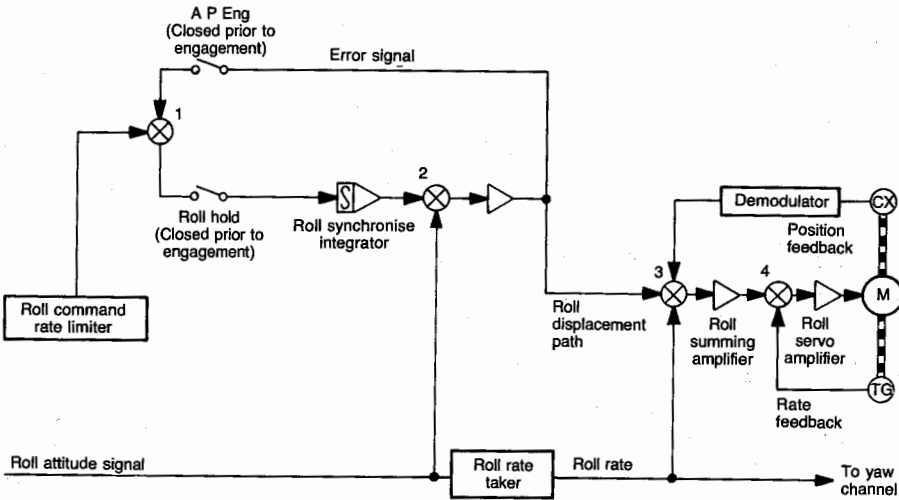
Fig. 5.3 Synchronising.

an amplifier and the roll displacement path, and also back to summing junction 1. Since the AP ENG and ROLL HOLD switches are both closed prior to engagement of the control system, then the error signal is also applied to the input of the integrating amplifier so as to drive it until its output to summing junction 2 is equal to the roll attitude signal, thereby zeroing the output from junction 2.

During the time that the foregoing synchronising process is taking place, an error signal is also produced at the output of summing junction 3 as a result of summing the signal along the roll displacement path, with a roll rate signal developed from the roll attitude signal after passing it through a 'rate taker' circuit; the purpose of the rate signal is to provide a short-term damping of servo-actuator operation. The resulting error signal passes to the aileron servo-actuator, via the summing and servo amplifiers, and so it is driven in the appropriate direction; the ailerons are not displaced of course, since with the control system not engaged the servomotor clutch is de-energised. At the same time that the motor operates, it drives a position feedback synchro (CX), the output of which is demodulated and fed to summing junction 3 to oppose the error signal which, as a result of the synchronising process, becomes less than the feedback signal, leaving the latter as the sole means of driving the servomotor until the feedback signal itself is reduced to zero.

The servo-actuator also drives a tachogenerator which supplies a rate feedback signal to summing junction 4, the purpose of this signal being to stabilise servomotor operation and to prevent any tendency for it to over-shoot its nulled position. Thus, in the synchronised condition, the net signals at summing junctions 2 through 4 are zero, and since the ser-

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vomotor is stopped at a position synchronised with the datum attitude detected by the vertical-axis gyroscope unit, it can be engaged with the aileron control system without snatching.

On engagement, the AP ENG and ROLL HOLD switches are opened and so the integrator is isolated from the circuit. Since the aircraft is still at some angle of roll, the appropriate roll attitude signal is now predominant, and from summing junction 3 is able to drive the synchronised servo-actuator motor which thereby displaces the ailerons to restore a wings-level attitude.

Demodulation

In the processing of command signals, it is necessary at certain stages of appropriate control channels to convert those signals having an alternating waveform into signals of unidirectional waveform; furthermore, the changes of phase polarity must also be detected in order to obtain comparable waveform characteristics. Such a conversion process is known as demodulation and the circuit element performing the conversion is called a demodulator. A typical example of its application is shown in fig. 5.3. In this case, the a.c. signal from the position feedback synchro is supplied as an input to the demodulator. A reference a.c. signal is also supplied to the demodulator so that the phase sense of the feedback signal may be detected. Both signals are passed through a transformer, full-wave diode, and resistance network, and as a result of comparison between input and reference signals, a d.c. output of the required polarity and amplitude is supplied to summing junction 3.

Gain and gain programming

The rates at which different types of aircraft respond to displacements of their flight control surfaces vary between types and their basic handling characteristics. In particular they vary with altitude, speed, aircraft load and configuration, and rate of manoeuvre. Thus, it is necessary to incorporate 'gearing' elements within flight control systems which will adapt them to aircraft and thereby reduce the effects which variations in flight parameters can have on handling characteristics. Similarly, in applying particular types of AFCS to individual aircraft control systems, it is necessary to provide facilities for altering the response of an automatic system to any given level of input signal, thereby obtaining a signal ratio best suited to the operation of the systems when working in combination. Such a ratio is known as 'gain' and may be considered as having a function analogous to the changing of gear ratios in a mechanical gearing system (see page 33).

Figure 5.4 shows a closed-loop control system in simple form. The signal
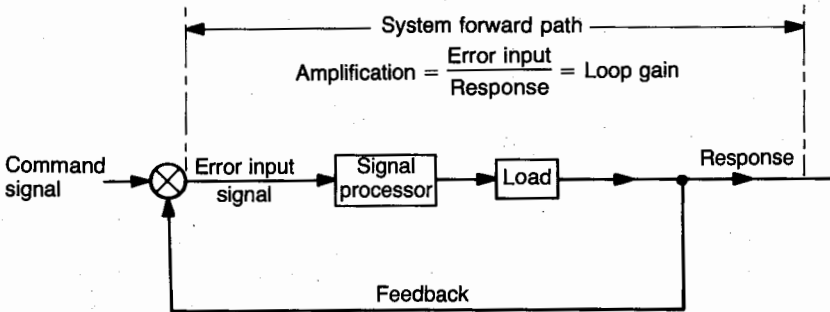
Fig. 5.4 Simple closed loop system.

path from error to response is known as the system forward path, and the amplification from error to response, measured as amplification ratio, is the loop gain. Within limits, increased gain improves performance in two ways: (i) residual error in steady state is reduced and so improves long-term accuracy, (ii) initial response to a given command is more rapid. The limit on these improvements arises from the need for adequate dynamic stability of the system. If, for example, loop gain is increased to some excessive value then dynamic instability would be produced so that response is grossly oscillatory and never settles to a steady state. Even before instability is reached, excessive loop gain reduces dynamic stability such that it would take too long for response to settle at a steady state, and furthermore, it would initially overshoot and then 'hunt' about the steady-state value.

Satisfactory closed-loop performance depends on determining a loop gain which compromises between long-term accuracy plus initial response, and acceptable settling time plus limited overshoot. These factors, in turn, require sufficient inherent damping in the load.

Certain adjustments of command and feedback signals can be pre-set within amplifier and/or computer units in order to produce gain factors which establish a basic 'match' between an AFCS and aircraft characteristics. Adjustments are based on the variation of electrical resistance at appropriate sections of signal circuits, and as in several types of control system, this is accomplished by means of potentiometers located on a calibration panel that forms an integral part of an amplifier or computer unit.

In addition to the foregoing adjustments, it is also necessary, particularly when a control system is operating in any of the outer loop control modes, for gain factors to be altered automatically to offset variations in handling characteristics resulting from changing flight conditions. This process is called 'gain programming or scheduling', and is part of a technique referred to as adaptive control. Some examples of programming the gain of servo

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control loops in response to airspeed changes have already been illustrated (see pages 123 and 124).

Another example of gain programming relates to an approach to an airport runway when the AFCS is coupled to the instrument landing system in the localiser (LOC) and glide slope (GS) modes (see also page 183). The purpose of gain programming in this case is to reduce the gain of beam deviation signals and thereby allow for convergence of the LOC and GS beams.

The results of a programme based on a representative aircraft/AFCS combination are shown graphically in fig. 5.5. The programme is accomplished by using signals from a low range radio altimeter which are supplied to a gain programmer control section in a vertical path module of the pitch control channel. After the GS mode is engaged plus 10 seconds, the GS deviation beam signal is modulated and amplified, and supplied as a pitch down command signal. Initially, the gain of the beam deviation amplifier is zero, but it then increases to, and is held at, 100% until the aircraft descends to 1,500 ft radio altitude. The altitude signal then produced by the radio altimeter develops a bias voltage which is applied to the GS beam deviation amplifier so that its gain is reduced as the aircraft descends (graph 'A'). The gain programmer control section of the vertical path module also supplies a gain control signal to a lateral path module in the roll control channel in order that the gain of the LOC beam deviation amplifier may also be reduced. In this case, the reduction is gradual from 100% to 57%.

In the event that there is an invalid signal from the radio altimeter, a time programme control is developed (see graph 'B'). In respect of the GS, this programme starts 10 seconds after GS has been engaged, and results in an initial gain of 80%, decreasing to approximately 20% over a period of time of 120 seconds. The LOC beam deviation amplifier gain programme is initiated directly at GS engage, and after 120 seconds time period the gain is reduced to 57%.

Figure 5.6 illustrates what is termed a self-adaptive control system and it is one which is capable of changing its parameters throughout an internal process of measurement, evaluation, and gain adjustments, without direct sensing of changing flight characteristics. The overall response of the system is optimised, irrespective of flight conditions, by means of an electrical analogue system referred to as a 'model reference'. The model defines the optimum dynamic behaviour of the aircraft, based on selected response characteristics subject to any constraints that the airframe may impose, and the control system parameters are adjusted to match the response of the aircraft to that of the model.

One form of model reference system is shown schematically in fig. 5.6 a. Command signals are fed through the model to the system, and the actual response tends towards that defined by the model as loop gain is increased; the error signal is amplified and fed to the control servo system. The system

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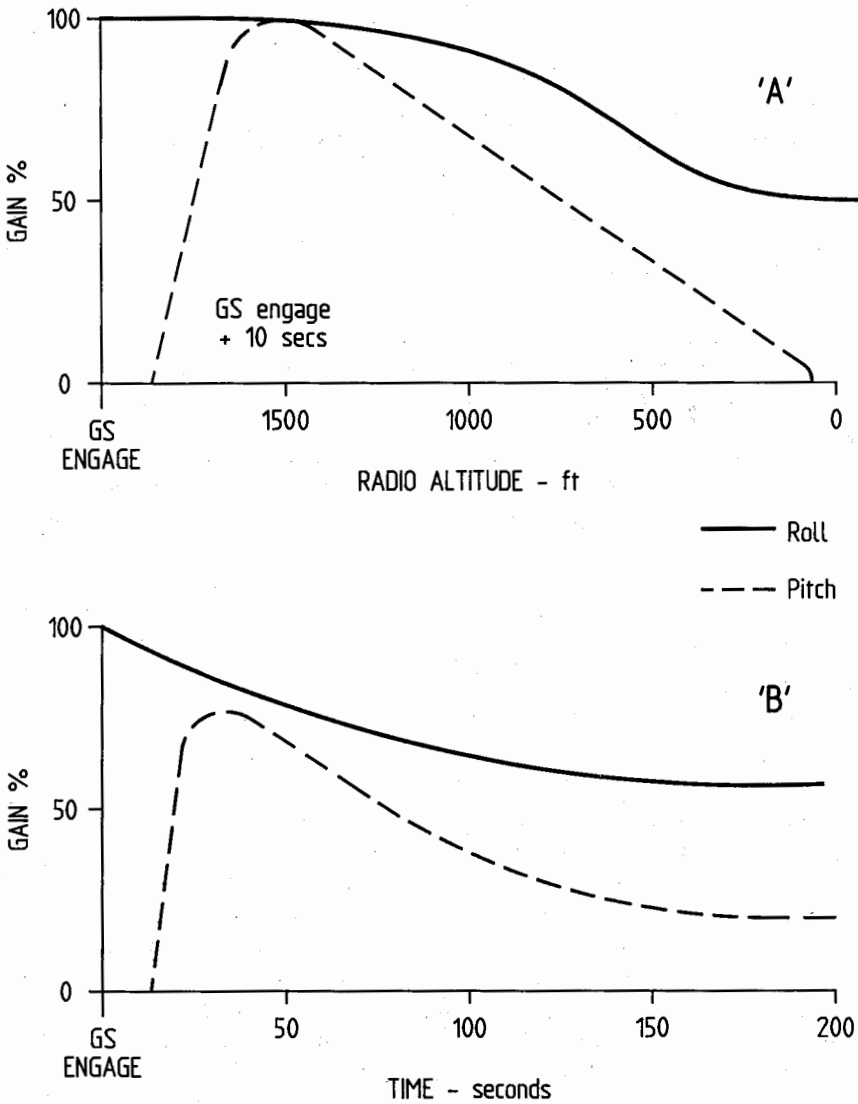
Fig. 5.5 Gain programme – ILS coupled approach.

overall is referred to as a high loop gain system since in order to optimise response relative to the model, maximum gain is required. Fig. 5.6 *b* illustrates an alternative form of model-reference system, in which command signals are fed to both the flight control system and the model. The output from the model is compared with actual response, and the external error so obtained is used to adjust the loop gain, in either sense, to minimise any discrepancy.

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(a)

(b)
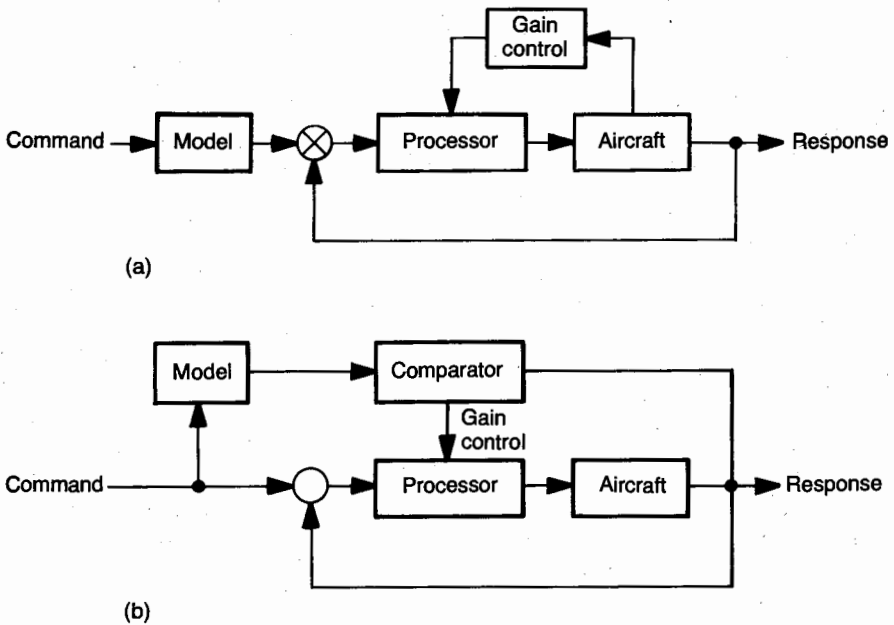
Fig. 5.6 Self-adaptive control system.

Feedback

When a control command signal is supplied to a servo control loop, then in accordance with closed-loop servomechanism principles, the operation of the associated servo-actuator or power contol unit (PCU) must be so controlled that correction proportional to the command signal input will be applied. This requirement is satisfied, as we have already seen from several examples, by means of _position feedback_ signals produced by sensors or transducers directly controlled by a servo-actuator or PCU. In some cases, transducers may be controlled by the relevant flight control surfaces.

There is also a second requirement, and that is for the amount of applied control to be limited to prevent overshooting a commanded position, and also 'hunting' about that position. This is satisfied by applying _rate feedback_ signals produced by a tachogenerator.

In some types of AFC systems, the feedback signals are a part of the output from the servo-amplifier itself, and after passing through an inte- gration network they are algebraically summed with the amplifier input.

Integration

In addition to the establishment of a synchronised condition, there are other operating conditions of an AFCS which require the 'washing out'

of error signals resulting from sustained or recurring displacements from stabilised references; for example, under crosswind conditions while operating in the localiser or VOR mode (see pages 183 and 190). Since the resulting errors are time-related, integrators such as the operating amplifier referred to on page 152 are commonly applied to the signal processing circuits of the appropriate control channels. The time constant (in seconds) of an integrating circuit varies depending on the particular control channel to which it is applied, and on the associated outer loop control mode selected.

Reversing links

These are provided in some control systems for connection across the output of a servo amplifier to permit phase reversal when, on account of the servo-actuator mounting or capstan installed in individual aircraft, it is necessary for the direction of servo-actuator rotation to be reversed to provide the required control surface displacement.

Computation of signals

The signals required for the operation of electronically-controlled systems utilised in any one type of aircraft are, in the first instance, related to inputs that are of an analogue nature, i.e. they are continuously changing in value. Some examples of analogue inputs are: engine-operating pressures and temperatures, fuel quantity, heading change, and, in respect of an AFCS, they relate to altitude, airspeed and attitude changes of an aircraft. As far as the processing of these inputs, and the computation of associated command signals are concerned, this can be accomplished by means of analogue computers. These computers are of the electro-mechanical type in that they comprise synchronous transmission devices and servo-mechanisms, in addition to electronic circuits performing such functions as signal amplification, differentiation, integration, etc.

The accuracy of such computers is quite high, but for aircraft systems which require a greater capacity for data processing, and a faster means of transferring the corresponding output data signals, they are limited in their application. These requirements are essential for the operation of the current generation of aircraft in which there is much closer integration between systems in order to achieve greater automation of in-flight management tasks.

In order, therefore, to fulfil the foregoing data signal requirements, the signals corresponding to input data are converted from their changing, or analogue, format into a discrete digital-coded format. This format is based on what is termed the *binary notation*, i.e. a system in which numbers are represented by the two digits 0 and 1 (see also Appendix 4). These digits

represent respectively, 'logic low' and 'logic high' values, and so the changing or analogue signal inputs are converted into a series of discrete pulses.

The devices used for signal conversion are of the solid-state integrated logic circuit type, and are referred to as *analogue to-digital (A/D) converters*. Conversion can also be required from digital-to-analogue and so the device used is then referred to as a *D/A converter*. Both devices perform encoding and decoding functions. Converters, together with all other circuit modules performing the various signal processing functions, are contained within digital computers whose architecture is designed appropriate to specific systems of an aircraft.

Note: There are many textbooks on digital computing techniques currently available, and the reader is referred to these for study of basic operating principles.

Data transfer

An essential feature of in-flight operation is the transfer of data between the individual computer systems of an aircraft. For example, under automatically-controlled flight conditions, an AFCS operates in conjunction with radio navigation systems, an FDS, INS, ADC, and a flight management system (FMS), all of which involve an exchange of data in order to provide appropriate commands to the control system. When conventional electrical interconnecting techniques are used between the units of individual systems, then the extent of 'hard wiring' required is considerable, particularly as individual wires must transfer signals dedicated to each of the parameters being monitored. In order to maximise on the utilisation of digital computer-based systems, it therefore became necessary to adopt an alternative method whereby information exchange is effected by a network of single data busses within an aircraft; this is known as a *data highway*.

Each data bus consists of shielded and twisted pairs of wires, and the voltage difference between them encodes a binary 0 or binary 1. All outgoing encoded data from the computers are identified by an additional binary-coded word called a *label*. This takes up the first 8 binary digits ('bits') of each word and is octal coded, i.e. it is coded to the base 8.

The designation of labels to particular functions is arranged by an aircraft manufacturer in relation to each of the specific systems installed in the type of aircraft concerned, and in accordance with standard specifications. One such specification accepted as an air transport industry 'standards specification' for digital data transfer, is known as ARINC 429; ARINC being the abbreviated name of the US organisation 'Aeronautical Radio Incorporated'. As separate bus systems are predictable for the different classes of aircraft systems, ARINC 429 includes some duplication of labels where it is known that the use of a common label on the same bus for

two different purposes will occur. For example, label 315_8 defines 'wind shear' for navigational purposes, but for flight control systems, the same label defines 'stabiliser position'.

Systems providing data outputs (referred to as transmitters) each have their own data bus connecting them to the 'receiver' systems in need of the data, as shown in fig. 5.7. The shielding of the wires comprising each data bus is connected to ground and in particular, at each branch to receivers. The maximum number of receivers that can be connected to the same bus line is 20.

The digital computers of the different aircraft systems, process data in the form of specific messages or parallel binary words. The messages are converted and transmitted in serial form, the reason for this being that weight of transmission lines is reduced, and also reliability is improved. The serial messages are then adapted into high and low voltage levels, and transmitted along the data bus lines in the form of strings of pulses. These comprise the word strings of a message and correspond to those appropriate to all the systems detailed in the ARINC 429 specification. Each word is formed of 32 bits, each bit being either a binary 1 or a binary 0. As noted earlier, the first 8 bits comprise the label which identifies the source of the message, while the remaining bits are assigned to data, parity, sign and status or validity. Two examples of a message are also shown in fig. 5.7 one is labelled 'Distance Measuring Equipment (DME) distance' and the other 'Radio Altitude'.

In each case, bits 9 and 10 are assigned to what is termed a source destination identifier (SDI); this applies when specific words need to be directed to a specific system of a multi-system installation, or when the source system needs to be recognised from the word content. In the examples indicated the systems are, of course, the DME and radio altimeter respectively.

Bits 11 to 29 are those assigned to the actual data being transmitted which, in the case of our examples, are distance in nautical miles, and radio altitude in feet. The groups of binary 1's and 0's (in bits 11 to 29) correspond to equivalent decimal numbers, which indicate that the DME system computer is transmitting encoded data corresponding to a distance of 257.86 nautical miles, while the radio altimeter computer is transmitting data corresponding to an altitude of 2450.5 feet.

Bits 30 and 31 are assigned to what is termed the sign/status matrix (SSM) which refers to plus, minus, north, south, left, right, etc., of binary-coded decimal numeric data. They also refer to the validity of data, and failure warning.

The detection of errors in codes and their correction is a very important aspect in the transmission of digital data, and for this purpose a *parity check* method is provided whereby a computer can test whether bits in a binary word have been accidentally changed during transmission. The test is done by automatic summation of the bits comprising a word to determine whether the total number is odd or even, and by calculating what is termed

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TO OTHER
RECEIVERS

Parallel/Serial
Conversion
(shift register)

Adapter/line
transmitter

Hi
A
B
Lo

Level
adapting &
converting
serial to
parallel

Parallel word

Serial message

TRANSMITTER

RECEIVER

A
B

$V_A - V_B$

+10V

0

−10V

BIT1

BIT1

BIT0

NOTHING
TRANSMITTED

| 1st BIT | 2nd BIT | 3rd BIT |

| 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 | 9 | 10 | 11 | 12 | BIT NUMBER |

BIPOLAR

| 1 | 0 | | 1 | 0 | 1 | 0 | 1 | 0 | 0 | 1 | 1 | DATA |

DATA BIT ENCODING LOGIC

32	31	30	29	28	27	26	25	24	23	22	21	20	19	18	17	16	15	14	13	12	11	10	9	8	7	6	5	4	3	2	1
0	0	0	0	0	1	0	0	1	0	1	0	1	1	1	1	1	0	0	0	0	1	1	0	0	0	1	0	0	0	0	1

SSM MSB

LSB

SDI Label DME
distance

Parity

Data = 257.86 nautical miles

32	31	30	29	28	27	26	25	24	23	22	21	20	19	18	17	16	15	14	13	12	11	10	9	8	7	6	5	4	3	2	1
0	0	0	0	0	1	0	0	1	0	0	0	0	1	0	1	0	0	0	0	0	1	0	1	0	0	1	0	1	0	1	0

MSB

LSB

Label
radio altitude

Data = 2450.5 ft

Fig. 5.7 Data transfer.
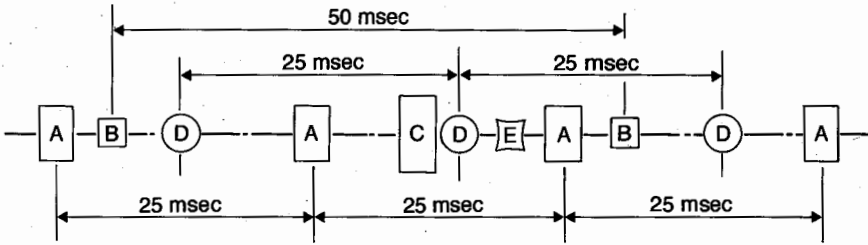
Fig. 5.8

a *parity bit*; this forms the last bit of a word i.e., bit 32. If, for example, there is an odd number of binary 1's among the first 31 bits, the parity bit is set to '1' to make the word of 'even parity'. 'Odd parity' can also be used where the parity bit is set to binary 0 to make the total number of binary 1's odd. This latter form of parity is adopted in the ARINC 429 specification.

Data are transmitted in batches at a specified repetition or refreshment rate along the appropriate busses, and at either high speed (100 kilobits per second) or low speed (12 to 14.5 kilobits per second) according to the frequency at which interfacing systems require an update of information. This is shown in fig. 5.8.

A standards specification of comparatively recent origin is the ARINC 629. It relates to a data bus system called DATAC (Digital Autonomous Terminal Access Communications) conceived by Boeing for use in the B777. Unlike ARINC 429 it is a two-way bus requiring fewer wires and having a very much faster data transfer rate.

Digital flight control computer

The architecture of a typical digital flight control computer (FCC) is illustrated in fig. 5.9. It processes the input signals from the various external interfacing systems to provide signals for the control of an aircraft about its pitch and roll axes.

The operation of this computer is centred around two central processor units (CPUs). CPU No. 1 performs most mode logic and control operations, while CPU No. 2 performs parallel computations for dual-channel autoland functions, and also provides pitch inner loop and limiting functions. In order to preclude a failure of a CPU from causing a simultaneous pitch and roll hardover situation of the automatic flight control system, computation functions are split between the two CPUs. For critical applications, signal and logic computations are provided by both units. All

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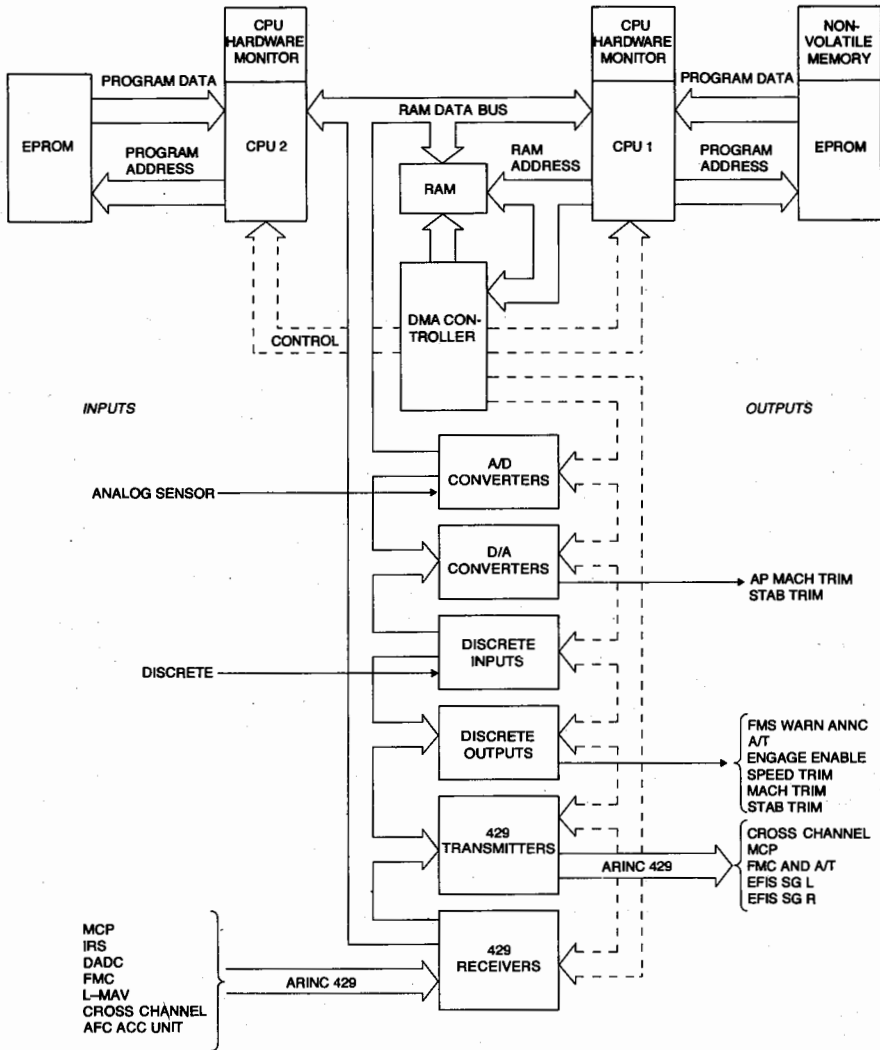
Fig. 5.9

signals to and from the FCC, as well as data transferred between CPUs, are controlled by a direct memory access (DMA) controller.

Inputs, valid data signals, and existing logic states are combined in software whose purpose is to compute current operating mode states. These states are of single purpose or, in computer terminology, they are discrete; and they are stored as logic vectors for subsequent reference by control laws and executive software. They are then supplied as discrete outputs to such interfacing systems and units as flight management, autothrottle, flight mode annunciator, and mode control panel of the autoflight control system.

The logic states are interrogated by software which selects and executes the routines appropriate to the operating modes and sub-modes of the control system. The software also generates autoflight commands for output to the relevant flight control surface actuators, and also to the command bars of an electronic flight instrument system (EFIS) attitude director indicator (see page 268).

As will be noted from fig. 5.9, there are three forms of input to the computer, namely, analogue, discrete, and ARINC 429. Analogue inputs are first converted to digital format, discrete inputs are 'latched' to the FCC, and the ARINC 429 digital words are received by ARINC 429 receivers. All the information is stored in a random access memory (RAM) which is addressed by the DMA controller. Accessing of particular items of information, as and when called for by the software program, is carried out by the CPUs. These two units provide various outputs based on control law computation, and the information is transmitted to respective output conditioning circuits under the control of the DMA. After conditioning of the output signals, they are transmitted to the appropriate interfacing control units and systems, in the form of analogue signals, discretes, or ARINC 429 digital data words.

Both CPUs continuously execute periodic self-test routines to detect failures in the processors, the RAM, the erasible programmable read only memory (EPROM) or the positioning of a servo-actuator of the autoflight control system. During autoland and/or go-around modes, the units also compare elevator and aileron position with that of the servo-actuators, and the existing autoflight commands with a model output. If any unacceptable discrepancies exist, the autoflight control system is automatically disengaged. Failures detected by the self-test routines, or by CPU redundancy monitoring, are recorded by built-in-test equipment (BITE) for maintenance inspection purposes.

TEST QUESTIONS

1 In connection with command signal processing, the function of adapting control system response to an aircraft's handling qualities, is known as:
   (a) limiting.
   (b) integrating.
   (c) gain programming.
2 In order to derive a d.c. control signal having the same phase relationship as an a.c. error signal, it is necessary for signals to be:
   (a) differentiated.
   (b) demodulated.
   (c) modulated.
3 What methods are employed to prevent an automatic control system

from applying excessive attitude changes or harsh manoeuvring of any aircraft?

4 State the function of a synchronising circuit and briefly explain its operation.

5 What is meant by 'the gain of a control system'?

6 What do you understand by the term 'gain programming'?

7 Why is a gain programme necessary when operating in the 'GS' and 'LOC' modes?

8 From which source is a signal required to initiate the gain programme during an ILS-coupled approach?

9 For what purpose would a gain programme based on changes in airspeed be used?

10 Why is it necessary for a demodulator to be supplied with an a.c. reference signal?

11 In what type of flight control system are servomotors not required?

12 Why is feedback necessary in command signal processing channels?

13 State the two types of feedback signals normally required, and how they are produced.

14 Is it possible for feedback signals to be produced without the use of transducers?

15 How are analogue data converted into binary-coded format?

16 In the ARINC 429 form of data transfer, how are the data identified according to function?

17 How are any errors in the codes used in the transmission of data detected?

6
Outer Loop Control

In addition to performing the primary function of stabilisation an AFCS can also be developed to perform the tasks of modifying the stabilised attitude of an aircraft by computing the necessary manoeuvres from inputs of such raw data as airspeed, altitude, magnetic heading, interception of radio beams from ground-based aids, etc. Such data inputs constitute outer loop control (see fig. 6.1), the number of inputs serving as an indication of the progressive development of automatic flight from the basic single-axis wing-levelling type of autopilot to the highly sophisticated flight guidance systems now used in many present-day transport aircraft. The provision of raw data inputs relevant to a particular flight path is referred to as 'coupling' or as a 'mode of operation', the selection of each mode being made by the pilot via appropriate control panel switching devices. Other terms commonly used in connection with operating modes are: 'hold', 'lock' and 'capture'; for example, an aircraft flying automatically at a selected altitude is said to be in the 'altitude hold' or 'height lock' mode. The term 'capture' relates principally to modes associated with the selection and interception of beams from ground-based radio navigation aids; for example, 'glide slope capture'.

In some cases, mode switching is automatic; thus, to switch from intercepting a beam or a heading to tracking the beam on reaching it, a beam sensor is installed. This device senses beam deviation and switches modes automatically when the aircraft flies into the beam. Glide slope capture can also take place automatically, in this case the pitch control channel is switched from 'altitude hold' mode to glide slope track when the aircraft flies into the glide slope beam.

The raw data is supplied from sensors which convert the data into appropriate electrical signals that can be mixed in with inner control loop signal data to produce the changes to the aircraft's flight path. The traditional raw data instrument displays are used by the pilot for monitoring, and programming management. Outer loop control modes which may be incorporated in a control system are listed in the table on page 169. The number of modes actually adopted depends, of course, on the aircraft/control system combination; for example, in a single-engined light

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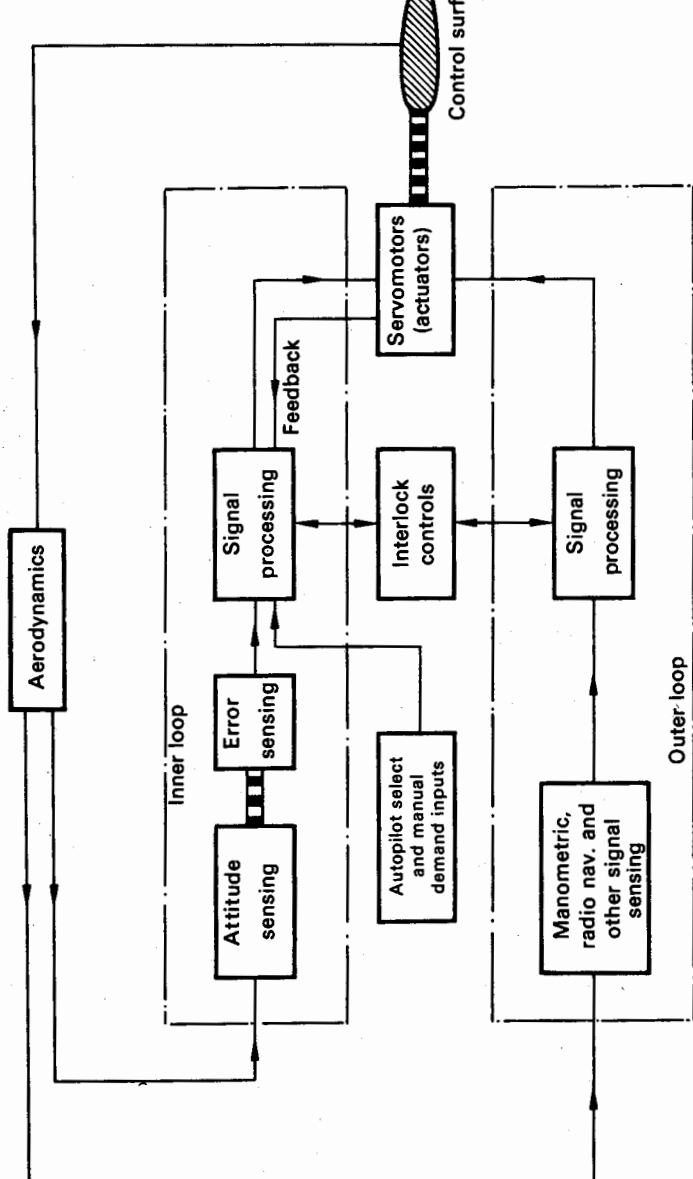
Inner loop stabilisation and outer loop control.

aircraft having a basic wing-levelling control system, only altitude and heading modes might comprise the outer loop control, whereas in a more complex type of transport aircraft using a flight guidance system, and having automatic landing capability, the outer loop could comprise all the modes listed in the table.

Pitch axis	Roll axis
Manometric or air data:	Heading select and hold
Altitude select and hold	
Vertical speed	Bank hold
Airspeed select and hold	
Mach hold	Radio navigation:
	VOR
Pitch hold	Back beam
Pitch trim	Area navigation:
	Doppler
Turbulence penetration	Inertial
Vertical navigation	
Instrument Landing System	
Glide slope	Localiser
Autoland	
Approach	Runway align
Flare	Roll out
	Control Wheel Steering
	Touch Control Steering

Manometric or air data

Raw data inputs which come under this heading are those associated with altitude, airspeed, vertical speed, and speed in terms of Mach number, each providing outer loop control about the pitch axis of an aircraft. Sensing may be carried out either by independent sensor units, or by an air data computer (ADC). The sensors operate on the same fundamental principles as the basic pitot-static flight instruments, the measuring elements being coupled to appropriate types of electrical pick-off elements in lieu of indicating pointer mechanisms.

Altitude hold

As we learned from chapter 4, any changes of aircraft attitude about its pitch axis while in straight and level flight, will be detected by the pitch

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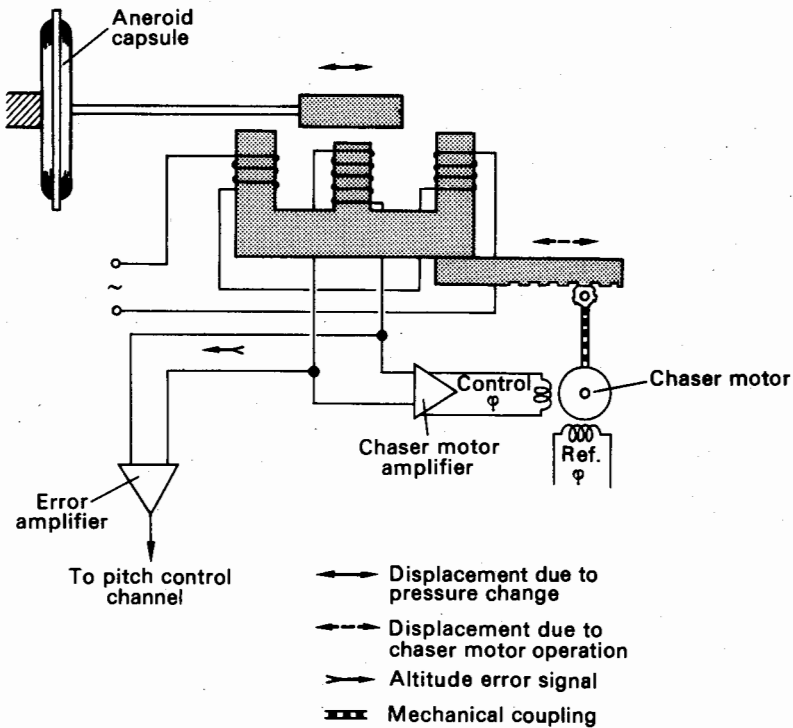
Fig. 6.2 Altitude-hold sensor.

attitude sensing elements of the AFCS, and the changes will be accordingly corrected. However, in the event that the changes are associated with a pure vertical displacement of the aircraft, detection and corrections might still be effected, but straight and level flight could possibly be resumed at some altitude above or below that at which flight is required. In other words, an attitude sensing element alone cannot detect an altitude change, and neither can it maintain a required altitude. In order to meet this requirement, and also to provide for automatic 'levelling off' at any desired altitude, an altitude hold, or lock sensor, is employed. Sensors vary in construction, particularly in connection with the type of pick-off element adopted for a specific flight control system. The fundamentals of the operating requirement may, however, be understood by considering the example illustrated in fig. 6.2.

The sensor consists of a pressure transducer comprising an evacuated capsule assembly, and E and I type of inductive pick-off element (see page 130) an amplifier, and a two-phase induction type of chaser motor. The capsule assembly is subjected to changes of static pressure supplied to the case of the sensor unit from the aircraft's static pressure system, and is mechanically linked to the I bar of the pick-off element. A change of

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altitude produces a change of static pressure to cause the capsule assembly to expand or close up; this, in turn, displaces the I bar and a signal is induced in the coil of the centre limb of the E bar; the signal being a measure of the direction and rate of altitude change.

With the altitude hold mode unselected, the induced signal is fed to the chaser motor which drives the E bar in the same direction as the I bar, and so reduces the signal to zero. When the mode is selected, the pick-off will be in the zero signal condition at the prevailing altitude of the aircraft; this condition thus becomes the datum from which altitude changes may be detected. When a change does occur, the capsule assembly displaces the I bar which induces a proportional signal in the centre limb coil winding, the signal being analogous to an altitude error. The signal, which cannot now cause operation of the chaser motor, is applied to a separate error amplifier, and finally to the pitch channel servo amplifier. The elevator servo-actuator is thus operated to apply elevator control and so restore the aircraft to the selected altitude. At the same time, the change in static pressure will cause the capsule assembly to displace the I bar in the opposite direction to reduce the error signal to zero once again.

In AFC systems which use digital computation circuits for the processing of signals, altitude hold is effected in a more sophisticated manner. In a typical system, the altitude hold signal is derived from an inertial reference system (see page 190). This signal which is termed inertial altitude, is combined with a barometric pressure-corrected altitude signal measured by the altitude sensor in an ADC which interfaces with the AFCS. This latter signal corresponds to the present altitude of the aircraft, and is passed through a sampling and hold circuit in the flight control computer. When altitude hold mode is entered, the signal is held as a reference against which the difference, or distance, between the aircraft's altitude at the time, and the new altitude selected for capture can be computed.

The computation is made as a function of the distance between altitudes and the vertical speed rate necessary to attain a selected altitude. The command signals generated as a result of this computation are supplied to the elevator servo-actuator, which produces such a pitch attitude change of the aircraft that an acceleration-limited exponential approach to the selected altitude is achieved. When this altitude is captured, commanded distance and rate signals are 'nulled', and the selected altitude is then sampled and this becomes the new reference against which further selections can be computed.

Airspeed hold

Since airspeed-hold sensors are used in conjunction with altitude-hold sensors, then in any one system design the methods of transmitting error signals are of a common nature; in fact, the only difference is that whereas an altitude sensor measures only static pressure changes, an airspeed sensor

is required to measure the difference between static and dynamic pressures. The capsule assembly, instead of being sealed, is open to the source of dynamic pressure, and static pressure is admitted to the sealed chamber in which the assembly is contained. Thus, the capsule assembly expands or closes up under the influence of a pressure differential created by a change of airspeed. The pick-off element actuated by the capsule assembly, is identical in construction, and operation, to that adopted in the altitude sensor, the speed error signal also being supplied to the pitch servo control channel.

The airspeed-hold sensor associated with the altitude-hold sensor shown in fig. 6.3 utilises an identical type of pick-off element, but instead of the error signal being transmitted to the pitch control channel by a CX synchro, transmission is by means of ganged potentiometers driven by the chaser motor. The capsule assembly is housed in the same chamber as the altitude capsule assembly.

Mach hold

In high-performance aircraft, the airspeed is measured in terms of Mach number, in addition to the conventional unit of speed, knots. There is, therefore, a requirement for both modes under automatically controlled flight conditions, the airspeed hold mode being most commonly used during the low-altitude cruise phase of flight, and Mach hold during the high-altitude phase. Since Mach number varies with airspeed and altitude, the signal outputs from the independent sensors can be integrated to provide the required Mach signal output. This is accomplished by incorporating all sensors in a unit called a central air data computer.

Air data computers

It is clear from the foregoing that the transmission of the raw data associated with airspeed, altitude, and Mach number is, primarily, the transmission of pressure from two sources; a static source and a dynamic source. In some types of aircraft, both sources are combined in a pitot-static probe, which is located in the airflow at an experimentally determined point. In the large majority of aircraft, however, the sources are independent; dynamic pressure is detected by a pitot pressure probe in the airflow, while static pressure is detected by a vent located at a point of undisturbed airflow, for example, at the side of a fuselage.

The pressures are transmitted to the primary flight instruments, i.e. airspeed indicator, altimeter, and vertical speed indicator, via pipelines the length and quantity of which will vary according to the size of aircraft, and the number of stations within an aircraft at which indications of the relevant parameters are required. In order, therefore, to minimise the 'pressure plumbing' arrangements, the idea of supplying the pressures to a

TMC
FCC
Stall
Stal
Yaw
Altit

Cap
Ins
IRU

EEC

F/O
Ins
EFIS
Fligh
GPW
EIC
FMC
Cab
Co
Yaw
Tran
IRU

1   2   3   4

SYNCHRO TO DIGITAL CONVERTER

BARO. CORRECTED ALTITUDE

ALTITUDE RATE CALCULATION

V/S

ALT.

Vmo CALCULATION

Vmo

CAS

MACH

AOA

TAS

SAT

TAT

ALTITUDE CALCULATION

COMPUTED AIRSPEED CALCULATION

MACH CALCULATION

TAS COMPUTATION

SSEC

A/C TYPE

CALIBRATION AND TEMPERATURE COMPEN-SATION

CORRECTED AOA

SAT COMPUTATION

A/C TYPE

FREQUENCY TO DIGITAL CONVERTER

SYNCHRO TO DIGITAL CONVERTER

TAT COMPUTATION

FREQUENCY MODULATED PRESSURE SENSOR

POWER SUPPLY MODULE

To all modules

CODER

GS

OR

Data Busses

EEC   Electronic Engine Control
EFIS   Electronic Flight Instrument System
EICNS Engine Indicating and Control System
FMC   Flight Management Computer

FCC   Flight Control Computer
GPWS   Ground Proximity Warning System
IRU   Inertial Reference Unit
TMC   Thrust Management Computer

central location from which they could be transmitted to any number of stations, and in the form of synchronous signal data links, was developed and resulted in the design of units designated as air data computers (ADC).

In common with the other units and systems, an ADC can vary in design and it is, therefore, beyond the scope of this book to go into specific operating details of the various computers in current use. However, the principal features of ADC design concept may be understood from fig. 6.3, which illustrates a computer of the digital type.

The sensing of pitot and static pressures is done by means of piezoelectric crystal type sensors, instead of by flexible metal capsule elements. The sensors consist of quartz discs with a metallic pattern deposited on them, and arranged in a thin stack such that they serve as a flexible diaphragm. Thus, when subjected to pressure changes, the resultant flexing sets up an electrical polarisation in the discs so that electrical charges are produced (this is termed the 'piezoelectric effect'). The polarity of the charges depends on the direction of flexing, in other words, whether the pressures applied are increasing or decreasing. The resulting signals are frequency-modulated and are then supplied to altitude, computed airspeed and Mach number calculation circuit modules via a frequency-to-digital converter.

Inputs from an outside air temperature probe are also supplied to the computer for the purpose of computing static air temperature (SAT) total air temperature (TAT) and true airspeed (TAS). Outputs from all modules of the computer are supplied to an ARINC 429 transmitter connected to four data busses from which all interfacing systems requiring 'air data' are then supplied. In order for the computer to automatically take into account the pressure error of the pitot–static system of a particular type of aircraft, and also its stall characteristics, it is 'matched' by programming the static source error correction (SSEC) and the angle of attack modules with the relevant data.

The purpose of the discretes coder module is to monitor signals relating to the status and integrity of particular circuits, e.g. the heater circuits of TAT probes, pitot probes, and angle-of-attack sensors, and to initiate appropriate warnings.

Vertical speed selection and hold

This mode is incorporated in a number of AFC systems, its purpose being to enable the flight crew to select a particular rate of climb as vertical speed (V/S) which, on engagement, will be automatically maintained. The mode select circuit arrangements, which can vary between systems, normally forms part of a flight or mode control panel. In order to understand the operating fundamentals we may consider the typical arrangement shown in fig. 6.4.

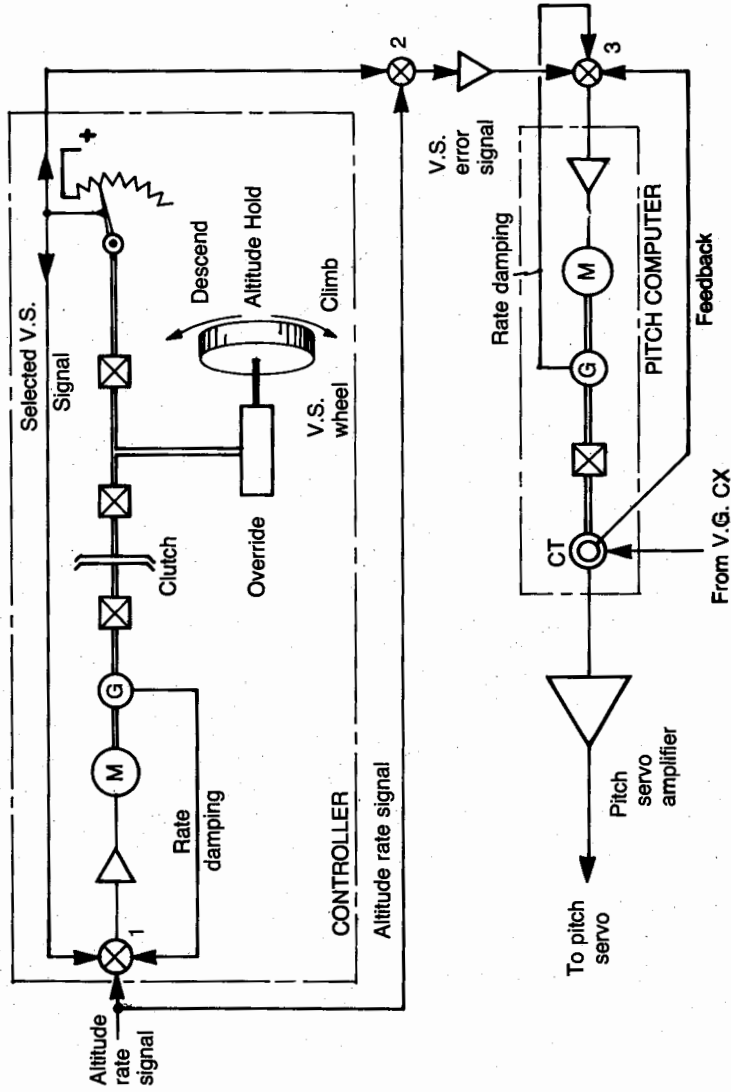Firstly, let us take the case of an aircraft in level flight and under

Fig. 6.4 Vertical speed selection and hold.

automatic control. The aircraft's ADC with which the V/S section of the control panel is interfaced, will not be supplying any altitude rate signals, and so the V/S motor and associated circuit will be in a static condition. Similarly, there will be no pitch attitude error signals from the attitude detector sensing element to the CT synchro in the V/S section of the pitch computer. The CT is therefore, at 'null' and so this section of the computer is also in a static condition.

Assume now that the aircraft is to climb to a new flight level at a specific V/S rate. The V/S mode is selected and this engages the clutch to connect the V/S motor to the differential gearing within the control panel (the two sets of gearing permit independent rotation of the motor, V/S selector wheel, and the potentiometer wiper). At the moment of V/S mode selection, the selector wheel is at a detent position and so there is no signal output from the potentiometer.

When the selector wheel is rotated in the climb direction to the desired V/S rate (indicated on the sides of the wheel) the potentiometer wiper is also rotated, thereby creating a corresponding V/S signal. This signal is supplied to summing junction 1, and is the dominant one because at this moment, there is no altitude rate signal from the ADC. After amplification, the signal drives the V/S motor so that it 'chases' in the direction of the selected V/S. The motor drives a tachogenerator whose output is fed back to summing junction 1 to provide rate damping of motor operation.

In addition to summing junction 1, the selected V/S signal is also supplied to junction 2, and again because there is no altitude rate signal from the ADC, the selected signal is dominant. After amplification, it is supplied to summing junction 3, and from there it is further amplified to operate a servomotor in the V/S section of the AFCS pitch computer. The motor drives the CT rotor from 'null' to induce an error signal output as a result of the 'standing' attitude signal from the pitch attitude sensor CX. The error signal is then supplied to the pitch channel servo via its amplifier to apply elevator control and to initiate the climb attitude of the aircraft. A tachogenerator provides rate damping signals to the V/S motor via summing junction 3.

In the climb attitude, the V/S section of the ADC produces a corresponding altitude rate of change signal and this is supplied to summing junction 1 to balance out the selected signal at which point the V/S motor in the control panel stops rotating. The altitude rate of change signal is also supplied to summing junction 2 to balance out the selected signal at this junction.

Simultaneous with ADC V/S section operation, the pitch attitude sensor will have detected the climb attitude of the aircraft, and will supply signals to the CT synchro to restore it to the 'null' position in the climb attitude.

If the aircraft's vertical speed (as detected by the ADC) should subsequently deviate from the selected speed, then an altitude rate signal will be applied to summing junction 2. After amplification, this V/S error

signal will then be supplied to the elevator servo via the V/S section of the pitch computer, and together with signals from the appropriate attitude sensing element, the selected V/S and corresponding climb attitude is restored.

In order to level off at the new flight level, the V/S wheel is rotated back to the detent position thereby repositioning the potentiometer wiper to take off the original selected V/S signal.

As already noted, this is only one example of V/S select and hold application; details of the differences between this and applications to other types of AFCS should therefore be obtained from the appropriate aircraft maintenance manual chapters.

Heading hold

As the name suggests, in this mode of operation the automatic flight control system holds the aircraft on a pre-selected magnetic heading. Since turning of an aircraft is carried out by displacement of the ailerons, the heading hold mode relates to control about the roll axis, and the heading error signals are applied to the roll control channel of the flight control system. The error signals may be derived in a variety of ways depending on the source of magnetic heading data provided in a particular type of aircraft, and on how such source may be utilised in AFCS operation.

In the case of some types of basic control systems utilising a directional gyro for the supply of heading data within the turn control loop, it is usual to provide a heading selector which may be a separate unit or incorporated in the pilot's control panel. The selector contains a heading dial which can be rotated by means of a setting knob so that it may be referenced to any heading indicated by the directional gyro. The dial is mechanically coupled to a CT synchro which develops a heading error signal voltage as a function of dial position. The heading set knob is linked to a switch, and when the knob is pushed in the switch connects the error signal voltage to a summing amplifier in the roll control channel of the autopilot. Thus, the aircraft will turn on to the selected heading, and when this is reached the error signal is 'nulled' and the aircraft is held on the selected heading.

In the majority of aircraft, it is usual for heading data to be supplied from either a magnetic heading reference system (MHRS), or from a flight director system (FDS) (see chapter 9).

In both the foregoing systems, heading data are obtained by direct sensing of the earth's magnetic field as opposed to directional gyro indications which must always be referenced to those of a magnetic compass. Sensing is effected by a flux detector unit located at some remote point in the aircraft, e.g. a wing tip, the detector forming the transmitter (CX) of a synchronous control system (see also page 134). In this application, the complete detector turns with the aircraft so that a resultant field is produced

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which is aligned with the earth's magnetic field. The signals induced in the transmitter coils are fed to the stator of a slaving CT synchro, the function of which is to monitor the azimuth position of a directional gyroscope via a slaving amplifier and torque motor. The gyroscope may be contained within a panel-mounted indicator, but more generally it is designed as a separate and remotely-located unit. The fundamental arrangement of such a system is illustrated in fig. 6.5.

On a constant heading, the synchro system is in the 'null' condition, the heading being indicated by a heading card referenced against a lubber mark. During a turn, the flux detector senses the changing heading, and the field in the slaving CT synchro will rotate and thereby induce a heading error voltage in the synchro rotor. The signal is then amplified by the slaving amplifier and fed to the torque motor which precesses the directional gyroscope in azimuth and, at the same time, causes the rotor of a servo CX synchro to rotate. The signals induced in the stator of the CX are applied to that of a servo CT synchro in the indicator and, because turning of the aircraft causes misalignment to exist between the rotors of both synchros, a servo-loop error voltage is induced in the CT synchro rotor. This voltage is then applied to a servo amplifier, and after amplication, it is applied to the reference winding of a servomotor which is mechanically coupled to the CT synchro rotor and to the heading card of the indicator. Thus, the rotor and heading card are rotated, the card indicating the direction of the heading change taking place.

On cessation of the turn, the rotor reaches a 'null' position, and as there is no further input to the servo-amplifier, the servo-actuator stops rotating. Oscillation of the servo system is prevented by velocity feedback damping signals from a tachogenerator which is also driven by the servo-actuator. The slaving CT synchro rotor is also coupled to the servomotor, the purpose of this being to drive the rotor in synchronism with the rotating field in the stator. Thus, during turns, the directional gyroscope is continuously slaved to the changes in magnetic heading, and by means of the servo synchro loop, the complete system is monitored such that when a turn ceases, the system is in a 'null' condition, and the indicator displays the corresponding heading.

In order to apply the foregoing principle of remote-heading indication to an AFCS as a heading-hold mode of operation, and as a means of automatically turning an aircraft on to a selected heading, it is necessary to provide additional data synchros in the compass system. In the example shown in fig. 6.5, the data synchro for heading hold is contained within the directional gyroscope unit, its rotor position being controlled by the gyroscope in the same way as the servo CX synchro. The data synchro stator is connected to the roll control channel of the flight control system via the mode selector circuit; the stator output therefore, provides the heading hold reference.

The signals for turning the aircraft onto any desired heading are derived
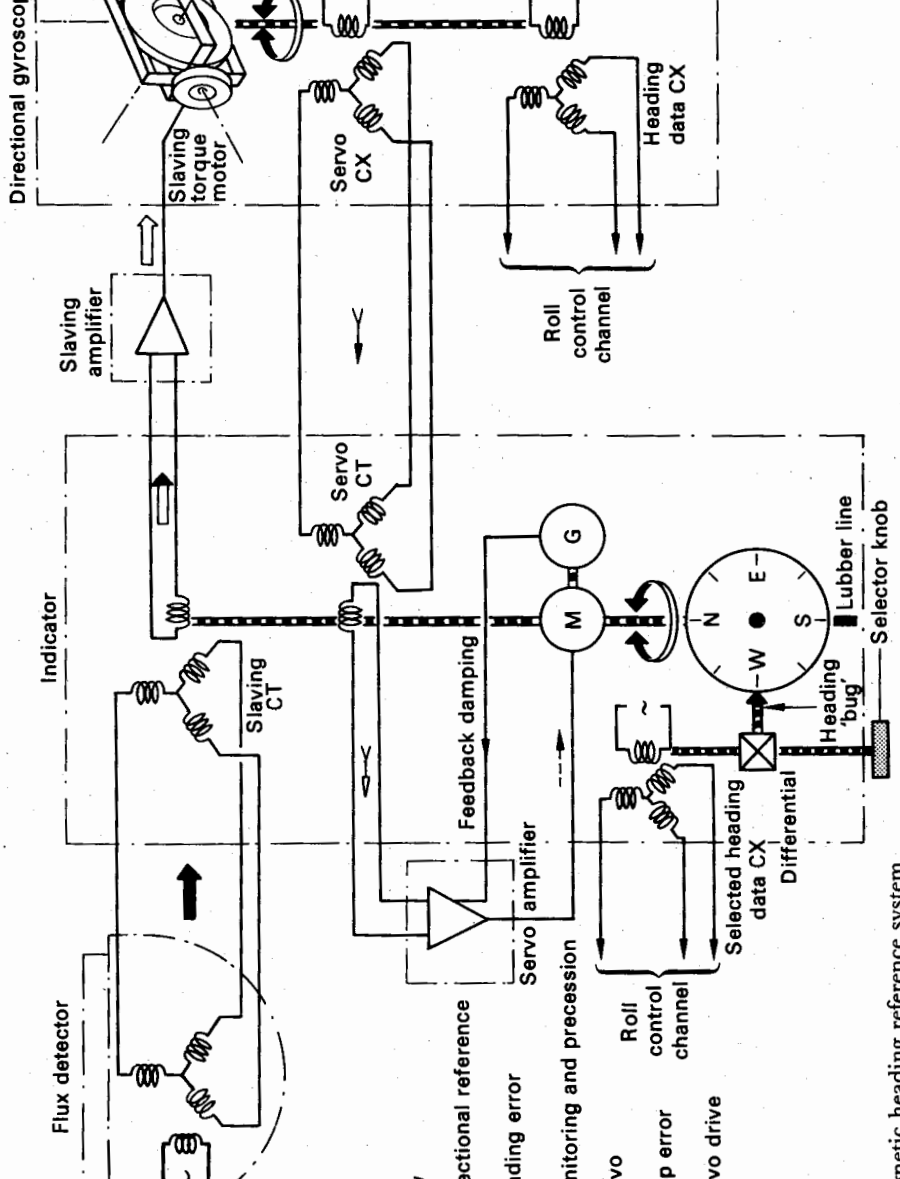
Magnetic heading reference system.

from a data CX synchro contained within the indicator. The synchro rotor is mechanically coupled to a selector knob via a differential gear which also drives a heading 'bug' with respect to the heading card. When the knob is rotated to select a heading, the synchro rotor induces an error signal proportional to the difference between the aircraft's heading and the selected heading, the signal being supplied to the roll control channel as a turn command. As the aircraft turns, the compass system will respond in the manner already described, until the aircraft is on the selected heading and the data synchro error signal is balanced out. The heading-hold mode is then selected so that the AFCS is monitored by the data CX synchro in the directional gyroscope unit.

Since the 'heading select' facility of the compass system provides automatic turn control it is comparable in function to the turn control provided on a pilot's control panel (see page 97). It is necessary, therefore to incorporate an interlock circuit between the two to prevent their signals from opposing each other. Thus, before the heading selector knob is rotated, the turn control knob on the control panel must be at its centre 'detent' position. When the turn control knob is rotated from 'detent', the compass signal circuit to the roll control channel is interrupted, but because the compass system continuously senses the heading changes produced, the appropriate signals will be reapplied when the control knob is returned to 'detent'.

In aircraft that have an inertial navigation system installed, it is usual to integrate this system with an MHRS, and also with an FDS. Details of such integration will be given later in this chapter.

Turbulence penetration

Flight in turbulent air conditions can impose varying degrees of load on the structure of an aircraft, and although designed to withstand such loads, it is necessary for the pilot to adjust power and speed, and to operate the flight control system in a manner compatible with the flight conditions prevailing.

If an aircraft penetrates turbulent air conditions while under automatically-controlled flight, the control system will sense the turbulence as disturbances to aircraft attitude, but in applying corrective control it is possible for additional structural loads to be imposed. The reason for this is that the rate of control system response tends to get out of phase with the rate at which disturbances occur, with the result that control response tends to become 'stiffer'. In turbulent conditions, therefore, it is normal to disengage the automatic flight control system. In some current systems, however, turbulence penetration may be selected as a mode of operation such that the gain of both pitch and roll channels is reduced thereby 'softening' flight control system response to turbulence.

Instrument landing system

An Instrument Landing System (ILS) is a short-range navigational aid which provides azimuth and vertical guidance during the approach to an airport runway. The system comprises ground-based transmitting elements and also receiving elements carried on board an aircraft. The ground-based elements are: (i) a *localiser* which transmits runway azimuth approach information; (ii) a *glide path*, or glide slope, transmitter which provides vertical approach information; and (iii) *marker beacons* which transmit information on the distance to the runway threshold. The airborne elements are: (i) a *localiser signal receiving antenna* (usually this is the same antenna as the one used for the VOR installation since both the localiser element and the VOR operate in the same frequency band); (ii) a *glide path signal receiving antenna*; (iii) an *ILS receiver unit*; (iv) an *indicator* which shows whether the aircraft is on the correct approach path; (v) *marker beacon antenna and receiver*; and (vi) *marker lights* on the main instrument panel.

Localiser element

The transmitter is located at the far end of the runway and it transmits on a given frequency in the band 108.0 and 112.0 MHz. To direct an approaching aircraft onto the extended centre line of the runway, the transmitter radiates azimuth guidance signals to the left and to the right of the centreline as shown in fig. 6.6. The signal transmitted to the left has a 90 Hz signal superimposed on it, while a 150 Hz signal is superimposed on the signal transmitted to the right. The two transmissions overlap along the runway centreline, and when an aircraft is approaching on the centreline extension, the ILS receiver receives both signals at equal strengths. This condition is indicated on the indicating element, usually by a vertical bar which takes up a central position over the dial of the indicating element.

If the aircraft deviates to the left of the centreline, the strength of the 90 Hz signal will be greater than that of the 150 Hz signal. Both signals pass through a comparator circuit within the receiver which then produces an output causing the vertical bar of the indicating element to be deflected to the right, thereby directing the pilot to 'fly right' in order to intercept the centreline again. Similarly, if the aircraft deviates to the right of the centreline, the 150 Hz signal is stronger than the 90 Hz signal, and so after comparison, the receiver output causes the vertical bar to be deflected to the left to direct the pilot to 'fly left' in order to intercept the centreline.

Glide path element

The transmitter is located near the threshold of the runway and it transmits on a given frequency in the band 329.3 to 335.0 MHz. The glide path

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90 Hz

150 Hz

LOCALISER

Transmitter

Indicated
commands

Fly right
Fly down

Fly left
Fly up

Both beams
captured

90 Hz

150 Hz

Transmitter
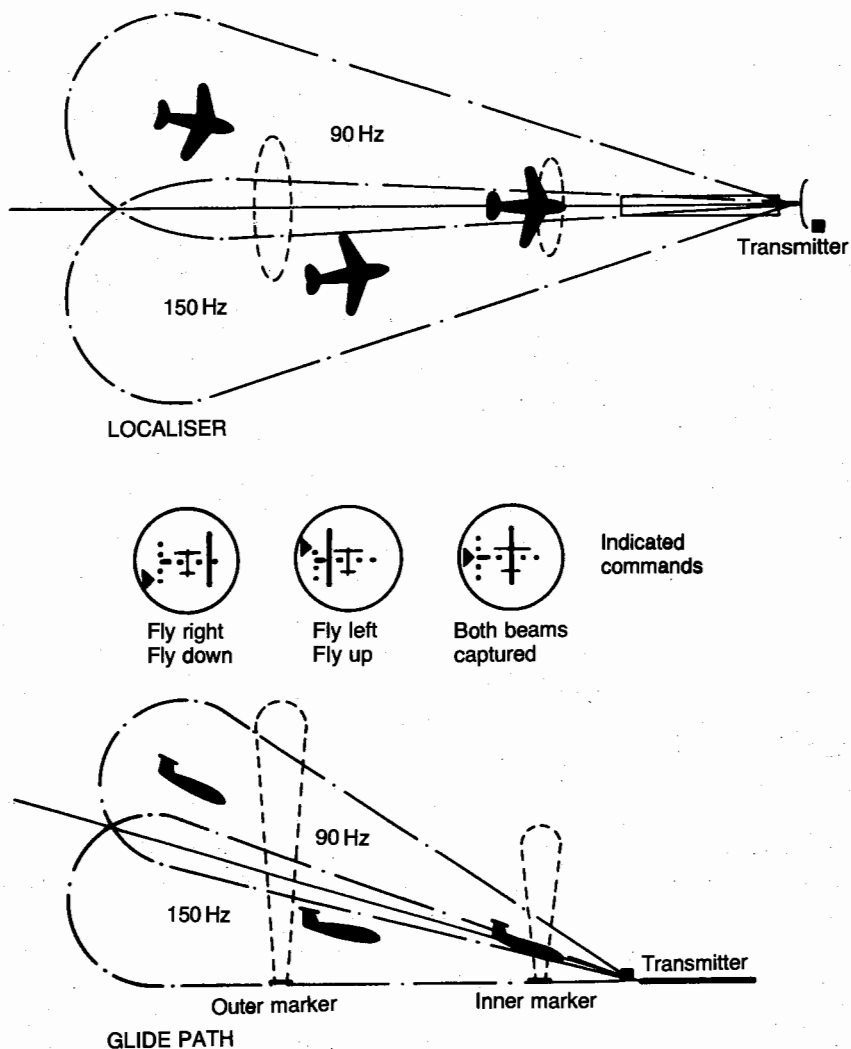
Outer marker

Inner marker

GLIDE PATH

Fig. 6.6 ILS guidance signals and commands.

transmitter radiates a signal pattern which is similar to that of the localiser but, as will be noted from fig. 6.6, the transmissions provide vertical guidance above and below a descent path established at an angle of between 2.5° and 3°.

When the aircraft approaches along this path both the 90 Hz and 150 Hz signals are received at the same strength, and this is indicated by a horizontal bar or, as in FD systems, by a pointer which takes up a central position over a scale in the indicating element. If the aircraft deviates below

or above the established path, the pointer will be deflected respectively up or down and this will provide the pilot with the corresponding flight directions.

Indicating element

When ILS was first introduced, the deviations of an aircraft from the localiser and the glide path, were presented on a separate ILS indicator having a presentation as shown in fig. 6.7. However, with the introduction of FD systems, it was logical to combine the indications with the relevant instruments (see chapter 9). Although there are many types of these systems in use, each varying in the method of data presentation, the examples of the latter shown in fig. 6.7, may be considered generally representative of such methods. In all cases, the localiser bar, or lateral deviation bar as it is more usually referred to, and the glide path pointer are deflected by independent meter movements to which the respective receiver channel signals are supplied. Scales are provided for reference against the localiser bar and glide path pointer, and thereby indicate by a series of dots the amount of deviation in degrees from the respective beam centrelines. For example, a two-dot deflection of the localiser bar corresponds to an aircraft deviation of 2.5°, and a full-scale deflection of the glide path pointer corresponds to a deviation of 0.5°. The dots are also indicative of the value (in micro-amps) of the signal currents producing meter deflections. Warning flags are provided and are visible when power is off, whenever signals are below the required strength, and whenever any malfunction or failure of the system occurs.

Marker beacons

Marker beacon transmitters are located at points along the runway approach path (see fig. 6.6), and their signals are beamed vertically into the descent path on a frequency of 75 MHz. The signals are coded, and when an aircraft flies over a beacon, they are received and can be heard over the aircraft's intercommunication system; they also illuminate appropriately coloured lights on the cockpit instrument panel. The outer marker is situated approximately four nautical miles from the runway threshold, and for identification purposes, its transmitted signals are coded with continuous dashes. The inner marker is sited approximately 3,500 feet from the threshold, its signals being coded with alternate dots and dashes.

ILS coupling

When an aircraft's navigation receiver has been tuned to a localiser frequency, the glide path frequency is also automatically tuned in, since for a given localiser frequency there is a corresponding glide path frequency. The
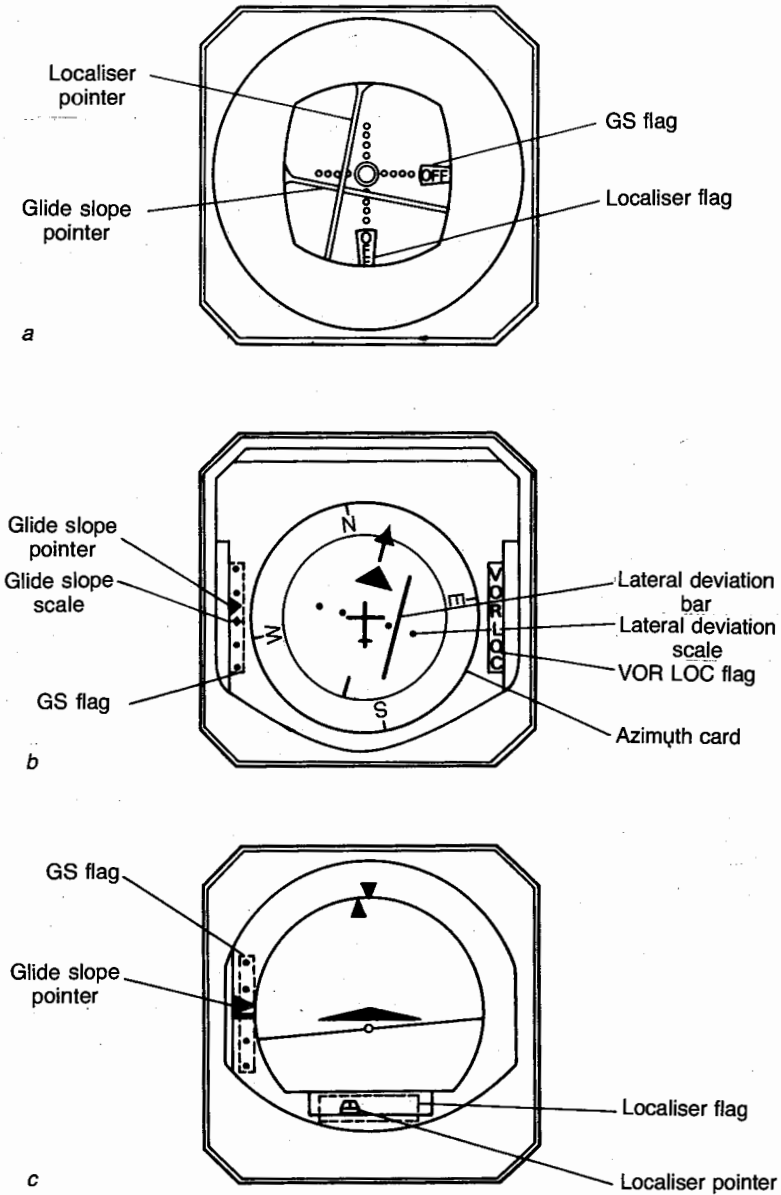
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Fig. 6.7 (a) ILS indicator (b) Horizontal situation indicator (c) Attitude director indicator.

receiver output signals are fed to the appropriate indicating element, and by following the displayed commands a pilot is able to carry out an ILS approach to an airport runway. Such an approach is normally effected in two stages; in the first stage, the aircraft intercepts and captures the

localiser beam so that it is aligned with the extended centre line of the runway, while in the second stage, the aircraft intercepts and captures the glide path beam enabling it to fly in alignment with the runway and also in the correct pitch attitude.

In order to carry out the approach under automatic control it is necessary for the flight control system to be 'coupled' to the ILS so that the system will capture the beams smoothly and at the desired beam angles. The coupling is initiated by selecting the appropriate switches on the mode selector panel.

As we have already learned, ILS signals are purely of the command type and they vary in amplitude with displacement from beam centres but, since they have no directional properties, in the sense that they cannot take into account the heading of an aircraft with respect to the runway, coupling of these signals alone to an automatic flight control system would serve no useful purpose. For example, if a localiser signal provides a 'fly right' command, and the aircraft is on such a heading that it does not come within the normal width of the beam, then such a command would remain constant for the reason that the localiser signal at that part of the beam is a constant. If this signal were to be supplied to the control channel of an automatic control system, the aircraft could well be flown in a circle, and with little chance of capturing the beam. Similarly, if heading data alone were supplied to a control channel, the heading error signals would be proportional to the difference between actual heading of the aircraft and runway direction, and would provide a 'fly left' command resulting in the aircraft flying parallel to the localiser beam.

In the practical case, therefore, it is necessary to feed the signals from both sources to the roll control channel of an automatic control system, the beam intercept and tracking commands being the resultant of both signals. The effects are shown in fig. 6.8. At point 'A', it is assumed that the aircraft is under automatic control in the heading mode of operation, and that the ILS localiser frequency has been tuned in. The deviation pointer of the appropriate indicator will, therefore, display a 'fly left' command. In order to satisfy the command, the aircraft must, of course, change its heading and since it is being flown in the heading mode, the change can be made via the heading selector of the appropriate indicator. The heading selected is the magnetic direction of the runway. Thus, a heading error signal is established and is applied to the roll control channel which then initiates turning of the aircraft on to an intercept heading. The localiser (LOC) mode is then selected so that the roll control channel can now respond to the resultant of the beam signal and the heading error signal. When the heading is such that the signals are in balance, the aircraft is then controlled so that it will fly straight and level on the intercept heading (point 'B') and at an angle governed by the ratio between the beam and heading signals. By pre-adjustment of the signal circuits, the angle of intercept can be set at a constant value as required for a particular automatic flight control system;

Fig. 6.8 ILS coupling.

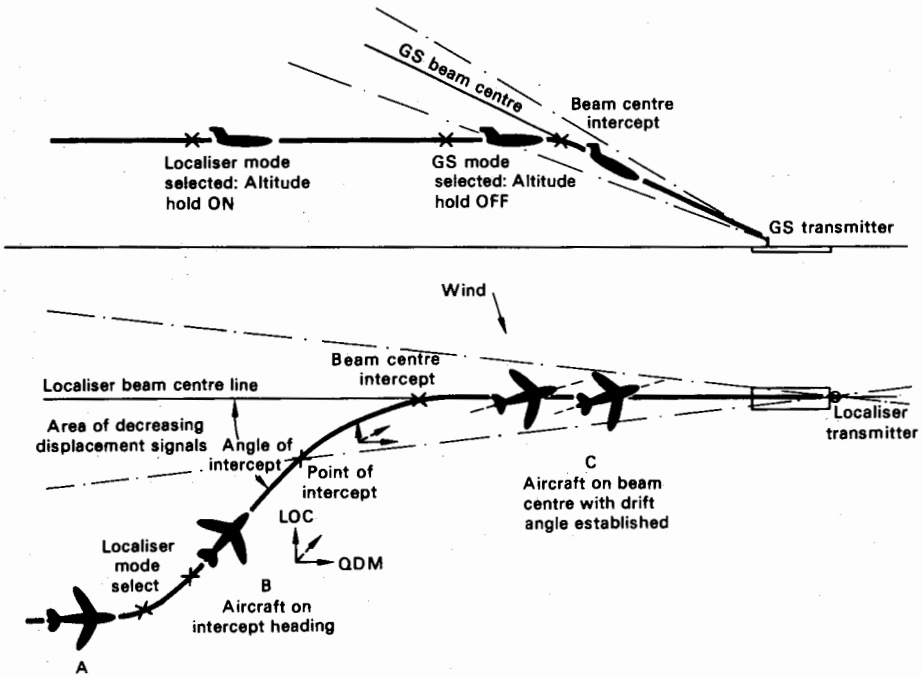some typical values are 35°, 45° and 65°. As the aircraft enters the normal width of the beam (in which the signal is proportional to deviation) the signal is correspondingly reduced, and the now predominant runway heading signal causes the aircraft to turn towards the centre of the beam until both beam and runway heading signals are again in balance. This control action is continuous until the aircraft is on the beam centre line and tracking it, when both the beam and runway heading signals are reduced to zero.

During an automatic approach, allowance must be made for the effects of crosswinds otherwise the aircraft would take up a position downwind of the localiser beam centre and would fly a 'stand-off' track parallel to the beam, and at a particular drift or 'crab' angle. This distance from the beam centre and the crab angle would depend on the strength of the crosswind, and because the aircraft takes up a position where the beam and runway heading signals balance each other, 'stand-off' and crab angle would be maintained. The method of compensating for crosswind effects varies depending on the type of automatic control system. In some cases, it is necessary to pre-calculate the drift angle appropriate to the wind conditions, and to adjust the compass indicator such that its indications and heading error signal output are offset from the runway heading by the amount of

drift angle. The relevant signal circuits are so arranged that the aircraft is turned in response to the localiser beam signal, and when this and the heading error signal are reduced to zero, the aircraft is then flown along the beam centre line with its heading off-set by the drift angle (point 'C' in fig. 6.8).

In a number of systems, it is more usual to supply part of the beam signal to an integrator circuit with a long time-constant. The circuit is activated when the aircraft is near the beam centre line, i.e. when the beam signal falls to a low value, and remains in action until the signal has fallen to zero. Since the presence of a crosswind slows the speed of interception of the beam centre line, the integrator signal is built up in porportion to the amount of crosswind. The build up continues until the beam centre line is reached, at which point the aircraft's heading is controlled by the integrator signal and the pre-selected heading signal.

As noted earlier, when the navigation receiver is tuned to the localiser frequency, the glide path frequency is also tuned in; thus, when the aircraft is correctly established on the localiser beam, the glide path signals can be used to effect the final phase of the approach. The glide path can be intercepted and captured while the aircraft is being flown in the altitude-hold mode, and from above or below the glide path. Ultimately, the aircraft must, therefore, fly through the glide path and its proximity will be evidenced by 'fly down' or 'fly up' commands of the associated indicator. At the point of beam interception, the glide path mode is selected on the mode selector panel, and the beam signals are then supplied to the pitch control channel of the flight control system. Thereafter, changes of position of the aircraft from the glide path produces control channel response to restore the required pitch attitude.

At the same time that the glide-path mode is selected, the altitude-hold mode is automatically disconnected. A pitch control channel incorporates a circuit network which phase advances the glide path error signals so that they are damped to give progressively tighter control as the aircraft flies down the glide path, and thereby allowing for the converging nature of the beam (see also page 156). In flight control systems which use a vertical-axis gyroscope as the attitude sensing element, the pitch error signal produced would tend to 'back off' the beam error signal and cause the aircraft to fly above the beam. This is also prevented by passing the signals through an integrator circuit.

VOR system

A VOR (VHF omni-directional radio range) system not unlike the ILS is a short-range navigational aid comprising ground-based transmitting stations, or beacons, and receiving elements carried on board an aircraft. It differs in function however, in that it provides en-route information on the bearing of
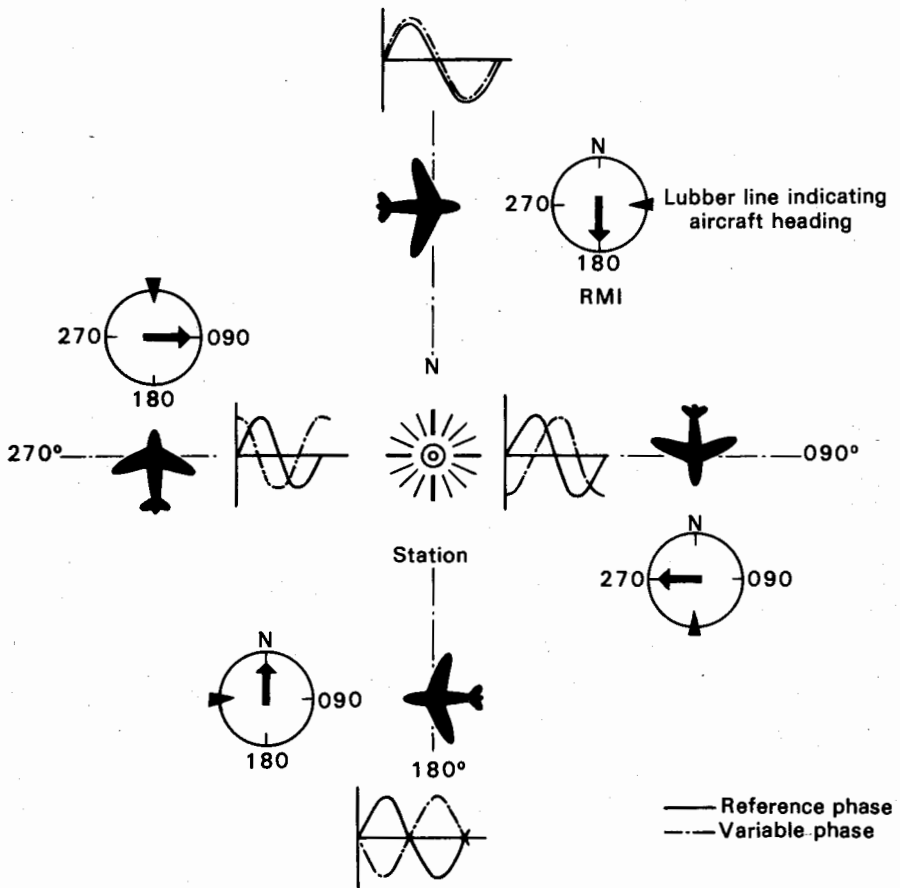
Fig. 6.9 VOR system.

an aircraft from the points at which the stations are geographically located. The stations are spaced at intervals of 50 to 80 nautical miles within what is termed the 'airways system'.

A VOR station transmits a very high frequency (VHF) carrier wave operating in the 108 to 118 MHz band, and on which are superimposed two low frequency modulating signals. One of these signals, known as the reference signal, is radiated in all directions (hence the term omnidirectional) with a fixed phase, while the other is a rotating beam signal and varies in phase to produce an infinite number of 'radials'. Thus, at any particular point relative to the station (which is lined up on magnetic north) a specific phase relationship exists between the two signals, and this is indicated in fig. 6.9. Each station is identified by a Morse code signal which is received by the navigation receiver in the aircraft when the
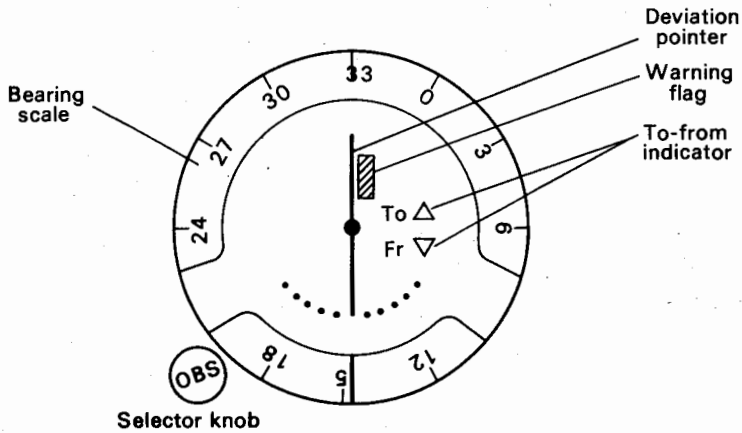
Fig. 6.10 Omni-bearing indicator.

corresponding station transmitting frequency has been selected. The display of bearing information is presented on an indicator mounted on the aircraft's main instrument panel. The type of indicator depends on the equipment specified for a particular type of aircraft; for example, it may be an omni-bearing indicator, a radio magnetic indicator (RMI), or it may be a horizontal situation display indicator (HSI) which forms part of a flight director system (FDS) (see chapter 9). In general, however, the display comprises three main indicating elements: (i) a bearing scale or an index, which is positioned by a selector knob to indicate the radical on which the aircraft is to be flown; (ii) a deviation pointer indicating whether the aircraft is flying on the radial; and (iii) a TO-FROM indicator to indicate whether the aircraft is flying towards a station or away from it. A warning flag is also provided to indicate 'system off', receiving of low-strength signals, or no signals at all.

Figure 6.10 illustrates an omni-bearing indicator, and this serves as a useful example in understanding the fundamentals of VOR operation as a navigational aid. After a station has been identified from its Morse signal and tuned-in, the omni-bearing selector knob is rotated to set the required bearing on the bearing scale, and also to position the rotor of a resolver synchro. The navigation receiver which continuously compares the phases of the transmitted signals, and compares them in turn with the phase shift produced in the resolver synchro, supplies an output signal to a meter movement controlling the vertical pointer of the indicator. When the pointer lies in the centre of the indicator it indicates that the phasing of the required bearing radial signals has been matched and that further rotation of the selector knob is unnecessary. The reading shown on the bearing scale at that instant indicates the heading on to which the aircraft must be turned in order to fly along the corresponding radial to the station.

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At the same time, the TO-FROM indicator indicates TO usually by means of an arrow-shaped pointer. Any departure from the radial as a result of, say crosswind drift, the deviation pointer will be deflected to one or other side, and so command a turn in the direction of the deflection in order to intercept the radial once again. Thus, the command function is identical to that of the localiser pointer of an ILS indicator and, since localiser and VOR operate in the same frequency band, omni-bearing indicators and the indicators of flight director systems, are in fact designed such that their deviation pointers serve the dual command function.

As the aircraft approaches overhead the station, so the deviation pointer mechanism becomes more sensitive as a result of the convergence of the radials, and its indications, together with those of the TO-FROM indicator, become erratic. In this part of the approach, overhead the station and in departing from it, the aircraft is said to be in the 'cone of confusion'. The extent of the cone varies with altitude and groundspeed, typically from a few seconds at low altitude, to as much as two minutes at high altitude. Outbound flight from the station is indicated by the TO-FROM indicator changing to FROM and, provided an accurate heading is maintained, the deviation pointer continues to give corrective command information for the interception of the outbound radial.

VOR coupling

Operation in the VOR coupled mode is similar in many respects to that of the localiser mode, in that it also involves the capture of radio signal beams radiated in azimuth in association with steering commands from magnetic heading data; such data, however, being related to a pre-selected VOR station radial instead of a runway heading. When the navigation receiver has been tuned to the station frequency and the required radial has been selected on the course indicator, coupling to the roll control channel is initiated by selecting the LOC/VOR mode. As a result of the convergence of the radials, and of the cone of confusion over a VOR station, the signal from a selected radial is unreliable and could, therefore, cause the automatic flight control system to carry out unwanted manoeuvres. To prevent this, it is usual to 'cut off' the VOR signal at the entrance to the cone of confusion, so that the roll control channel responds only to heading data signals. Cut off takes place automatically by means of an over-station sensor (OSS) circuit, and is maintained for a pre-set time period, after which, and with the aircraft departing from the cone of confusion, the VOR signal corresponding to the outbound radial is switched in.

Inertial navigation system

The requirement for the navigation of an aircraft is, quite simply, the one of determining its position in relation to its point of departure, and points en-
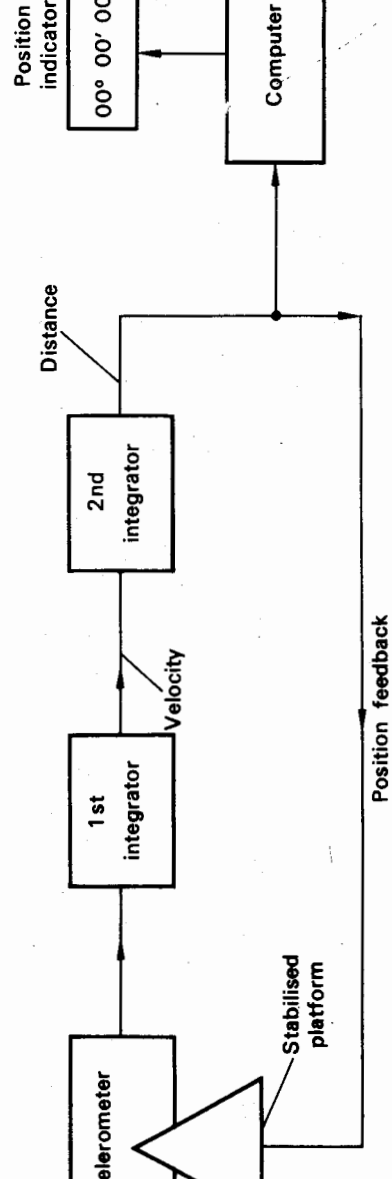
route, in order to reach a known destination. The basic data necessary for this purpose are principally time, speed, distance between points, longitude and latitude, magnetic heading, wind speed and direction, and bearings relative to known points on the earth's surface and to celestial points. The provision of such data is made by a variety of navigational aids, most of which are dependent on an external reference of one form or another.

Although such aids can provide reasonably accurate answers to the problems associated with the navigational task, the remaining errors involved make them only relatively useful. In the continuing development of navigational aids, particularly in relation to the stringent requirements laid down for the strategic roles of military aircraft, for missiles, and for spacecraft, it became essential to provide aids having much greater accuracy. These also needed to be independent of references derived from ground-based navigational beacons in particular. The attendant research ultimately resulted in the application of a theory which, as is so often the case, is based on old established laws; in this instance the theory of inertia applied to what subsequently became known as an Inertial Navigation System (INS), the basic elements of which are shown schematically in fig. 6.11. The navigational problems which can be solved by INS are shown in fig. 6.12.

Inertial theory is based on Newton's second law, i.e. that force is equal to time rate of change of momentum, without approximation or correction for any effect. In plainer terms, it can be stated that velocity is the rate of change of distance with time, and acceleration is the rate of change of velocity with time. If this deduction is reversed, it can be seen that a vehicle's position at any time after starting to move can be determined solely from its acceleration history, and by double integration of measured accelerations the distance travelled during a given time may also be determined.
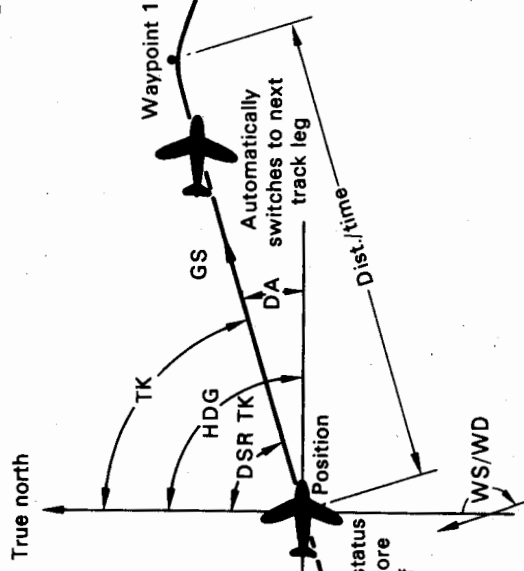
A typical INS comprises the three principal units illustrated in fig. 6.13(a) and (b), they are a mode selector unit, an inertial navigation unit comprising a digital computer and inertial platform, and a control display unit. The 'core' of the system is the inertial platform which establishes the stable reference plane from which measurements can be computed. The inputs needed to keep the system in touch with the outside world come from accelerometer-actuated pick-off elements, the signals being fed to the computer in proportion to acceleration forces. Since the functions of navigational geometry lie in a horizontal plane, two accelerometers are mounted on the platform; one oriented north-south, the other oriented east-west. Thus, the computer is able to record the acceleration history of an aircraft, and therefore determine its position in relation to its starting point. Angles that fall between the north-south or east-west relationship are calculated by the computer solving algebraic vector problems. A third accelerometer measures acceleration in the vertical direction.

To make the measurements refer only to translational accelerations, the

Basic elements of an inertial navigation system.

True north

TK
HDG
DSR TK
Position
WS/WD
GS
DA

Automatically
switches to next
track leg

Dist./time

Waypoint 1

Aircraft on track

GS
DA
DSR TK
HDG
TK
XTK
Position
Distance
TKE
Time
Waypoint

Aircraft off track

...avigational problems solved by INS.
... speed): Speed of aircraft over surface of earth.
...ngle): Angle in degrees that aircraft track is to right
...rcraft heading.
...k angle error): Angle in degrees that aircraft track is
...ght of desired track.
...s track): Distance left or right from desired track to
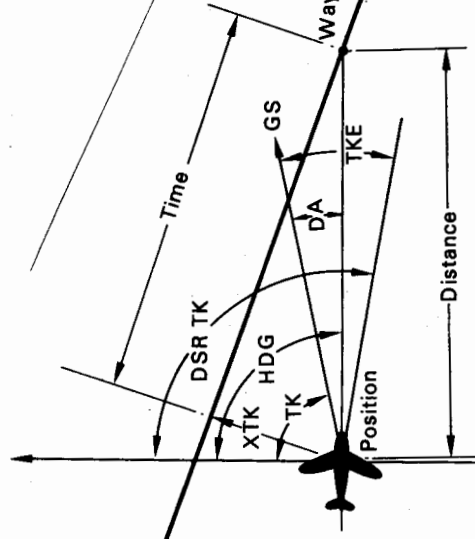
present position, measured perpendicular to track desir...
TK (Track): Actual path of aircraft over earth's su...
Measured clockwise from true north through 360°.
DSR TK (Desired track): A great circle path on surface of...
connecting two waypoints.
HDG (Heading): Angle between true north and longit...
axis of aircraft.

accelerometers must at all times be maintained in accurate alignment with their respective directions, i.e. the platform must be maintained in correct azimuth alignment. This is accomplished by mounting the platform in a gimbal ring system and controlling the position of the rings by three rate-integrating gyroscopes and torque motors. In addition, the platform must be maintained accurately horizontal in relation to the earth's surface in order to prevent the accelerometers from sensing misleading gravity accelerations. Errors resulting from earth rate and transport rate (see pages 110 and 112) and from Coriolis effect, are also automatically corrected by computed signals applied to the torque motors. The arrangement of the platform and gimbal system, and method of stabilising, have already been described in chapter 3 (see page 124).

To allow for changes in latitude, the output of the north-south accelerometer is doubly integrated so that the angular acceleration of the aircraft above the earth's centre can be followed. A slightly more complex correction must be applied to allow for changes in longitude because such changes are also dependent upon latitude. The output velocity signal of the east-west accelerometer integrator is therefore multiplied by the secant of the latitude and then passed to the platform monitoring torque motor.

The inertial platform has one further characteristic and that is of being tuned so that at all times it accurately seeks a local vertical reference coincident with the earth's centre. This is achieved by applying the principle of what is known as the Schuler pendulum in such a way that by utilising error compensations from the computer together with a mathematical model in the computer, the platform is made to behave like a pendulum whose arm is equal in length to the earth's radius, and whose centre of rotation is coincident with the earth's centre. The result is that any build-up of errors in the accelerometers and gyroscopes is limited and minimised, and the platform stays parallel to the earth's surface regardless of which latitude and longitude it is put into operation.

A digital computer in the system's navigation unit continuously solves the associated navigation equations, from the signal data fed into it by the control display unit, and by a central air data computer. The signals from the latter relate to true airspeed, which information is used for determining wind speed and direction and thereby the drift angle, and to altitude information for damping an integrating loop in altitude computation. Output signals are used for displaying computed data, and also for manoeuvring the aircraft towards the selected waypoint, via the roll channel of the automatic flight control system.

The control display unit utilises a digital readout which presents the pilot with all the data necessary for monitoring flight progress. Information for display is inserted by means of a 'keyboard'; it can also be obtained by turning the knob of a data selector switch to desired settings. A digital readout also provides information on the changes of track of the aircraft as it proceeds from one waypoint to the next.

FLIGHT DIRECTOR
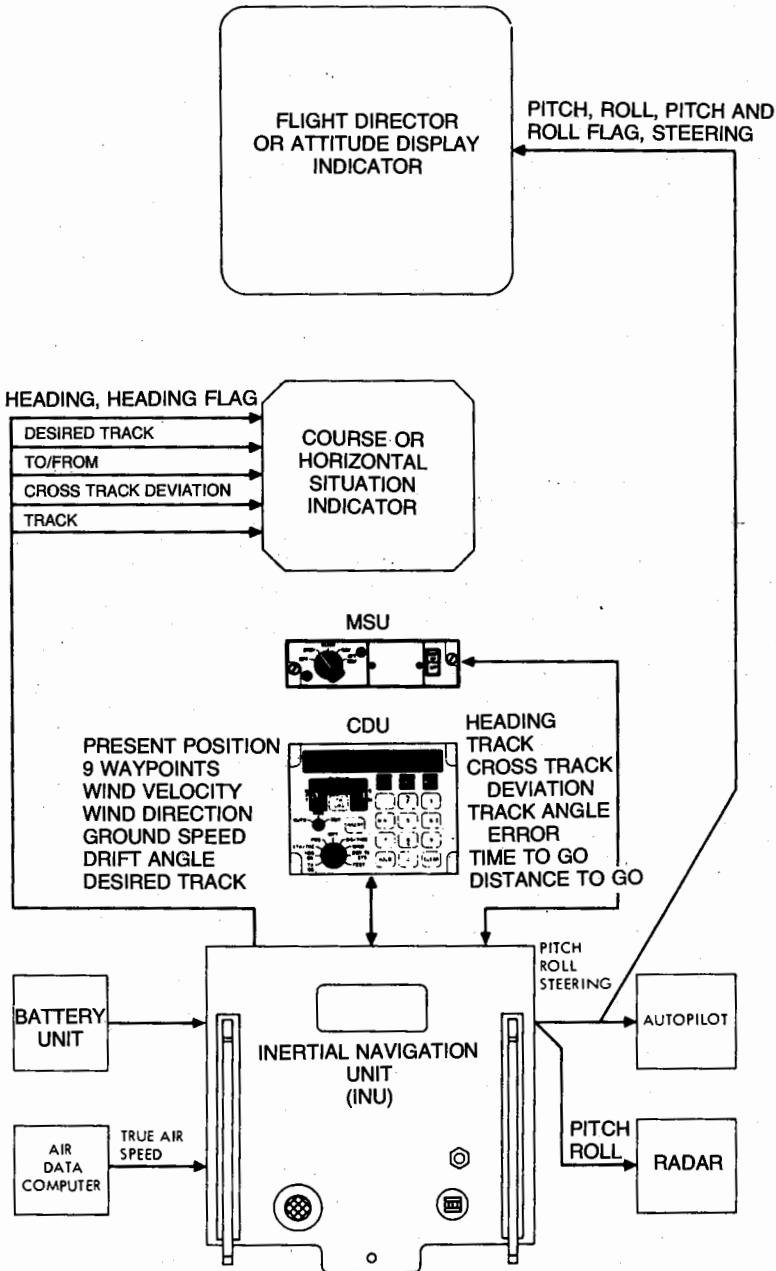OR ATTITUDE DISPLAY
INDICATOR

PITCH, ROLL, PITCH AND
ROLL FLAG, STEERING

HEADING, HEADING FLAG

DESIRED TRACK

TO/FROM

CROSS TRACK DEVIATION

TRACK

COURSE OR
HORIZONTAL
SITUATION
INDICATOR

MSU

CDU

PRESENT POSITION
9 WAYPOINTS
WIND VELOCITY
WIND DIRECTION
GROUND SPEED
DRIFT ANGLE
DESIRED TRACK

HEADING
TRACK
CROSS TRACK
DEVIATION
TRACK ANGLE
ERROR
TIME TO GO
DISTANCE TO GO

PITCH
ROLL
STEERING

BATTERY
UNIT

INERTIAL NAVIGATION
UNIT
(INU)

AUTOPILOT

AIR
DATA
COMPUTER

TRUE AIR
SPEED

PITCH
ROLL

RADAR

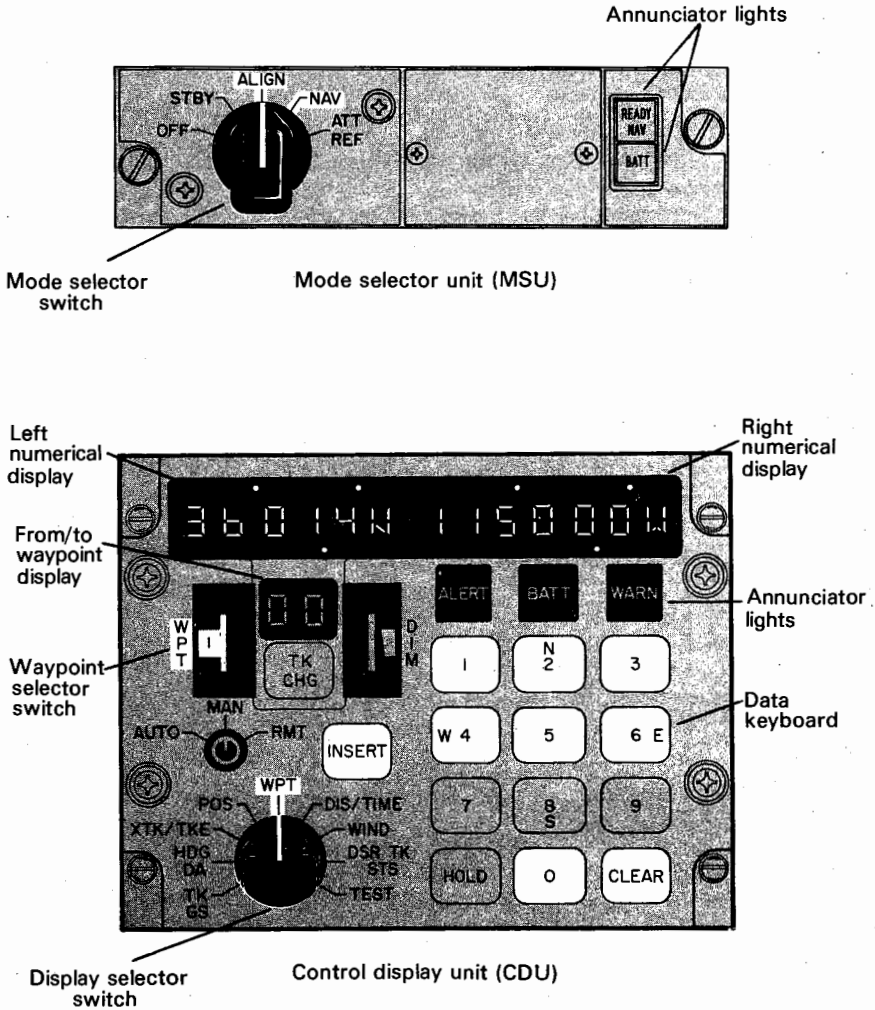Fig. 6.13(a) Inertial navigation system units.

Fig. 6.13(b) Inertial navigation system units.

Prior to departure, and before any movement of the aircraft takes place, the system is operated in a 'standby' and an 'align' mode, during which the inertial platform is automatically aligned to the aircraft's axes. The latitude and longitude of the aircraft at its departure position are also inserted at the control display unit, and this information is integrated into a mathematical model within the computer and, by a procedure known as gyrocompassing, the system is also aligned to its north reference point. The co-ordinates of the en-route waypoints and of the destination are also subsequently entered in the control display unit. When the alignment sequence is completed, an annunciator light illuminates to indicate that the system is 'ready to
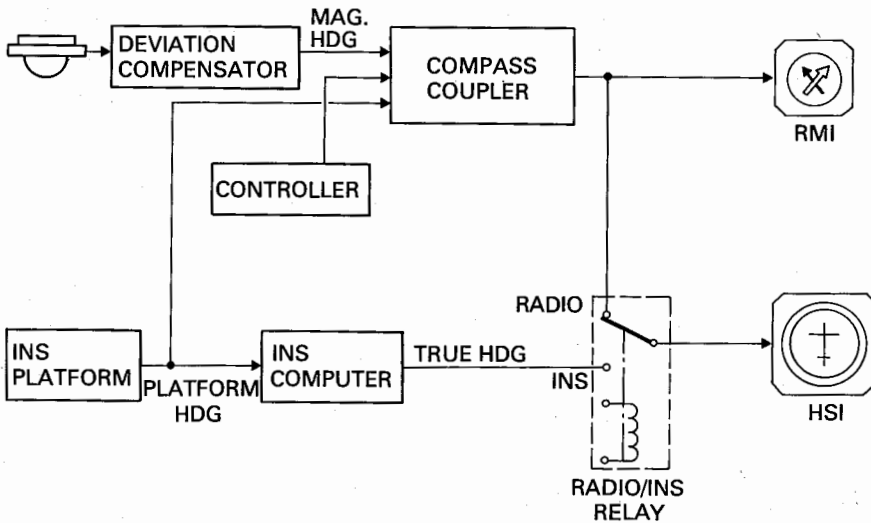
Fig. 6.14 INS/MHRS integration.

navigate' and that following selection of the navigate mode on the mode selector unit, the aircraft may commence its flight.

During flight, and approximately two minutes before a waypoint is reached, a light on the control panel comes on, and then begins to flash on and off thirty seconds before the waypoint is crossed, thereby signalling for the aircraft to be turned. With the automatic flight control system engaged, the INS supplies command signals to the roll control channel so that the aircraft will automatically roll into a turn towards the waypoint and will level out on the new heading from the waypoint. This is repeated at each en-route waypoint until the final destination is reached.

INS/MHRS integration

The inertial platform of an INS also provides short-term stabilising of heading references, and, this being so, it can also be integrated with a magnetic heading reference system (MHRS) and also a flight director system (FDS), as shown schematically in fig. 6.14. As far as the MHRS is concerned, there is an advantage in that there is no longer a requirement for it to have its own directional gyro unit. Other differences are:

(i) Magnetic heading indications are displayed on an RMI; and
(ii) slaving/heading and servo signal transmission and control circuits are contained within a compass coupler unit.

The signal inputs to the compass coupler are; magnetic heading as detected by the MHRS flux detector unit, and platform heading derived from a
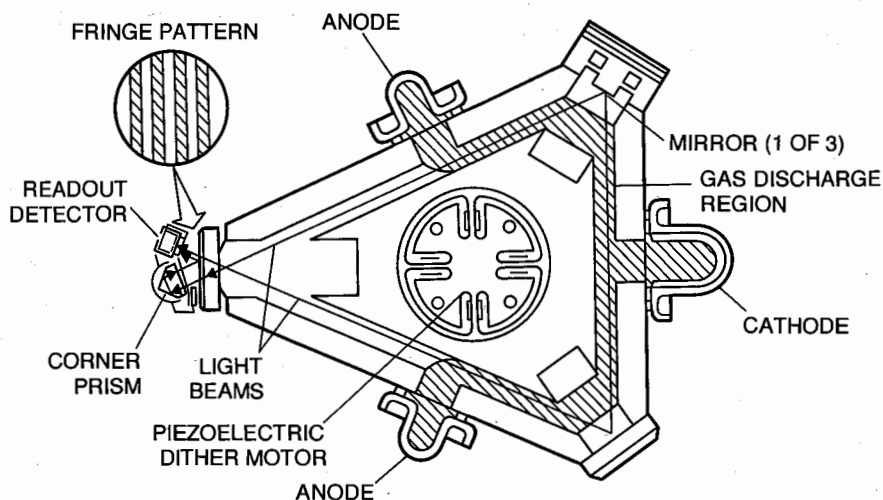
Fig. 6.15 Ring laser gyro.

control synchro actuated by the inertial platform. The detection and servo circuits within the compass coupler unit maintain synchronism between the two signal inputs. True heading can also be derived from the inertial platform and INS computer, and, as will also be noted from fig. 6.14 when selected it is displayed on the HSI of an FDS.

Ring laser gyro

In several types of current generation aircraft, the inertial systems are of somewhat sophisticated design in that conventional forms of gyroscope are replaced by units to which laser beam techniques are applied. They still operate in conjunction with accelerometers and a display unit, but with the advantage of eliminating rotating parts and gimbal systems. Such gyros are referred to as ring laser gyros and are arranged as shown in the simplified diagram of fig. 6.15.

The first thing to note is that it is quite unlike a conventional gyroscope in that there is no rotor and gimbal system; rotating parts are in fact unnecessary. The basis of it is a triangular block of specially fabricated glass (known as cervit glass) that it is extremely hard and does not expand or contract with heating and cooling. By means of computerised ultrasonic diamond drilling techniques a cavity is formed within the whole block. A mirror is fitted at each corner of the block, and a cathode and two anodes are located as shown.

The cavity is filled with a lasing medium (typically helium-neon) and when excited by an electrical potential across the cathode and anodes two

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light beams are emitted and, by design, are made to travel in opposite directions. Since they are 'bounced off' by the mirrors, they then follow closed paths around the cavity. Both beams combine in a prismatic type of optical sensor at one corner of the block.

In applying the technique to the inertial system mentioned earlier, three laser gyro assemblies are required, and they are mounted mutually at right angles to each other so that when they are 'strapped down' to an aircraft's structure they relate directly to movements about its pitch, roll and yaw axes.

With the aircraft in straight and level flight, all three gryros are in a static condition, and in this condition, the resonant frequencies of the beams are equal.

If the aircraft's attitude is now changed about, say the pitch axis, the corresponding gyro will also be rotated about this axis. This sets up a difference between the resonant frequencies of the two beams or, putting it another way, one beam takes slightly longer than the other to complete a circuit around the triangular block.

When the two beams combine in the optical sensor an interference pattern results, and the movements of this pattern are counted by the sensor which then produces a digital output that is proportional to the angular rate at which the gyro has been rotated.

Similarly, the other two gyros would produce output signals proportional to the angular rates at which they rotate about their respective axes. After computerised integration processes, the rates provide an inertial position reference equivalent to that achieved when gyroscopes of the more conventional type are used.

Systems utilising ring laser gyroscopes perform the same basic navigational functions as the earlier forms of INS, but they are much more sophisticated in that their utilisation of digital computing techniques permits them to be pre-programmed with so much other relevant reference data that they are now classified as inertial reference systems (IRS).

Control wheel steering

A control wheel steering mode (CWS) is provided in some AFC systems, its purpose being to enable the pilot to manoeuvre his aircraft in pitch or roll, through the automatic control system by exerting normal manoeuvring forces on the control wheel. In adopting this mode, it is not necessary for pitch and turn controls to be provided on a system control panel.

When the control wheel is released, the AFCS holds the aircraft at the newly-established attitude. The pitch and roll forces applied by the pilot are sensed by force transducers which generate output voltage signals proportional to the forces; the signals are supplied to the pitch and roll channels of the AFCS. The locations of transducers depend on the type of

aircraft and control system, but typically they are either in the hub of each pilot's control wheel, or in the control system below each pilot's control column.

In some cases, limits may be imposed, e.g. if a roll angle is less than 5°, wing levelling will automatically occur, and the control system will hold the aircraft on a heading established one second after the wings are level. Prior to, and during, capture phases of radio navigation modes of operation, the pilot can use CWS for supervisory override of the AFCS. Thus, with the aid of the CWS mode, the pilot always has a control capability of the aircraft, and the aircraft does not have to follow a pre-programmed flight path when particular conditions dictate a different manoeuvre.

In one example of CWS sensor unit used in a current type of AFCS, piezo crystal elements are employed which have the property of varying their electrical resistance whenever they are subjected to varying pressures. The piezo-resistive elements as they are generally known, are connected to the form of an electrical bridge, and are embedded in the upper and lower portions of a specially designed spring contained within the sensor unit casing which is itself secured within the hub of the control wheel on the pilot's control column.

One bridge circuit responds to roll command forces applied through rotation of the control wheel, while the second circuit responds to pitch command forces applied by fore and aft movement of the control column. Pairs of elements of each bridge circuit are located on opposing sides of their appropriate portions of the spring, and they are supplied with a d.c. excitation voltage of 9 V from the flight guidance system computer. The output signals from the bridge circuit are fed as appropriate to the roll and pitch channel circuits of the computer, only when the CWS mode of operation is selected, and when forces are manually exerted on the control wheel. With no forces applied each bridge circuit is in a balanced condition and so no signal outputs are produced.

When, for example, the control wheel is rotated, the force exerted causes the roll spring to bend thereby compressing the pair of piezo-resistive elements on the inner side of the spring causing their resistance to decrease. The pair of elements on the outer side are stretched and as a result their resistance is increased. The bridge circuit is therefore unbalanced and a change in polarity and voltage occurs, corresponding to the direction of control wheel movement and amount of force applied. After processing by the computer the signal is supplied to the aileron servo-actuator which displaces the ailerons in the direction required; typically, positive and negative output signals result in roll displacements to the right and left respectively.

The pitch spring and its associated piezo-resistive elements respond to forces in the same manner to produce output signals for the operation of the elevator actuator. In this case, positive and negative signals result in nose up and nose down displacements respectively.

In another example of AFCS, the sensing of control wheel steering forces is accomplished by devices known as dynamometric rods. These are connected directly into the appropriate flight control circuits of the aircraft, and consist essentially of a cylinder containing a spring-controlled synchro. One end of the cylinder is connected to the controls via a fixed rod while the other end is connected via a rod in such a way that relative movement between this rod and the cylinder is obtained when the controls are operated by the pilot. The rod is also connected to the spring so that as appropriate movement takes place, the spring is subjected to either compression or tension forces. Rod movement also causes rotation of the synchro rotor which produces output signals proportional to the forces applied.

Force transducers may also be of the E and I bar type (see page 130) housed inside a spring unit mechanically coupled at each end into the elevator and aileron control systems. When either the control column or control wheel is displaced, then depending on the direction of displacement the spring is compressed or extended. This results in relative linear displacement of the E and I bars, and an output voltage to the control channels that is proportional to the force applied.

Touch control steering

Touch control steering (TCS) also permits a pilot to manoeuvre his aircraft in pitch or roll, but unlike CWS the appropriate automatic control channels and servomotors are disengaged while the pilot flies the aircraft to the desired attitude. Disengagement is established by operation of a switch on the pilot's control wheel. At the same time that the switch is operated, a feedback signal path is provided into the integrating circuits of the appropriate control channel to ensure synchronisation with the new attitude required, and also nulling out of the servo loop output.

When the TCS switch is released, the feedback path is interrupted, and the relevant integrator retains the attitude signal existing at that time. The servomotor is re-engaged to hold the new attitude. If the aircraft deviates from the synchronised attitude, an error signal is produced at the integrator output and is used to drive the servomotor and aircraft back to the synchronised attitude.

TEST QUESTIONS

1 What outer loop control inputs constitute manometric data, and to which control axis are they related?
2 Why is it necessary to provide an altitude hold mode?
3 How is it ensured that an altitude hold unit is always at a datum altitude?

4 Explain how in the altitude hold mode, a typical unit detects and corrects any departure from the datum altitude.

5 State the sequence in which an altitude alerting system operates when an aircraft descends to a pre-selected altitude.

6 What sources of air pressure are required for the operation of airspeed and Mach hold sensor units?

7 What is the purpose of an air data computer?

8 In terms of a pressure ratio, Mach number is equivalent to:
(a) $P - S/P$;
(b) $S/P - S$;
(c) $P - S/S$.

9 Describe the operation of a digital type air data computer.

10 Heading hold signals for an automatic flight control system are supplied to the:
(a) yaw channel;
(b) roll channel;
(c) pitch and roll channels.

11 How are the heading signals transmitted from a compass system to the appropriate channel of an automatic flight control system?

12 How is it ensured that corresponding magnetic heading change signals are supplied after the pilot has injected a turn command signal through his control panel?

13 What is the purpose of the turbulence penetration mode, and what effect does it have on a control system when selected?

14 Azimuth information during an ILS approach is provided by signals from:
(a) the glide slope transmitter;
(b) marker beacons;
(c) the localiser transmitter.

15 Explain how deviations of an aircraft from the signal paths transmitted by the ILS are indicated, and by means of a simple diagram show a typical situation.

16 How is it ensured that during an automatic approach an aircraft intercepts the ILS localiser beam at the correct angle and point on the beam centre?

17 What is meant by 'beam stand-off' and how is compensation made for it?

18 Briefly explain the operation of a VOR system.

19 How is the VOR mode selected and to which channel of an automatic control system are the signals supplied?

20 What is the purpose of an over-station sensor?

21 In connection with the provision of navigational data, in what way does an INS differ from other systems?

22 Define the following: TKE, XTR and DSR TK and by means of a diagram show how they are related to the position of an aircraft.

23 How are accelerometers utilised in an INS and what method is adopted to ensure that they are maintained in accurate alignment?

24 What part does the Schuler pendulum principle play in the operation of an INS?

25 Briefly describe the construction of a ring laser gyro.

26 How does a ring laser gyro detect movements of an aircraft about an axis?

27 What is the purpose of the control wheel steering mode?

28 Describe a typical control wheel steering transducer and how it transmits signals to the appropriate control channel.

29 How does touch control steering differ from control wheel steering?

7
Conversion of Command Signals to Powered Control

The power output element of any automatic flight control system consists of servomotors, or servo-actuators as they are sometimes called, connected in the aircraft's primary flight control system circuits; the number of servo-actuators employed is governed by the number of control loops required. In addition to the actuation of primary flight controls, servomotors may also be used, in some cases, for the actuation of the secondary flight controls provided for trimming purposes and for yaw damping (see pages 213 and 222). In the case of aircraft that utilise hydraulic-power-operated control units (PCUs), command signals can be applied directly to the actuators of the relevant PCUs, thereby eliminating the need for separate servo-actuators, or they can be applied to servo-actuators that control the PCUs via mechanical interconnects.

In general, servo-actuators operate on either electro-pneumatic, electro-mechanical, or electro-hydraulic principles, the choice, and constructional features adopted in applying such principles being dependent on the type of automatic control system, and on the methods adopted for actuation of the primary flight control surfaces. Servo-actuators may be connected either in series or in parallel with the normal flight control system of an aircraft. A series-connected servo-actuator is one which moves the flight control surfaces without moving the pilot's controls, while a parallel-connected servo-actuator moves both the control surfaces and the pilot's controls.

Electro-pneumatic servo-actuator

An example of a servo-actuator designed for use in one particular type of three-axis autopilot system is shown schematically in fig. 7.1. It consists of an electro-magnetic valve assembly, comprising dual poppet valves which are connected via pressure ports and orifices to two cylinders containing pistons sealed against pressure loss by rolling diaphragms (also called 'roll-frams'). The valves are controlled by electrical command signals from the autopilot signal processing element, and the pressure for actuation of the pistons is supplied either from an engine-driven pump or from a tapping at
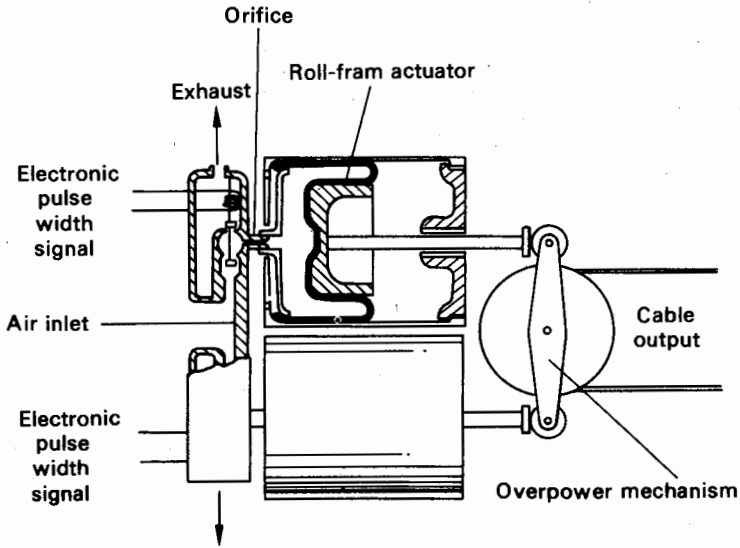
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Fig. 7.1 Electro-pneumatic servo-actuator.

a turbine engine compressor stage. The operating pressure is determined by the control force characteristics of the aircraft in which the particular autopilot is installed; a typical pressure range is from 7.5 to $30\,lb/in^2$. The piston rods are designed to drive an output linkage assembly which is connected to the appropriate flight control system circuit through a cable drum. The control input to the solenoid of each poppet valve is, in this example, a 22 Hz pulse-width modulated, square-wave signal, the particular frequency being chosen for minimum pressure disturbance and maximum valve life. The input pulses alternately open and close the valves.

With no command input, each valve is open for an equal period of time, and so there is equal pressure in both cylinders and no output torque is transmitted to the control system. When a control command signal is introduced, the open-time period of one valve is increased, while the open-time of the other valve is decreased. Thus, a differential pressure is developed in the two cylinders causing one piston rod to be extended and the other to be retracted, thereby causing rotation of the output linkage and deflection of the control surface(s) to which it is connected. A fixed orifice is installed at the inlet to each cylinder, the size of the orifice being selected to establish the response rate, or control gain, for the particular aircraft type.

As is conventional for the power output element of any automatic control system, the output linkage assembly of the servo-actuator shown can operate in any of three conditions: namely disengaged, engaged, and override. In the disengaged condition, the links and piston rods are retracted, allowing free rotation of the output shaft and cable drum, which is con-
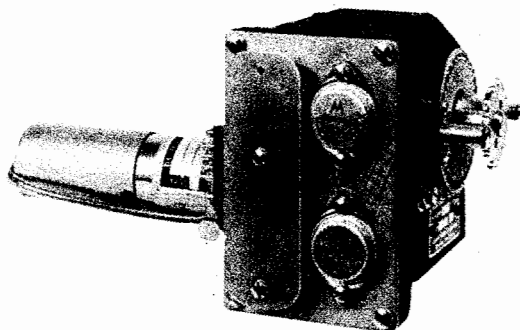
Fig. 7.2 Direct-current operated servo-actuator.

nected into the control system, to equal or to exceed the stop-to-stop travel of the appropriate control surface. When the servo-actuator is pressurised on engagement of the autopilot, the piston rods and links move into contact with the output shaft arm and limit rotation of the arm such that the nominal control range of the autopilot does not exceed 50% of the range of the aircraft's flight control system. To provide full control capability for overriding the automatic control system, the linkage assembly is articulated, but is normally held rigid by springs. Application of override force by the pilot allows deflection of the springs, and provides extension of the servo-actuator's nominal control range to that of the flight control system.

Electro-mechanical servo-actuators

Depending on the type of automatic control system, these servo-actuators may utilise either direct current or alternating current for operation. An example of a direct current operated servo-actuator is shown is fig. 7.2. It consists of a motor which is coupled to the flight control system via an electro-magnetic clutch, a gear train, and a sprocket and chain. The servo-actuator also carries a solid-state servo amplifier which amplifies the error signal transmitted by the attitude sensing transducer. Feedback is provided by a potentiometer, the wiper of which is driven by the motor.

Alternating-current-operated servo-actuators may be either of the two-phase induction motor type, or of the type using the principle of hysteresis as applied to the gyroscopes of certain attitude sensing elements. A two-phase induction motor type of servo-actuator (fig. 7.3) has its reference phase constantly supplied with 115 volts alternating current at a frequency of 400 Hz. The control phase is supplied by the output of the associated servo amplifier, the voltage varying from zero to 240 volts. The motor
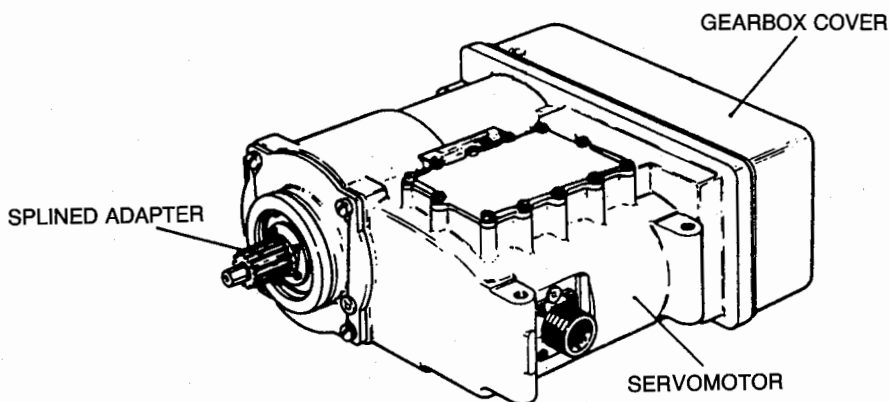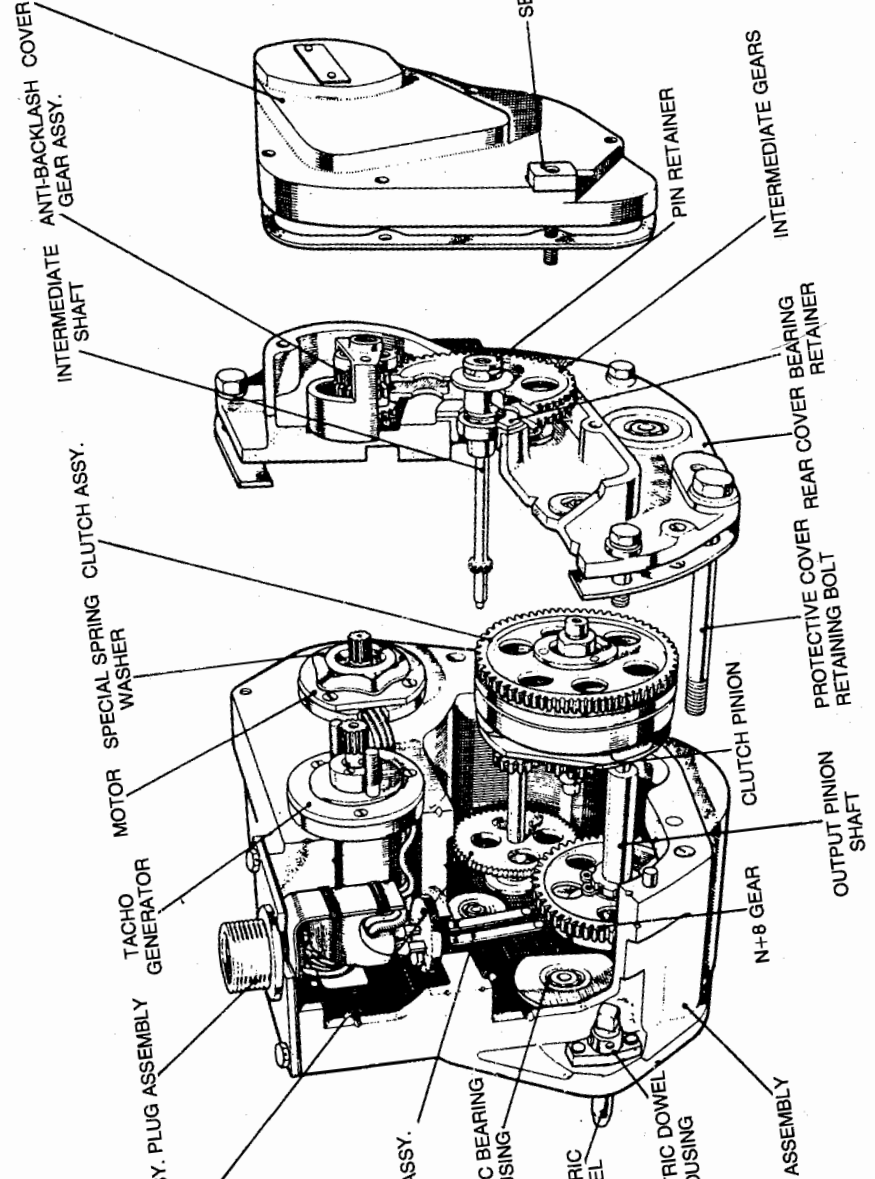
Fig. 7.3 Two-phase induction type servo-actuator.

drives an output pulley via a gear train, and an electromagnetic clutch, the pulley providing the coupling between the servo-actuator and cable of the aircraft's flight control system. A CX synchro, and a device known as a tachogenerator, are also geared to the motor, their respective functions being to provide position and rate feedback signals.

A servo-actuator utilising a hysteresis motor is shown in fig. 7.4. It operates on the relevant magnetisation and de-magnetisation principles, but whereas in conventional hysteresis motors the stator is directly connected to a three-phase supply of 115 volts at 400 Hz to produce a unidirectional rotating field, the three-phase stator in the example of servo-actuator illustrated, is fed from a single-phase supply, and field rotation in either direction is obtained by splitting the phases by means of capacitors. The single-phase supply is connected to, or disconnected from, the stator by means of silicon controlled rectifiers (SCRs). Activation, or 'firing', of one or other SCR is achieved by connecting the firing circuit to those circuits supplying the command signals which determine the direction in which the stator field, and hence the servo-actuator, must rotate in order to apply corrective control. Coupling between the motor and the aircraft's flight control system is by means of a gear train and an electromagnetic clutch, and feedback signals are supplied by a tachogenerator coupled to the motor gear train.

Servo-actuator mountings

Servo-actuators, as we have already noted, are mechanically connected into the cable runs of an aircraft's primary flight control system. The method of connection is governed by such factors as the type of automatic system, and the type of primary flight control system with which it is to be used. In

ANTI-BACKLASH COVER

ANTI-BACKLASH COVER GEAR ASSY.

PIN RETAINER

INTERMEDIATE GEARS

INTERMEDIATE SHAFT

REAR COVER BEARING RETAINER

PROTECTIVE COVER BEARING

CLUTCH ASSY.

SPECIAL SPRING WASHER

MOTOR

TACHO GENERATOR

PLUG ASSEMBLY

CLUTCH PINION

OUTPUT PINION SHAFT

PROTECTIVE COVER RETAINING BOLT

N+8 GEAR

BEARING

DOWEL

ECCENTRIC DOWEL HOUSING

ASSEMBLY

ASSY.

Fig. 7.4 Servo-actuator utilising a hysteresis motor.

those systems which may be considered as more basic in concept, servo-actuators are designed to have a direct method of connection, i.e. their output shafts are fitted with either a cable drum or sprocket around which control cable or chain may be directly fitted. In the application of a number of systems, however, there is another factor which has to be taken into consideration; this relates to the disturbing effect which the removal and replacement of servo-actuators, under certain aspects of system maintenance, might have on the settings and adjustments made to a primary flight control system. In order, therefore, to minimise this effect a large majority of systems utilise servo-actuators which are designed for attachment to, and removal from, special mountings connected into the primary control system on a more permanent basis.
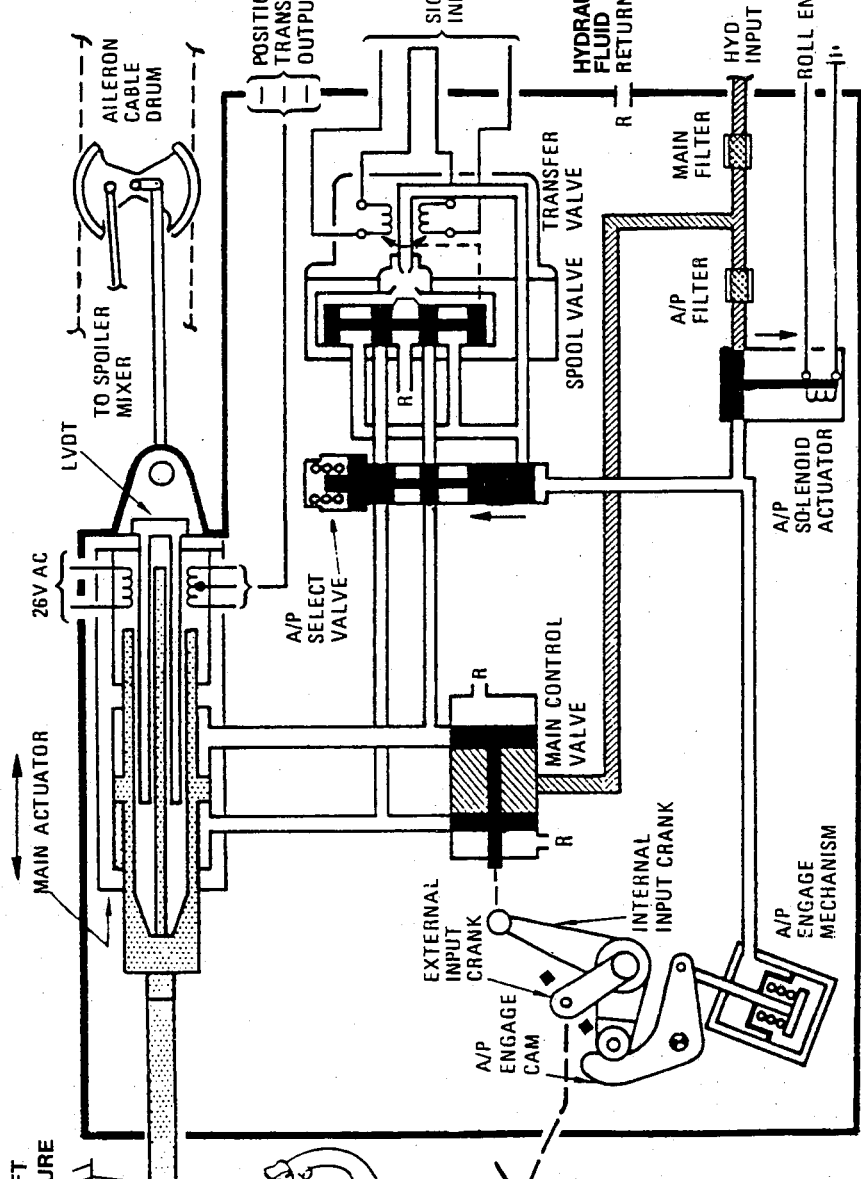
Power control units

In primary flight control systems of the hydraulic power-operated type, displacements of the control surfaces are effected through the medium of power control units; it is possible, as noted earlier, to directly apply automatic control command signals to these units and thereby eliminate the need for independent servo-actuators, as in the conventional forms of automatic control systems. This control concept is adopted in several types of high-performance aircraft for elevator, aileron and rudder control, the latter serving the role of yaw damping in particular.

An example of a power control unit used in an aileron control system is schematically illustrated in fig. 7.5. The principal components of the control unit directly associated with automatic control are (i) solenoid actuator, (ii) engage mechanism, (iii) select valve, (iv) spool valve, (v) transfer valve, and (vi) LVDT position transducer.

When the aileron channel is not engaged with the power control unit, the solenoid actuator is de-energised, and so the select valve and engage mechanism are isolated from the hydraulic system fluid supply. The transfer valve and spool valve are inoperative since they are isolated from the main actuator by the select valve. The cam spring of the engage mechanism is extended to release the cam so enabling the pilot to move the control wheel, and through the medium of the external and internal input cranks, the main control valve piston is positioned to admit hydraulic fluid to one or other side of the main actuator piston. Since the actuator piston is secured to the aircraft structure, the pressure of the fluid causes the entire power control unit to move in the appropriate direction. The unit itself, being connected to the aileron cable drum, will therefore displace the ailerons and also the wing spoilers in this case. In other words, the system functions as a basic power-operated system (see also page 35).

As will be noted from fig. 7.5, the body of the main control valve will also move with respect to its piston which is mechanically linked to the pilot's

AILERON CABLE DRUM

POSITION TRANS OUTPU[T]

SIG[NAL] INPU[T]

HYDRA[ULIC] FLUID RETURN

HYD INPUT

ROLL EN[D]

TRANSFER VALVE

TO SPOILER MIXER

SPOOL VALVE

MAIN FILTER

A/P FILTER

LVDT

R

26V AC

A/P SELECT VALVE

A/P SOLENOID ACTUATOR

MAIN ACTUATOR

MAIN CONTROL VALVE

R

R

R

R

INTERNAL INPUT CRANK

EXTERNAL INPUT CRANK

A/P ENGAGE MECHANISM

A/P ENGAGE CAM

[...]FT [...]URE

...ower control unit.

control column; thus, the valve is repositioned to shut off the hydraulic fluid supply when the required aileron displacement is attained.

When the roll control channel is engaged, the solenoid actuator is energised to allow the hydraulic fluid under pressure to the engage mechanism so that as its spring is compressed the cam is rotated and locks the input cranks to the power control unit. This action also prevents the main control valve piston from moving with respect to its housing. At the same time, hydraulic pressure is admitted to the select valve, causing its piston to open the lines between the spool valve and the main actuator, and also the line to the electromagnetically-operated transfer valve. Automatic control command signals from the appropriate servo-amplifier are supplied to the coil of the transfer valve to deflect it to one or other side. This in turn develops a pressure differential across the spool valve, causing it to move to open one or other side of the main actuator piston to hydraulic pressure, via the appropriate line already opened by the select valve. As in the case of manual control, the power control unit will move bodily to displace the ailerons to the commanded position. Since the input cranks are locked by the engage cam, as the power control unit moves, the pilot's control wheel will also rotate; in other words, the system operates as a parallel type. Movement of the control unit also displaces the core of an LVDT transducer in the main actuator, to provide position feedback signals necessary for limiting and taking off control as commands are satisfied.

As mentioned earlier in this chapter, a PCU can also be controlled by a servo-actuator, and an example of this, based on that applied to the Boeing 767, is schematically illustrated in fig. 7.6.

The servo-actuator is in two main sections: a control section and an output section. The former section comprises two solenoid valves (SV1 and SV2), an electro-hydraulic servo valve (EHSV), a pressure regulator, servo-actuator piston, a dual detent piston assembly, and an LVDT. The LVDT provides feedback signals, corresponding to servo-actuator piston movement, to the flight control computer (FCC). The output section comprises internal and external output cranks which provide a mechanical interconnection with the appropriate flight control 'run' and main PCU. An LVDT is also actuated by the cranks to provide feedback signals corresponding to control surface position, to the FCC.

When the AFCS is disengaged (see fig. 7.6) valve SV1 is disengaged and so isolates the servo-actuator from the hydraulic pressure. The servo-actuator piston is centred by means of springs, the output crank moves with the flight control system when manually operated, and the internal crank is free to move within the servo-actuator piston. Movement of the output LVDT establishes a datum position signal that is used during AFCS synchronising prior to engagement of the AFCS. On engagement of the AFCS, the FCC provides a discrete d.c. signal to arm the valve SV1, causing it to supply hydraulic pressure to the EHSV (via supply line 'C'), valve SV2 and the pressure regulator.
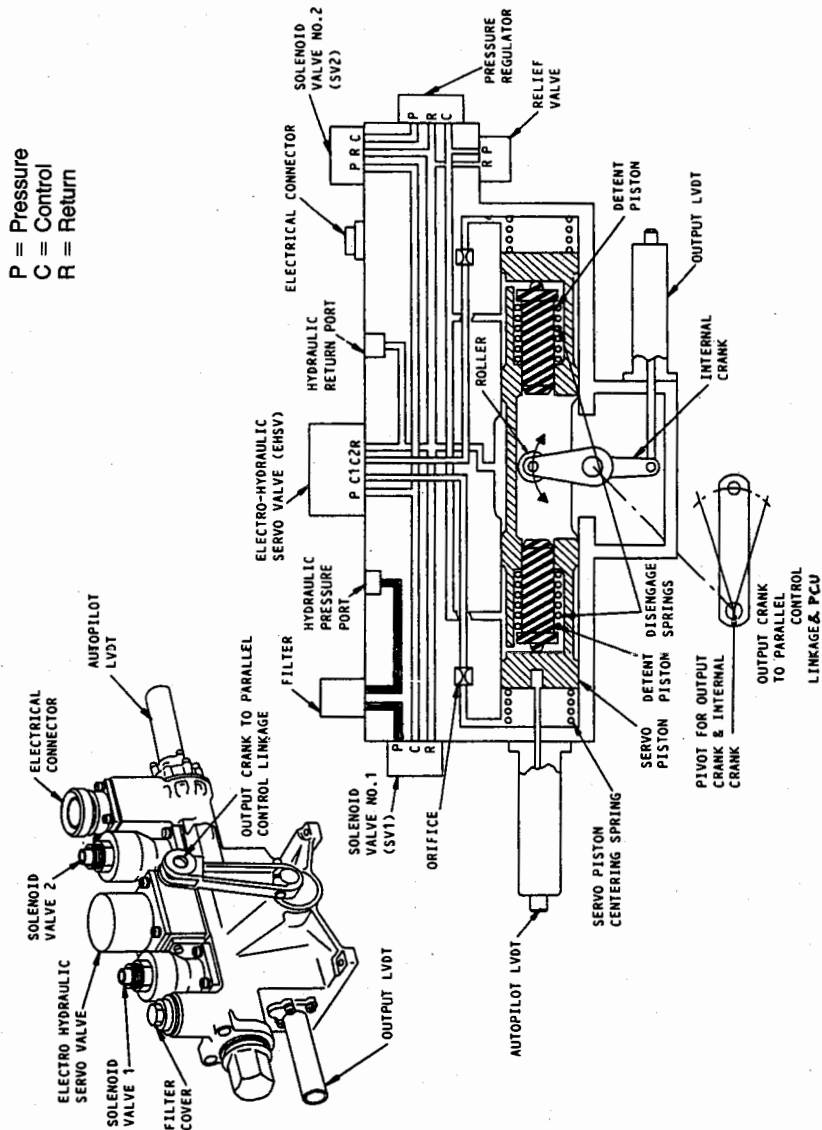
P = Pressure
C = Control
R = Return

ELECTRICAL CONNECTOR

SOLENOID VALVE NO. 2 (SV2)

PRESSURE REGULATOR

RELIEF VALVE

P R C

R P

HYDRAULIC RETURN PORT

DETENT PISTON

OUTPUT LVDT

ROLLER

INTERNAL CRANK

ELECTRO-HYDRAULIC SERVO VALVE (EHSV)

P C1C2R

HYDRAULIC PRESSURE PORT

DISENGAGE SPRINGS

DETENT PISTON

OUTPUT CRANK TO PARALLEL CONTROL LINKAGE & PCU

FILTER

SERVO PISTON

PIVOT FOR OUTPUT CRANK & INTERNAL CRANK

P
C
R

SOLENOID VALVE NO. 1 (SV1)

ORIFICE

SERVO PISTON CENTERING SPRING

AUTOPILOT LVDT

ELECTRICAL CONNECTOR

AUTOPILOT LVDT

OUTPUT CRANK TO PARALLEL CONTROL LINKAGE

SOLENOID VALVE 2

ELECTRO HYDRAULIC SERVO VALVE

SOLENOID VALVE 1

FILTER COVER

OUTPUT LVDT

Fig. 7.6 Servo-actuator/PCU control.

As with all other types of AFCS, the servo-actuator must be synchronised with the flight control system to prevent 'snatching' of controls, and in this case it is done by comparing the signals from the output and autopilot LVDT's within a synchronising circuit of the FCC. It there is a difference between signals, the FCC provides a signal to the EHSV commanding it to supply hydraulic pressure to the servo-actuator piston via either supply line C1, or C2. Thus, the piston is moved until the FCC detects that the autopilot LVDT signal is synchronised with that of the output LVDT.

When synchronising is complete and all other engage requirements are satisfied, the FCC provides another discrete d.c. signal to arm the valve SV2. This allows hydraulic pressure to the pressure regulator and to the detent pistons, causing them to close against their centring springs and onto the internal crank roller, thereby clamping or engaging it with the servo-actuator piston. The purpose of the pressure regulator is to reduce the system pressure to the detent position to allow manual override of servo-actuator commands.

Under automatically-controlled flight conditions, the signals commanding displacement of a flight control surface are supplied by the FCC to the EHSV which then supplies hydraulic pressure to one or other side of the servo-actuator piston via the output lines C1 or C2. The orifices included in the output lines restrict any pressure surge tendencies, and so ensure smooth movement of the servo-actuator piston and smooth application of control surface movement.

The servo-actuator piston and 'engaged' detent pistons are moved as an integral unit in the appropriate direction to drive the internal and output cranks. The cranks, in turn, provide the mechanical movement necessary for the PCU to hydraulically power the control surfaces to the commanded position.

Movement of the servo-actuator piston and cranks also actuates the autopilot and output LVDTs which provide the feedback signals for limiting servo operation (via the relevant circuit of the FCC and EHSV) and thereby limiting control surface movement as the control command is being satisfied.

Automatic trim control

In addition to the servo control of primary flight control surfaces, it is also necessary to provide methods of controlling the trim of an aircraft via its secondary flight control system (see also page 42). However, whereas in manually-controlled flight, trimming is usually effected about the three axes, under automatically controlled conditions it is generally confined to control about the pitch axis. In most cases it is accomplished by a separate trim servo-actuator coupled to the elevator trim tab system, and operating in parallel with the elevator servo-actuator. An example of a trim tab servo-
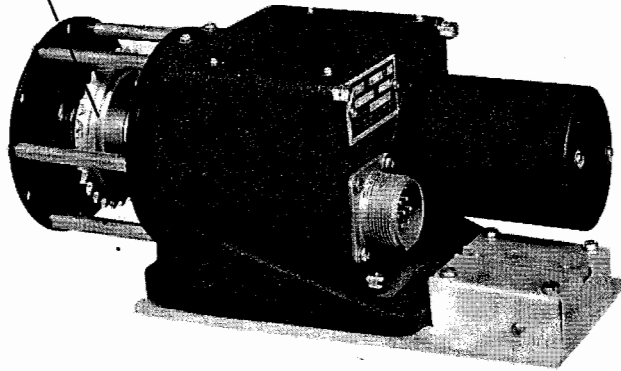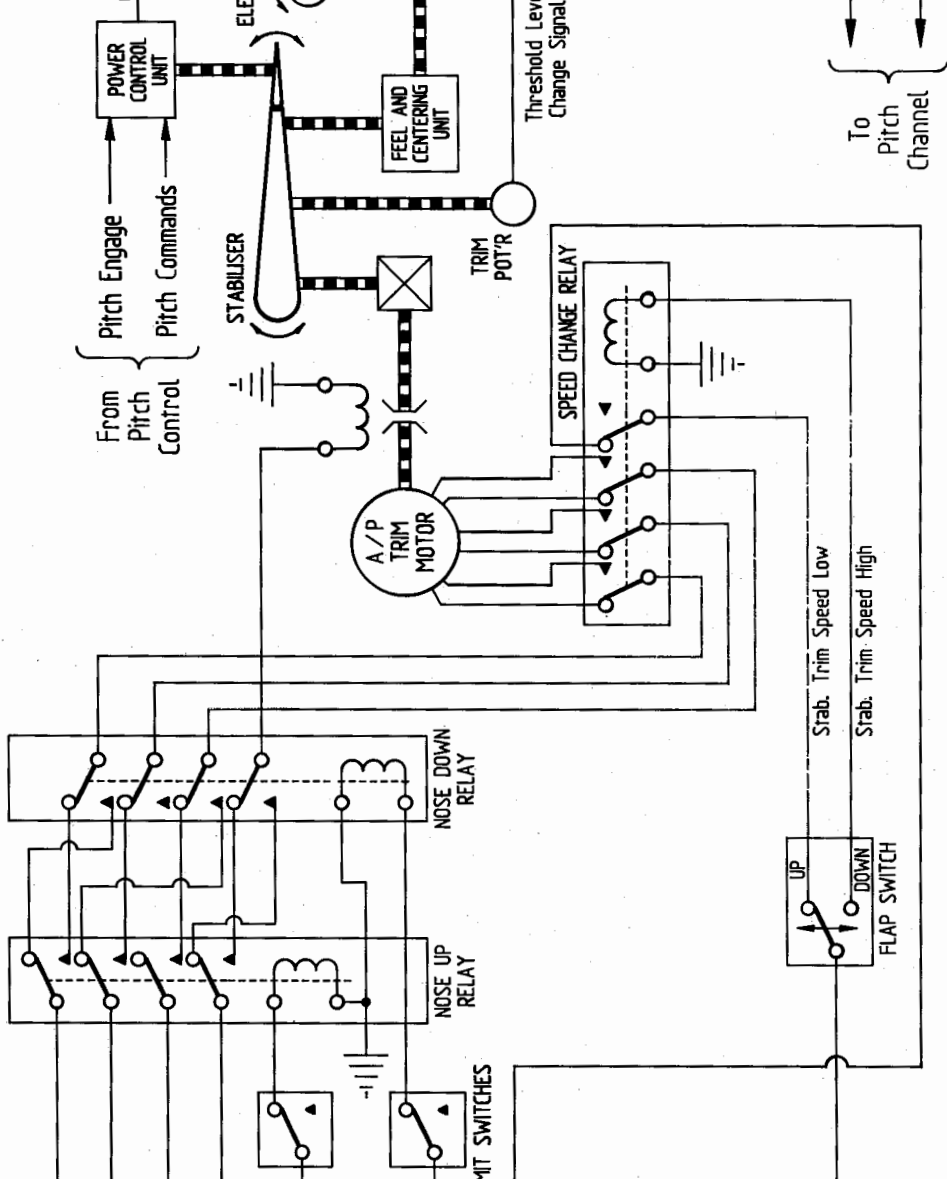
Sprocket for chain
drive to tab

Fig. 7.7 Trim tab servomotor.

actuator is shown in fig. 7.7. For those aircraft in which trimming is effected by means of a variable incidence horizontal stabiliser, a separate trim servo-actuator is coupled to the stabiliser. In cases where the stabiliser incidence is varied by hydraulic motors (e.g. Boeing 747 and 767 aircraft) the required automatic trim signals are used to control the flow of hydraulic fluid to the motors.

Figure 7.8 is a schematic representation of a control system adapted for trimming by means of a horizontal stabiliser of the type described on page 44. The trim servo in this application is a three-phase, dual-speed, dual-winding motor which operates in parallel with the elevator power control unit.

The 115 volts a.c. power is supplied to the windings of the motor through the contacts of the aircraft 'nose up' and 'nose down' relays, and also a speed change relay. The motor is coupled to the stabiliser jackscrew through an electromagnetic clutch and a differential gearbox. The 'nose up' and 'nose down' relays are energised by appropriate command signals from the pitch channel of the automatic control system. Energising and de-energising of the speed change relay is controlled by a flap position switch. In the aircraft trimmed condition shown in fig. 7.8, all three relays are de-energised and so the trim motor is de-clutched from the stabiliser and has no power supplied to it. The elevators which provide the primary control of the aircraft's attitude are locked to their hydraulic power control unit (see page 209) when the automatic control system is engaged.

Assuming that a climb command has been initiated, the command signal will be applied to the transfer valve of the power control unit causing it to apply 'up' elevator. At the same time, the pitch control channel will supply

ELE...

POWER CONTROL UNIT

FEEL AND CENTERING UNIT

Threshold Level Change Signal

To Pitch Channel

Pitch Engage

Pitch Commands

From Pitch Control

STABILISER

TRIM POT'R

SPEED CHANGE RELAY

A/P TRIM MOTOR

Stab. Trim Speed Low

Stab. Trim Speed High

NOSE DOWN RELAY

NOSE UP RELAY

UP

DOWN

FLAP SWITCH

...MIT SWITCHES

...matic stabiliser trim system.

a signal to the coil of the 'nose up' relay which, on being energised, completes an engage signal circuit (across the contacts of the de-energised down relay) to the trim motor clutch, and also completes the a.c. supply circuit to one set of trim motor windings.

The operation of the trim motor is governed as a function of stabiliser position versus elevator displacement from the neutral shift position, and this establishes threshold levels at which the stabiliser will start and/or stop trimming. The threshold levels can, in turn, be changed by a trim potentiometer which is mechanically connected to the stabiliser. If the 'up' elevator displacement already assumed exceeds the threshold level at which stabiliser trim will start, then the trim motor will drive the jackscrew in a direction that moves the stabiliser leading edge down. The movement of the stabiliser also moves the elevator feel and centring unit (see also fig. 1.33) and a neutral shift sensor. The sensor supplies a feedback signal to the pitch channel as the stabiliser takes up its new trim position. Since the elevators must also take up a compatible neutral position, as the aircraft responds to the required attitude change the pitch channel supplies a signal to the elevator power control unit transfer valve commanding a downward displacement of the elevators. A feedback signal from the elevator position transducer limits elevator displacement and by summing it with that of the neutral shift sensor the new neutral position of the elevators is determined.

Signals from the pitch trim potentiometer are also fed back into the pitch channel to change the threshold levels of trim motor operation appropriate to the new trim position of the stabiliser. The purpose of the limit switches shown in the trim signal circuits is to disengage the nose up and nose down relays, and thereby the clutch, in the event that the stabiliser is driven beyond its pre-set limits in either the upward or downward directions.
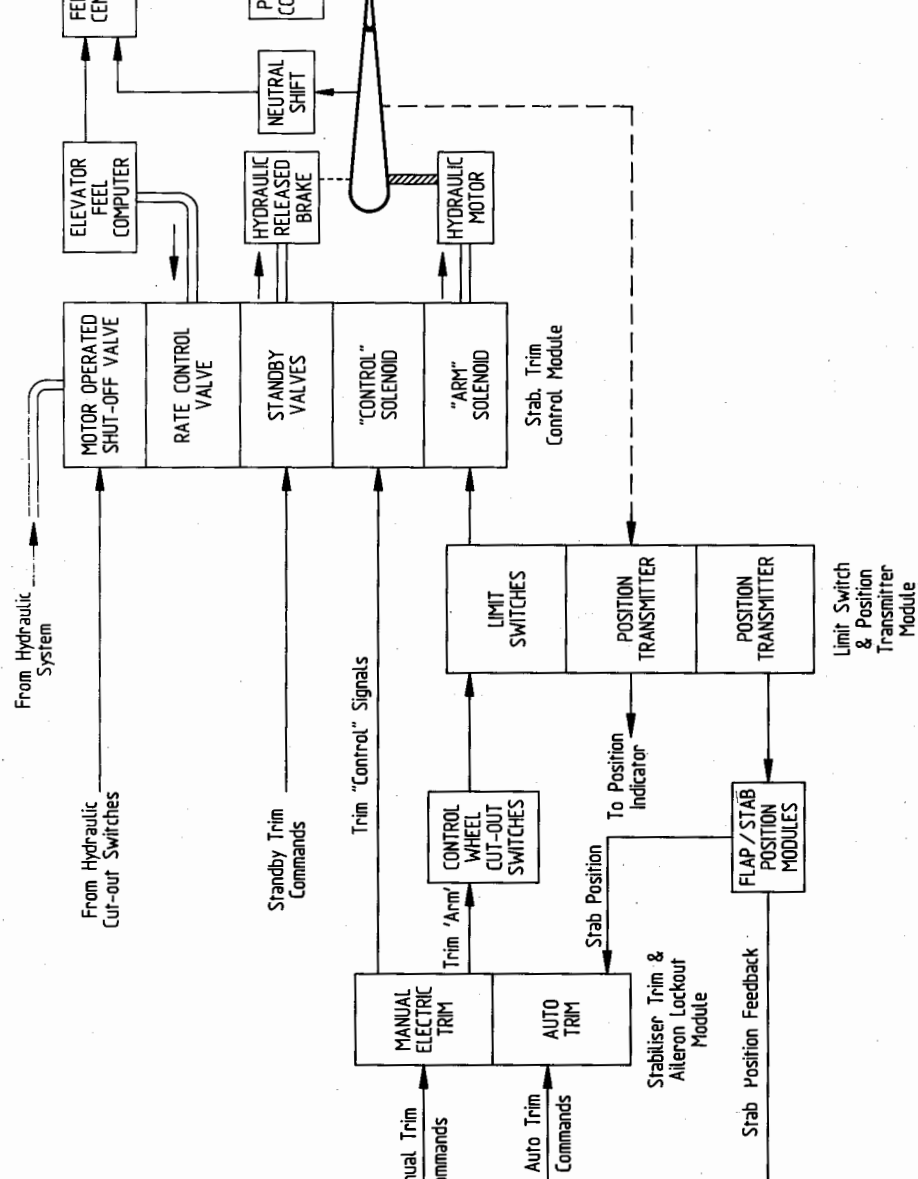
The speed change relay forms part of what is termed a flap compensation circuit, the purpose of which is described on page 219.

Figure 7.9 illustrates in schematic form the stabiliser trim system as used in Boeing 757 and 767 aircraft. In operation it is fundamentally the same as that just described, except that processing of command signals is based on more sophisticated digital computing techniques, and that hydraulic motors are used for positioning the stabiliser.

In certain types of small aircraft flight control systems, e.g. the Century III autopilot, automatic control of pitch trim is based on the principle of sensing the difference in tension between the 'up' and 'down' elevator control cables, when the elevator control is applied.

The trim system (see fig. 7.10) consists of a trim sensor, a trim servo and amplifier, and a switch mounted on the pilot's control wheel. The trim sensor consists of a fixed mounting plate with two adjustable contacts and a sliding bar with a common contact which is attached to pulleys that ride on the 'up' and 'down' elevator cables. The trim servo is a geared, d.c. reversible motor installed on the same mounting bracket as its solid-state circuit amplifier. The motor output is transmitted to the elevator trim tab

FEEL
CEN...

P...
CO...

NEUTRAL
SHIFT

ELEVATOR
FEEL
COMPUTER

HYDRAULIC
RELEASED
BRAKE

HYDRAULIC
MOTOR

MOTOR
OPERATED
SHUT-OFF VALVE

RATE
CONTROL
VALVE

STANDBY
VALVES

"CONTROL"
SOLENOID

"ARM"
SOLENOID

Stab. Trim
Control Module

From Hydraulic
System

From Hydraulic
Cut-out Switches

Standby Trim
Commands

Trim "Control" Signals

LIMIT
SWITCHES

POSITION
TRANSMITTER

POSITION
TRANSMITTER

Limit Switch
& Position
Transmitter
Module

To Position
Indicator

CONTROL
WHEEL
CUT-OUT
SWITCHES

Trim 'Arm'

Stab Position

FLAP / STAB
POSITION
MODULES

MANUAL
ELECTRIC
TRIM

AUTO
TRIM

Stabiliser Trim &
Aileron Lockout
Module

Stab Position Feedback

...ual Trim
...mmands

Auto Trim
Commands

...abiliser trim system (hydraulic motor operated).
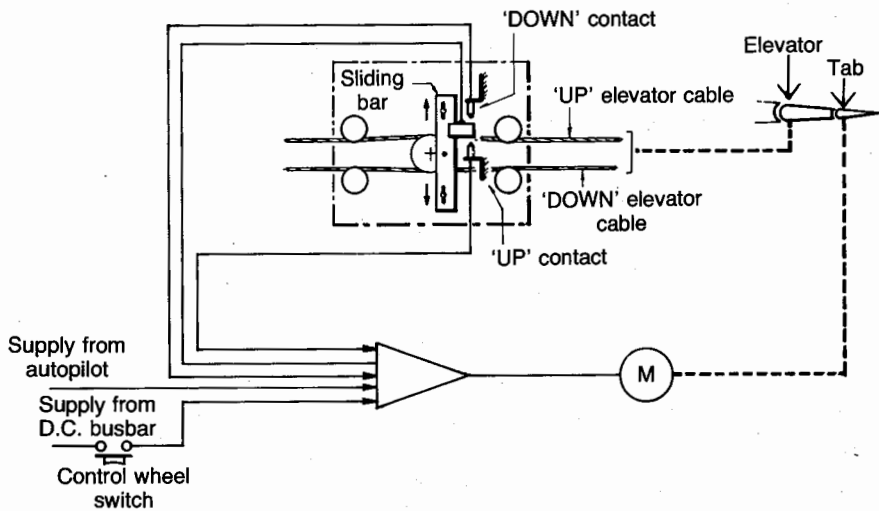
Fig. 7.10 Trim sensor system.

via a capstan which has a short length of cable, known as a bridle, wrapped round it; the ends of the bridle are fastened mechanically to the trim tab control cables. The amplifier supplies the power for the servo-actuator, and may be considered as 'dual-circuit', i.e. one circuit for running the servo-actuator in a clockwise direction, and the other circuit for running it in an anticlockwise direction. The direction and duration of rotation are determined by the trim sensor contacts.

When the AFCS is engaged, power is available to the trim servo system and, with the elevator in a neutral or streamlined position, both elevator cables will have equal tension, and the sliding bar and common contact of the trim sensor will be in neutral so that no trim signal is generated. If say, the elevator is deflected upwards, the force of the airstream acting on it creates a greater tension on the 'up' elevator cable and this causes the sliding bar and contact of the trim sensor to slide downwards and make contact with the fixed 'up' command contact. The appropriate circuit of the amplifier is therefore activated to cause the servo-actuator to move the trim tab until it is deflected sufficiently to relieve the tension in the 'up' elevator control cable. At this point, the sliding bar and contact of the trim sensor will have moved back to its neutral position thereby interrupting the circuit to the servo amplifier and of the servo-actuator motor. Thus, as per conventional trim tab operation, the elevator will be maintained in an upward position by a downward deflection of its trim tab (see also page 43).

The system operates in a similar manner with the autopilot disengaged, but in this case, the pilot supplies the power for servo operation by depressing his control switch.

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The elements of another example of an automatic trim system are shown in fig. 7.11. In this case, the output signal from the elevator servo-actuator is used as the appropriate command signal, its direction being sensed by up-trim and down-trim sensor circuits. These in turn operate relays which apply a signal to the trim servo-actuator to run it in the appropriate direction. The signal is pulse-width modulated, the pulse duration being a function of airspeed. The holding force, or elevator servo-actuator torque, resulting from elevator displacement is detected by a trim threshold sensor in terms of a motor current signal proportional to torque. The signal is passed through a time delay circuit which sets a time threshold of three seconds before trim servo-actuator operation takes place.

Assuming that the holding force is associated with an 'up' elevator displacement, the direction signal is applied to the down-trim sensor, and together with the time threshold sensor signal it is then applied to the relay driver circuit. After the appropriate time delay, the driver circuit output energises the 'down' relay, thus allowing the rate signal from the pulse-width modulator to pass to the 'down' winding circuit of the trim servo-actuator via the contacts 1 of the relay, and also contacts 1 of the de-energised 'up' relay. The trim tab is therefore displaced downwards at a particular rate thereby relieving the load on the elevator until the trim threshold sensor no longer senses a torque/current signal from the elevator servo-actuator.

It will also be noted from fig. 7.11, that the pulse rate signal to the trim servo-actuator also passes through the contacts of a trim transfer relay which is always energised when the autopilot is engaged. With the autopilot disengaged, the relay contacts are changed over so that operation of the trim tab servo-actuator is transferred to the pilot by means of a manually-operated trim control switch.

Flap compensation

When the flaps of an aircraft are operated there is inevitably a change in pitch attitude (see page 50) and under automatically-controlled flight this must be 'trimmed out'. It is usual therefore, to incorporate a flap compensation circuit within the command signal detection system. Circuit arrangements vary between types of automatic control system, but in general, a circuit is activated by passing a d.c. signal through the contacts of a switch or relay controlled by the flap system and, after processing, feeding it to the automatic pitch trim actuator. In a number of systems, the signal is also fed to a timer circuit which is set to correspond to a specific 'flaps in motion' time period.

In the system shown in fig. 7.8, flap compensation is accomplished by a switch that is actuated when the flaps are selected to the 'down' position. This causes the speed-change relay to de-energise and thereby connect the 115 volts a.c. supply to the trim servo-actuator via its 'fast' speed winding.
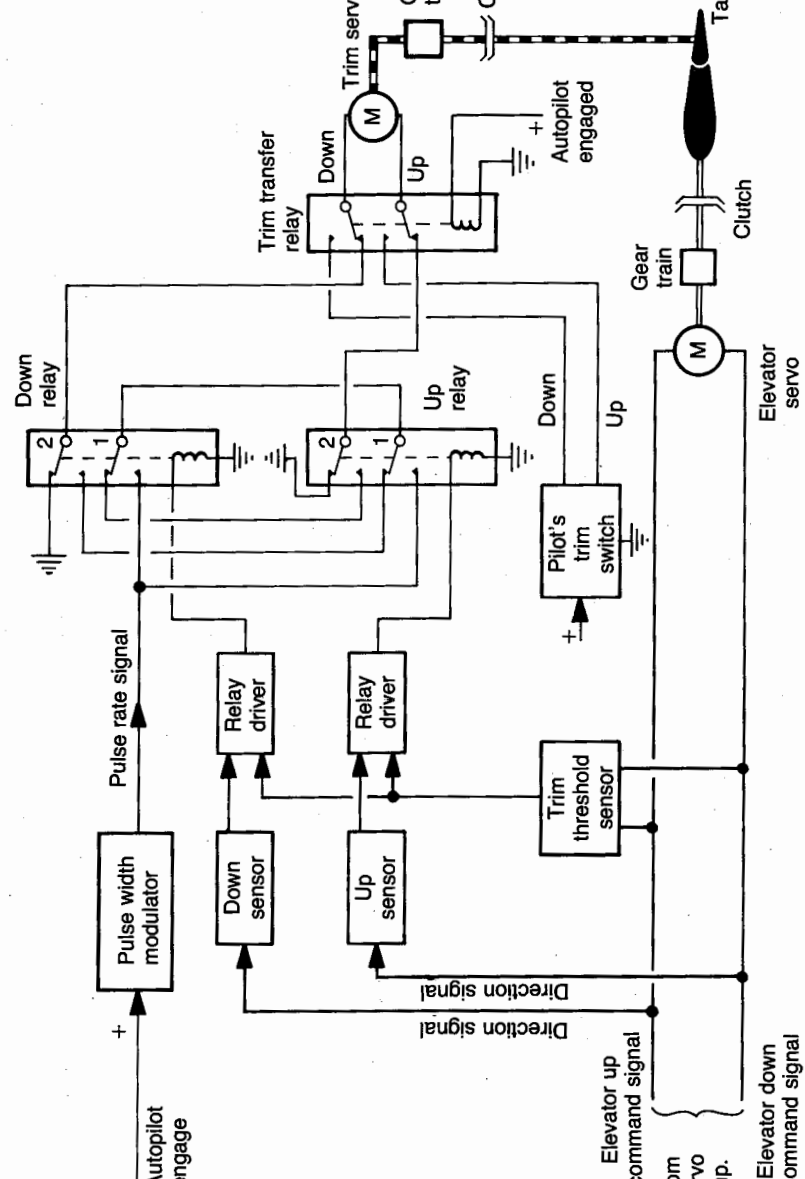
Automatic trim system.

Duplex servo-actuators

In some aircraft, the possibility of a hardover or runaway condition resulting from automatic control malfunctions, is prevented by utilising two independent control systems which displace control surfaces via duplex servo-actuators and differential gearing; a block diagram of such an arrangement is shown in fig. 7.12. The pitch and roll servo-actuators are of equal authority and torque, and their outputs are summed by their respective differential gearing; the yaw servo-actuator is of a standard single type with torque limiting.

When the commands to each motor of a servo are identical, as in the case of normal operation, the motions of both motors are also identical, so providing doubled authority to operate the appropriate control surface. If, however, a malfunction in one system occurs such that a hardover roll to port say, is commanded by that system, then it will turn the differential gear in the direction commanded. The other system, however, will at the outset, detect the undesired attitude change and will command its motor to rotate the differential in the opposite direction, with the net result that deflection of the control surface is prevented.

Each motor is coupled electrically to a sensor known as a speed monitor, which in turn, is connected to braking units. The purpose of the monitor is to identify a runaway motor, which it does within about two milliseconds, and then to apply a signal to the respective brake thus locking out half of the differential gear and enabling the good motor to drive the control surface through its half of the gearing. Since the servo power is halved, then any hardover risk in the remaining control system is reduced; this is further
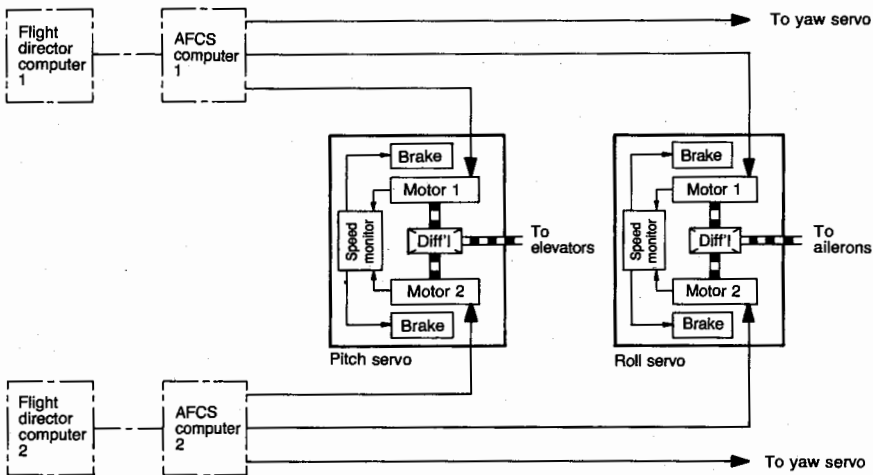
Fig. 7.12 Duplex servo-actuators.

ensured by the automatic engagement of a torque limiter on the active motor of the servo system.

Feedback

Assuming that an aircraft is subjected to a disturbance producing a pitch-up displacement, an attitude error signal is produced commanding the pitch channel servomotor, or power control unit, to apply down elevator. As the motor or control unit operates, it also drives a position transducer which produces a position feedback signal in opposition to the error signal thereby reducing it and limiting the extent of servo-actuator or control unit operation, and elevator displacement. The difference between the signals ensures proportionality of corrective control, and when the signals are equal, the error signal is reduced to zero and no further elevator control is applied.

As the aircraft responds to the downward displacement of the elevators, the attitude sensing element now detects this as a nose-down disturbance, and produces a command signal accordingly. The servo-actuator or control unit now applies control in the opposite direction to apply displacement of the elevators as the aircraft flies into the level flight attitude. As before, a feedback signal is produced to reduce the error signal until there is no further control, at which stage (and assuming that gain adjustments have been accurately made) the elevators will be in their neutral position, and the aircraft is again in the level flight attitude.

In order to ensure stabilisation of the complete feedback loop itself, and thereby eliminate oscillations or 'hunting', a rate feedback signal is also produced by a tachogenerator driven either by a servo-actuator, or by a motor contained within the servo-amplifier itself. The latter method is normally adopted for the automatic control of power-operated flight control systems. In each case, the signal is summed at the servo-amplifier input (see also Fig. 5.1).

Yaw damping

All aircraft, particularly those having a swept-wing configuration, are subject to a yawing-rolling oscillation popularly known as 'Dutch Roll' (see page 26) but different aircraft exhibit varying degrees of damping, i.e. the inherent tendency to reduce the magnitude of oscillation and eventual return to straight flight varies. A sudden gust or a short uncoordinated rudder deflection produces a yawing motion, and this, in turn, initiates the Dutch Roll oscillation. The vertical stabiliser and the rudder (if kept in a fixed streamlined position) develop opposing forces that tend to offset the yawing motion, but as a result of the inertias of the aircraft's motions,

stabilisation is regained in the form of a damped oscillation. As the aircraft recovers from the Dutch Roll, the magnitude of the oscillations gradually decreases. Thus, the Dutch Roll tendency may be comparatively mild in its effects and may, therefore, be tolerated without recourse to corrective action either manually or automatically.

For some aircraft, however, the natural damping of the Dutch Roll tendency is dependent not only on the size of the vertical stabiliser and rudder, but also on the aircraft's speed, the damping being more responsive at high speeds than at low speeds. It is, therefore, necessary in such cases for corrective action to be taken; such action requires displacement of the rudder in order to further assist the vertical stabiliser in its stabilising function, and is referred to as yaw damping.

It is usual therefore, for these aircraft to utilise a two-axis automatic control system with control about the third axis, i.e. rudder control, being provided by a sub-system called a yaw damper. The system is so designed that it can be operated independently of the automatic control system, so that in the event that the aircraft must be flown manually, Dutch Roll tendencies can still be counteracted. The system may be 'switched in' either by selecting a 'damper' position of the main engage switch on the automatic control system control panel, or by actuating a separately located yaw damper switch.

In those aircraft having upper and lower rudder sections (e.g. Boeing 747) a yaw damper system is provided for each rudder section.

Yaw damper systems

The operating fundamentals of yaw damper systems in general may be understood from fig. 7.13. The principal component of a system is the yaw damper coupler which contains a rate gyroscope powered directly from the aircraft's 115 volts a.c. supply, and the logic switching circuits relevant to filtering, integration, synchronising, demodulation, and servo signal amplification. Servo amplifier output is supplied to the transfer valve of the rudder power control unit; this unit differs from those used for aileron and elevator control (see fig. 7.5) in that it has an additional actuator (the yaw damper actuator) and does not include the automatic control system engage mechanism.

The actuator is also mechanically linked with a valve that controls hydraulic pressure to the main actuator, the piston rod of which is extended or retracted to position the rudder left or right respectively. A lever which forms part of the linkage provides for mechanical summing of yaw damper input to the power control unit, and an input from the rudder trim control system. Under manual control, inputs from the rudder pedals also pass through the summing lever to operate the main actuator.

Before the system is switched on, an integration circuit within the coupler unit ensures that the system is in synchronism; the yaw damper fail light is
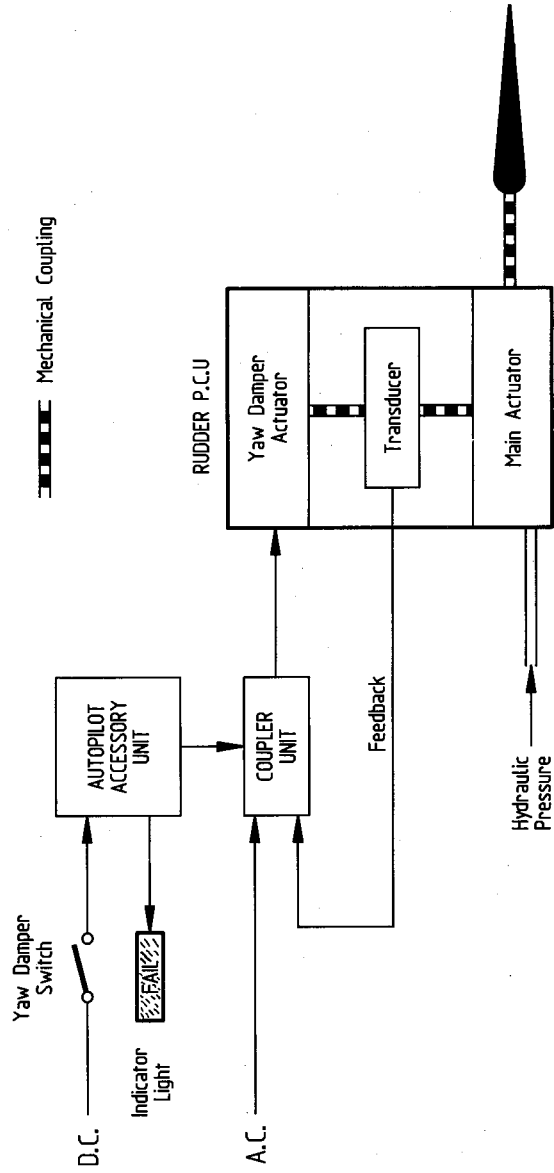
7.13 Yaw damper system.

illuminated at this time. When the yaw damper switch is at 'ON', d.c. power is applied to an engage relay in the accessory box, and on being energised, the relay completes a circuit to the actuator solenoid valve in the power control unit to allow hydraulic fluid to flow to the transfer valve; this is indicated by extinguishing of the fail light.

Whenever a yawing motion of the aircraft occurs, the rate gyroscope will be precessed and its sensing element will produce an a.c. output appropriate to the direction and rate of movement. The signal passes through a filtering circuit that discriminates between frequencies associated with flexing of the aircraft's fuselage, and with normal turns, so that only those frequencies related to Dutch Roll are allowed to pass. The signal is then demodulated and amplified, and applied to the transfer valve. The signal also passes through a circuit contained in the airspeed sensing section of an air data computer and is varied as an inverse function of airspeed. The circuit may be of the potentiometric type, or of the switched gain type; the latter being used particularly in digital air data computers.

The transfer valve is operated in the same manner as that described on page 211, but in this case, it directs fluid under pressure to the additional yaw damper actuator. Movement of the actuator piston is transmitted to a control valve in the main actuator which, through its piston rod, then moves the rudder in the required direction. The yaw damper actuator piston also positions the core of the LVDT to produce a position feedback signal to cancel the rate gyroscope input when the actuator piston rod has moved the required amount. The feedback signal is also supplied to a position indicator, the display element of which moves left or right appropriate to the direction of control applied by the yaw damper actuator.

When yaw oscillations have been damped, a constant signal passes from the LVDT to an integrator in the coupler unit; the integrator output then builds up to assist the LVDT in centring the yaw damper actuator, and thereby returning the rudder to its neutral position. The rudder pedals are not displaced during yaw damper operation, i.e. the system is of the series-connected type.

A two-position test switch is provided to simulate the effects of oscillations, and when operated torques are applied to the rate gyroscope causing it to apply left or right rudder as appropriate, to the power control unit. Movements of the position indicator display element provide for monitoring of the position feedback signal from the LVDT.

In order to compensate for differences in aerodynamic damping which in some aircraft can arise between the landing flaps down and flaps up conditions (see also page 51) the yaw rate gyroscope output signal is also passed through a gain change circuit controlled by a relay that is operated by a flap position switch. The control is such that energising of the relay by-passes a resistance so as to produce a faster rate of response when the flaps are down.

An automatic flight control system may be used in all modes with

the yaw damper system engaged; however the associated interlock circuit prevents the use of the control system when the yaw damper is disengaged. When the control system is operating in the localiser mode, the yaw damper is supplied with signals from the aileron control channel through a cross-feed circuit.

Torque limiting

In flight, particularly where high rates of control are to be produced, the movement of the flight control surfaces can result in loads which may impose excessive stresses on the aircraft structure. It is necessary, therefore, under automatically-controlled flight conditions, to safeguard against such stresses, and furthermore to safeguard against a servosystem 'runaway' condition which would cause control surfaces to be displaced to their maximum hardover positions. Such safeguards are implemented by limiting the torque applied by the servo-actuators, and also by allowing them either to slip, or to be completely disengaged, in the event that preset torque limits are exceeded. The methods adopted usually depend on either mechanical, electrical or electromechanical principles.

Phase advance

Phase advance refers to a method adopted in some automatic flight control systems as a means of providing powered control that will ensure maximum damping of an aircraft's response to the applied control. In order to understand why this is necessary, the response of an aircraft to disturbances should be considered.

Assuming that an aircraft is disturbed in pitch, inertia will momentarily carry it forward in the displaced attitude. The airflow over the aircraft acts on the tail of the aircraft and, because of the longitudinal moment arm and natural damping, a turning force about the centre of gravity is produced causing the aircraft to return to its normal attitude. Displacement of the elevators in the appropriate direction increases the turning force and therefore causes the aircraft to respond more quickly. Thus there are, in effect, two forces available for restoring an aircraft to its normal attitude; the natural damping force, and the additional force created by the use of flight control surfaces.

The damping force is a variable depending on the aircraft type and, in some cases, it is of such a low magnitude that the aircraft in responding to control surface displacement may oscillate about a mean attitude. The effect as appropriate to a disturbance producing a pitch-up displacement is shown in fig. 7.14 *a*. At point 'A', the displacement is produced and down-elevator is applied to correct the disturbance. The aircraft responds to the

a Aircraft disturbance and control movement

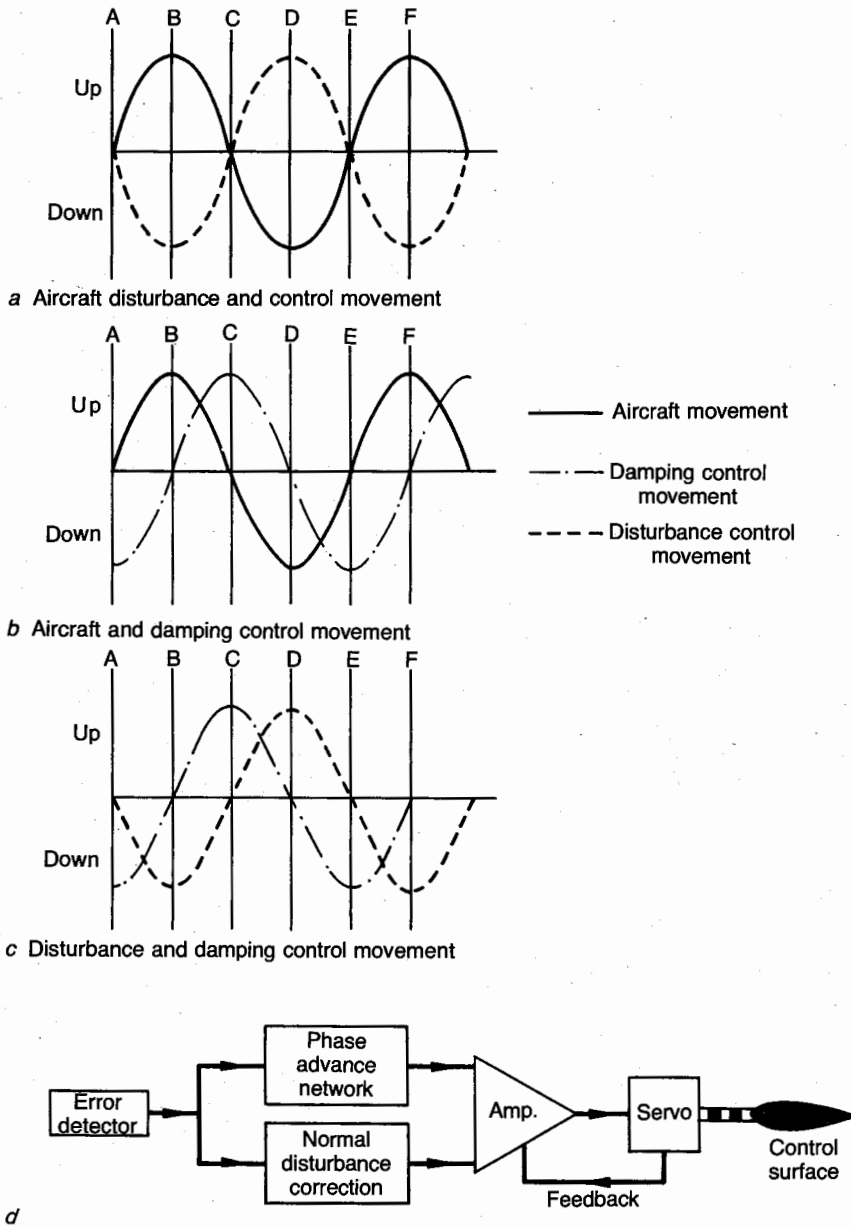b Aircraft and damping control movement

c Disturbance and damping control movement

d

Fig. 7.14 Phase advance.

elevator displacement (points 'B' to 'C') and during this stage elevator control is progressively taken off, until at point 'C' the normal attitude has been regained with zero control applied. The inertial effect of an aircraft with low natural damping causes it to overshoot the normal attitude, while up-elevator is being applied for correction purposes (points 'C' to 'D'). At point 'D', the aircraft is responding to up-elevator control and is once more approaching the normal attitude.

If the oscillation is to be prevented, then it will be appreciated that during the stage 'B' to 'C' it is not sufficient to take off the applied control, but that opposite control should be applied as the point 'C' is approached, as shown in fig. 7.14 b. Thus, at point 'A' the control surface is moved to a maximum value to counteract the pitch-up displacement, and from this point to point 'B', control is being taken off and the rate of change of aircraft attitude is decreasing. As the aircraft starts returning to its normal attitude (point 'B') so elevator control is taken off completely, and is applied in the opposite direction to prevent overshoot (points 'B' to 'C').

If the disturbance control and the damping control are referenced to the same time base then, as will be observed from fig. 7.14 c, the damping control is advanced in phase with respect to the disturbance control. The required control for a given disturbance is, therefore, the algebraic sum of both controls, and in any given automatic flight control system they may be applied as components of the signals from the attitude error detecting elements as shown in fig. 7.14 d.

For explanatory purposes, the phase advance has been assumed to have a phase angle of 90°. It should be noted, however, that this angle will vary from one aircraft to another, and even from one control surface to another on the same aircraft, e.g. the pitch damping requirement on a given aircraft will be very different from the yaw damping requirement.

The phase-advancing methods adopted in automatic flight control systems vary between systems, but we may consider the example shown in fig. 7.15. In this case, two detector signals are fed into the pitch control channel servo amplifiers, one such signal being derived from a pitch rate gyroscope and therefore being proportional to the rate of pitch change, while the other signal is derived from a vertical-axis gyroscope and is proportional to pitch displacement. Both signals are combined to produce an algebraic response in which the aircraft control is phase-advanced with respect to the disturbance.

Trim indicators

In most automatic flight control systems, a trim indicator is provided, its functions being to indicate that signals are being supplied to the servo-actuators, whether in the engaged or disengaged condition, and also to indicate any out-of-trim conditions of the aircraft under normal operating
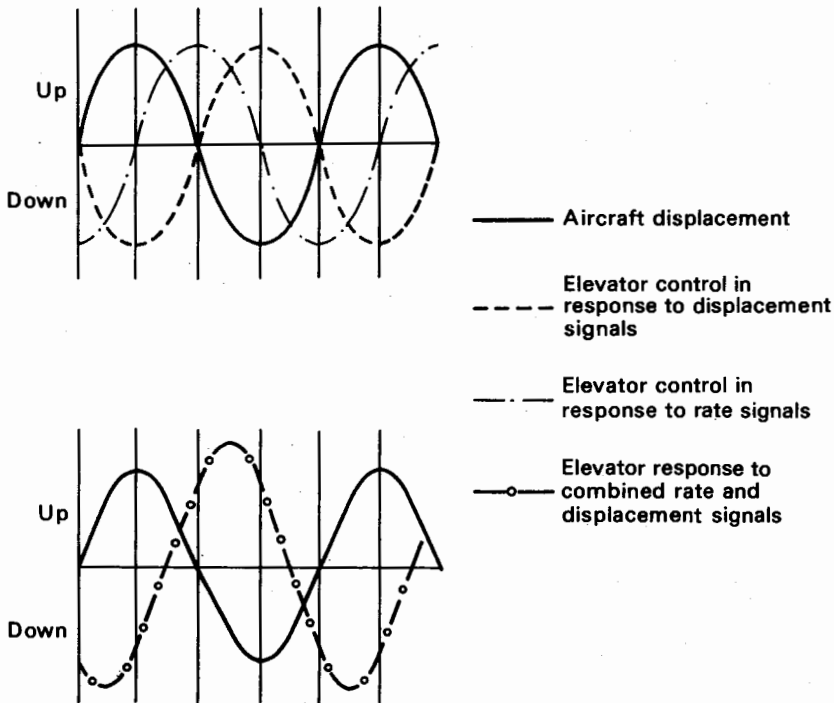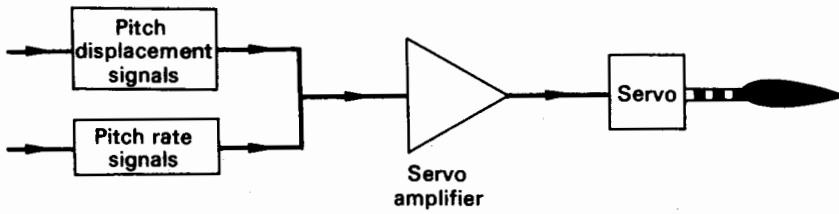
Fig. 7.15 Phase advancing method.

conditions of the control system. It performs these functions by monitoring the outputs from the rudder, aileron and elevator servo amplifiers, and by producing deflections of pointers from zero datum marks in response to signals supplied. Two examples of three-channel trim indicators designed for mounting on instrument panels are shown in fig. 7.16. In some control systems, monitoring of the elevator channel only is carried out, the appropriate indicators being incorporated in the flight control panels.

The pointers, which symbolise the flight control surfaces, are actuated by d.c. milliammeters, and are deflected each time servo amplifier signals are supplied to the servo-actuators. When the signals are balanced out as a result of servo control commands being satisfied, the pointers return to their
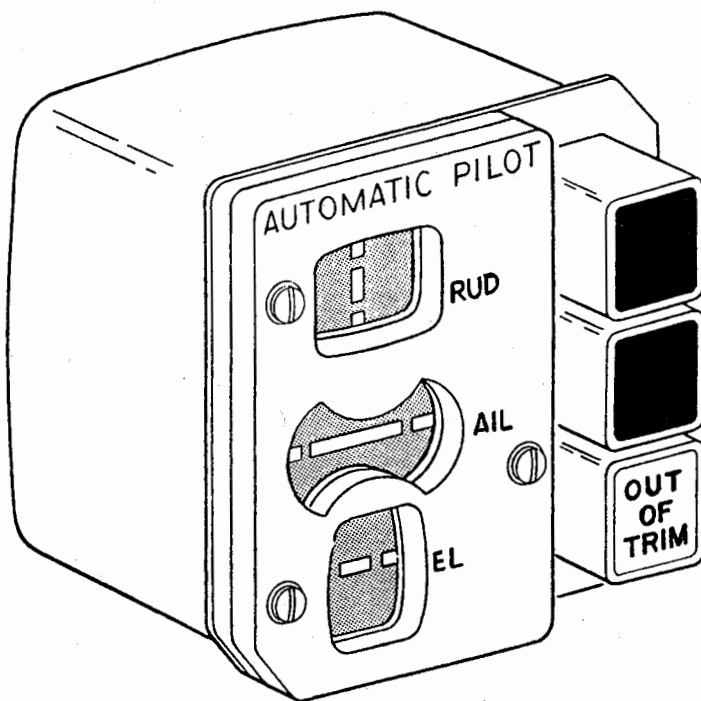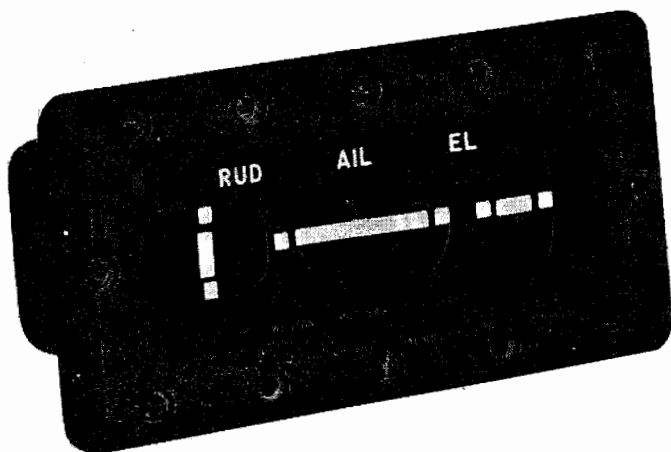
Fig. 7.16 Trim indicators.

zero 'trimmed' positions. In the event that a servo amplifier produces a continuous correction signal in order to maintain aircraft attitude, then the appropriate pointer will be continuously deflected as an indication that the aircraft is out-of-trim.

In systems which incorporate aileron and elevator position transmitters, signals from the latter are always present when the automatic control system is powered but disengaged. Thus, whenever the control column and/or the control wheel is moved, the associated trim indicator pointers will also be displaced. As an indication that each servo-actuator is engaged electrically-operated flags which display the word IN are provided in some types of trim indicator.

The indicator shown in the lower part of fig. 7.16, also incorporates an 'out-of-trim' indicator light which is associated with the elevator channel.

Mach trim system

In aircraft that are subject to the effect known as 'tuck-under' (see page 50) the pitch attitude change arising is compensated as a function of Mach number by a Mach trim system. As in the case of a yaw damper, it is a sub-system to an automatic control system in that it can be operational whether or not automatic control is engaged. A typical system is shown in block diagram form in fig. 7.17, and in comparing this with fig. 1.33 it will be noted that the actuating section of the system forms part of the horizontal stabiliser trim and elevator control system.

The principal components of the system are a trim coupler unit, an actuator, a test switch and a failure light. The coupler unit is electrically connected to an air data computer from which it receives signals corresponding to airspeed in terms of Mach number. It contains all the logic circuits necessary for the processing of the signals and their amplification before supplying them to the trim actuator.

The actuator consists of a 115 volts 400 Hz single-phase motor which drives a screw type shaft through a gear train. The shaft is connected to the stabiliser via the neutral shift mechanism, while the actuator body is pivoted to the elevator feel and centring unit. A brake mechanism supplied with 28 volts d.c. from the trim coupler unit unlocks the actuator to allow its movement when engaged and operating. Switches are provided to limit actuator motion at its fully extended and retracted positions.

At airspeeds below the value preset for the aircraft type (0.715M in one particular case) the Mach trim actuator is inoperative and its shaft is in the fully extended position. As speed increases and exceeds the set value a signal is supplied from the trim coupler unit to release the brake, and a speed signal from the air data computer is supplied to the motor which then rotates the screwed shaft. Since the shaft is connected to the stabiliser via the neutral shift mechanism the actuator body itself is traversed along the

ELEVATOR

STABILISER

MACH TRIM ACTUATOR

Input Crank to Control Valve

Power Control Unit

Position Transducer

Transfer Valve

Actuator Shaft & Neutral Shift Mechanism

Actuator Brake

Position Feedback

Limit Switches

Motor Drive

CX

M

Feel and Centering Unit

Neutral Shift Sensor

GT

COUPLER UNIT

Comparator Circuit

L O G I C

Command

Model

Mach Valid Signal

Test Switch

Indicator Light

Mach Speed Signals

Air Data Computer

1

2

Neutral Shift Signal

'UP' Elevator Command Signal

Position Feedback Signal

PITCH CHANNEL

Linkage and Cable System to Control Column

Mach trim system.

Fig. 7.18 Elevator correction vs Mach number.

shaft. Movement of the body also rotates the feel and centring unit, and assuming that the automatic control system pitch channel is not engaged, the main control valve of the power control unit will be directly actuated (see also page 209) and displacement of the unit will move the elevators up so that they counteract the 'tuck under' effect. The Mach trim actuator motor drives a CX synchro that provides a position feedback signal to the coupler unit. After demodulation it is supplied to the servo amplifier and when it opposes the command signal no further control is applied by the actuator motor. In addition to a pre-set 'start' value of Mach speed, there is also a corresponding value at which command signals are limited, e.g. the speed value of 0.715M referred to earlier has a corresponding limit value of 0.815M. The elevator correction versus the Mach number values quoted is shown graphically in fig. 7.18. The change in elevator correction commanded at 0.78M is produced by second stage amplification of the signal within the coupler unit.

When the aircraft's speed decreases, the command signal also decreases, and so the position feedback signal now predominates to drive the trim actuator back to the extended position. When this is reached, the circuit to the actuator motor is interrupted, and the elevators will at that time have been returned to their neutral position.

It will also be noted from fig. 7.17 that the air data computer provides two Mach speed signal data outputs to the trim coupler unit. These signals are of equal value and are supplied to a command channel and a 'model' channel. As the command channel is supplying a signal to the trim actuator, the 'model' channel furnishes a signal to a comparator circuit, in which it is compared to the feedback signal from the actuator position synchro. These

two signals should be of the same amplitude, but of opposite sign. If the signals are not within tolerance, the circuit to the actuator motor is interrupted and this is indicated by illumination of the fail indicator light.

The system operates in a similar manner when the pitch channel of the automatic control system is engaged, except that the power control unit now moves the elevators up as a result of a command signal being applied to its transfer valve from the pitch channel. This signal is balanced against a feedback signal from the power control unit's position transducer, and also a signal generated by the neutral shift sensor as it responds to movement of the feel and centring unit.

The test switch is provided for checking system operation and control movements by simulating the command signal input when speed is above the preset 'start' level, e.g. 0.715M. Since, under test conditions, the feedback signal from the Mach trim actuator is not equal to the output from the 'model' channel of the coupler unit, the comparator circuit will cause the fail indicator light to illuminate.

TEST QUESTIONS

1 A parallel-connected servo-actuator is one which:
   (a) moves the appropriate flight control surfaces without moving the pilot's controls.
   (b) is electrically connected in parallel with other servo-actuator.
   (c) moves both the appropriate flight control surfaces and the pilot's controls.
2 On what principles do servo-actuators generally operate?
3 In what manner are servo-actuators connected into aircraft flight control systems?
4 In what type of aircraft's flight control system can the use of conventional servo-actuators be eliminated?
5 Explain how a system referred to in Question 4 responds to automatic control command signals.
6 Describe the operation of a typical stabiliser trim servo system.
7 Explain how the sensing of control cable tension changes can be applied as a method of adjusting pitch trim.
8 What is the function of a flap compensation circuit when incorporated in an automatic pitch control channel?
9 What is the purpose of a yaw damper system?
10 What methods are adopted in a servo-actuator system to safeguard against excessive stresses being imposed on an aircraft's structure, and against a 'runaway' condition?
11 What provision is made in a servo control system to ensure maximum damping of an aircraft's response to the control applied?

12 What is the function of a trim indicator, and how is it electrically connected into the control channels of an automatic control system?
13 What effect is produced when the airspeed of certain types of aircraft is increased?
14 Briefly explain the operation of the system that automatically corrects the effect referred to in Question 13.

8
Automatic Control of Helicopters

Introduction

In general, automatic flight control systems for helicopters fulfil the same requirements as those designed for use in fixed-wing aircraft, and adopt similar operating principles. However, in view of the aerodynamic characteristics and manoeuvring capabilities that are specific to helicopters, there are appropriate differences, notably in the area of control actuation and trim systems. The operating fundamentals of some examples of these systems, together with some typical AFCS applications, form the subject of this chapter.

Artificial feel systems

As in the case of fixed-wing aircraft (see also page 34) the control systems of helicopters can also be of the hydraulically boosted type, and therefore require the incorporation of an artificial feel system. Typical systems are based on springs, clutches and trim motors. A typical trim motor system is shown in fig. 8.1.

Helicopter trim systems

In a helicopter, it is often necessary to make large changes in lateral and longitudinal controls, as well as yaw pedal position, and hold the controls away from their original position for long periods. The ensuing forces need to be quickly removed in more than one axis, and this has led to a unique type of trim, referred to as the *force trim* system. Typically, it operates by pressing a single button on the cyclic stick to remove forces in the cyclic (and pedals) while these are moved to a new position. When the button is released, the force gradients are re-established with the new location as the zero force point. Those helicopters which have complex AFCSs and autopilots will also normally have a 4-way trim switch (beeper trim). To avoid confusion, in this chapter these two systems will be called respectively, force trim button and beeper trim switch. A typical location is shown in fig. 8.2.

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Connecting Rod

Spring Package

Housing

Control Rod

Pilot Force

Spring Force

Untrimmed Position
Pilot Force Required

Trim Motor

Trim motor drives
spring housing to
reduce force

Force-Free Position
No Pilot Force Required
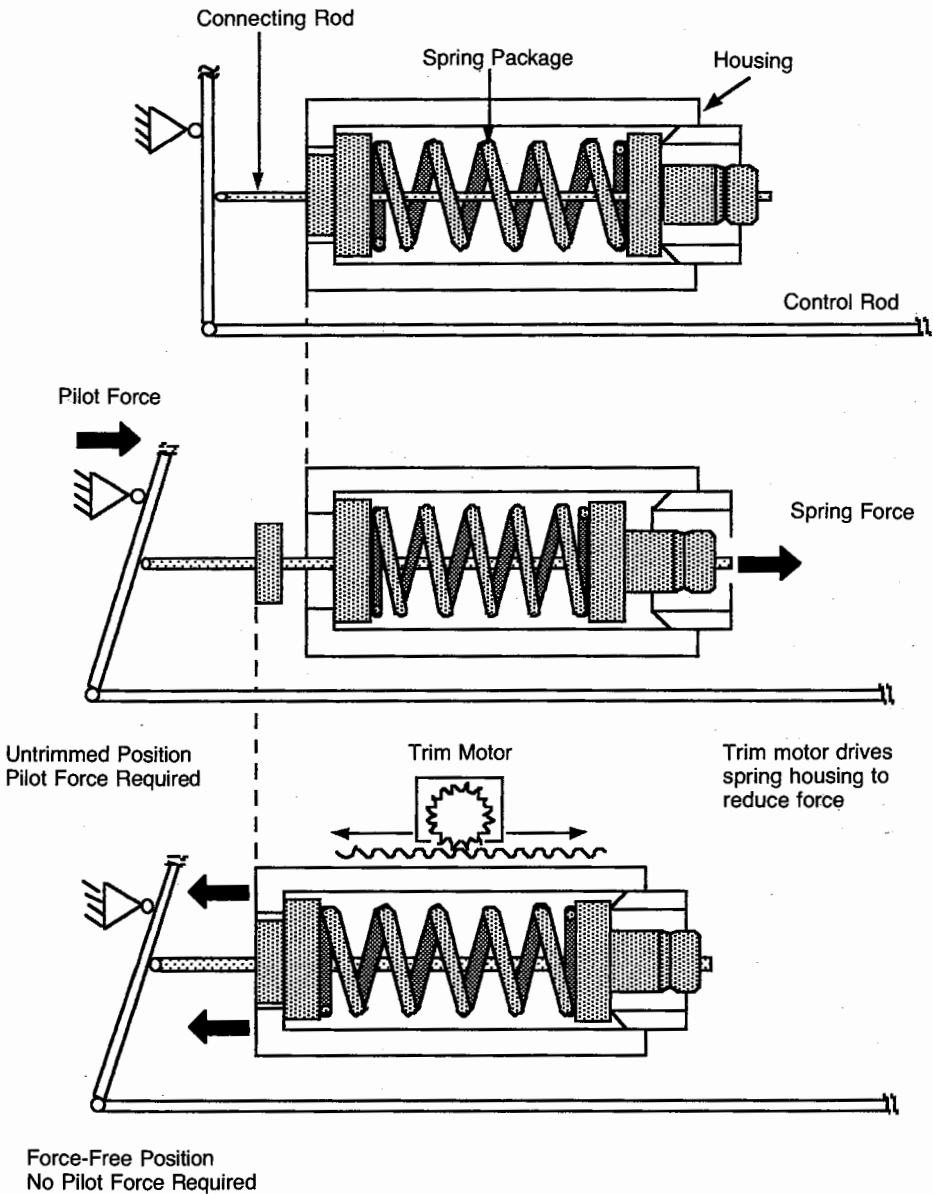
Fig. 8.1 Typical artificial feel system.

Actuators

There are basically two types of electro-mechanical actuators that can be
employed, namely, series and parallel; the choice is dictated by the com-
plexity of the AFCS. Correct operation of actuators is ensured by feeding
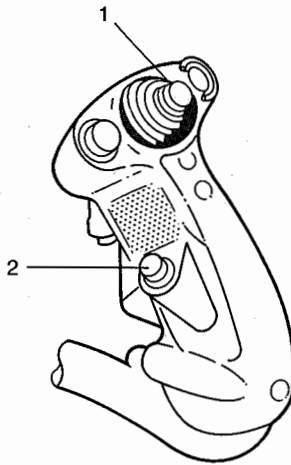
Fig. 8.2 Cyclic control grip: (1) Beeper trim switch (2) Force trim switch.

back position signals to the control system computer, and also, where appropriate, to an actuator position indicator (API) located on a panel in the cockpit. If the actuator has moved to almost the end of its travel, the pilot is able to adjust the cockpit control to re-centre it, and permit it to function in both directions.

Series actuator

The simplest form of a series actuator system is shown in fig. 8.3. In this configuration, the actuator alters the length of the control run causing a
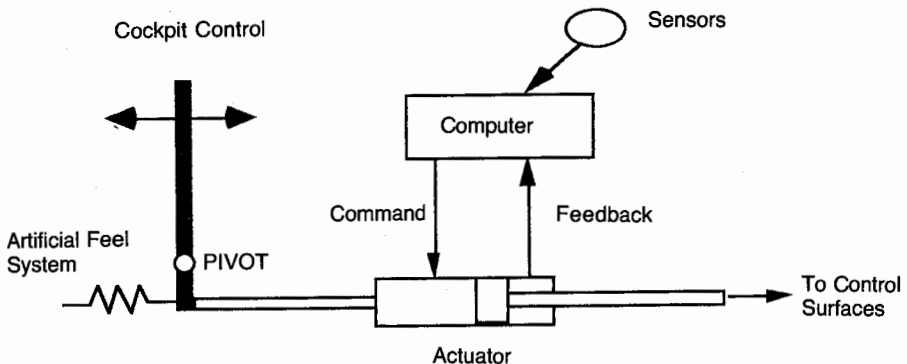
Fig. 8.3 A series actuator installation.
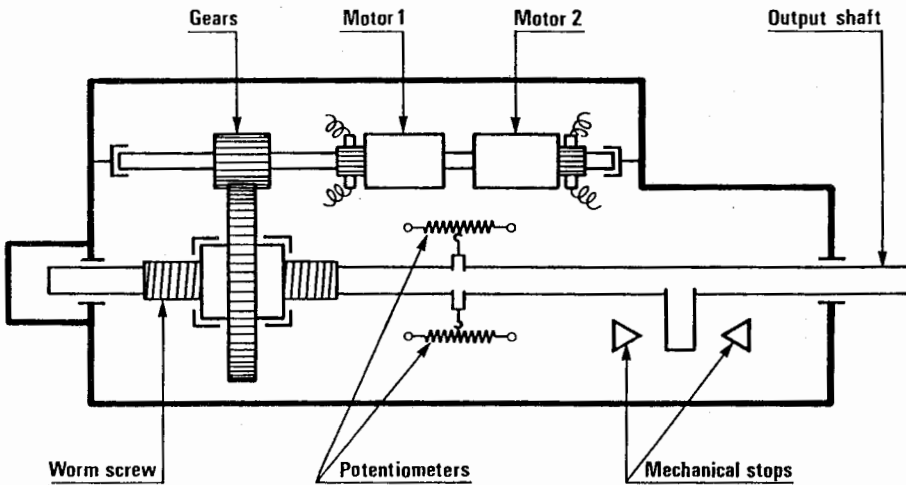
Fig. 8.4 Interior of series actuator.

movement of the pitch change mechanism at the main rotor. The cockpit control is prevented from moving by the artificial feel system. Figure 8.4 shows a more detailed view of the interior of a typical series actuator.

The characteristics of a series actuator system can be summarised as follows:

(i)   The actuator moves the control surfaces, but not the cockpit controls. If the artificial feel system is turned off, the actuator will move the cockpit control and not the control surfaces – obviously this would provide no stabilisation.
(ii) The actuator may become a rigid link when inoperative.
(iii) The actuator operates at high speed to counter small disturbances quickly.
(iv) The actuator authority must be limited (usually to approximately 10% of total control travel) so that failures (hardover or runaway) are not catastrophic. The cockpit control travel remaining following a movement of the actuator to one end of travel is adequate to control the aircraft.
(v)   A cockpit control position pick-off is not strictly required, but is often fitted with such a system.

When the cockpit control is moved, the cylinder of the actuator will move while the output shaft remains stationary. This allows the pilot to re-centre the actuator, which permits the system to continue functioning through a larger range. Due to its limited authority, such an actuator will prob-
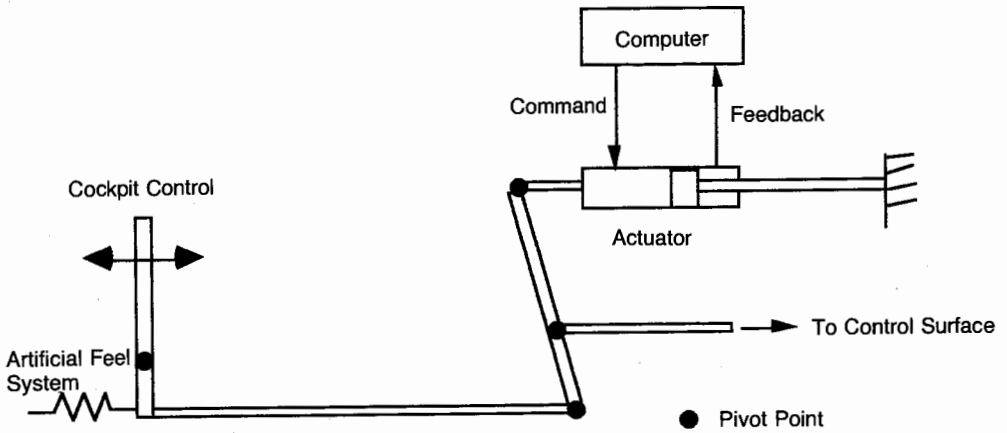
Fig. 8.5 Alternate configuration of a series actuator.

ably not be suitable for autopilot functions, which require larger control movements.

An alternative configuration for a series actuator is shown in fig. 8.5. Here the actuator is coupled to the control run via a differential-lever type of linkage. The actuator serves as an extending/retracting link that repositions the pivot point of the lever relative to the airframe. The output is the combined effects of the series actuator moving one end of the lever, and the pilot's input moving the other end. It should be noted that movement of the actuator in this configuration still does not produce movement at the cockpit control (unless the artificial feel is turned off).

Parallel actuators

The function of moving the cockpit control to re-centre the series actuator and retain its authority requires constant pilot attention to the API, repositioning and retrimming the cockpit controls. In more complex systems, this task is usually performed by a separate parallel actuator.

The parallel actuator is placed in the control run so that the cockpit control will move when the parallel actuator moves, thus making the pilot aware of the actuator operation. However, if the clutch is disconnected (normally by the force trim button or beeper trim) the actuator does not move the cockpit control, but only the clutch mechanism. A typical parallel actuator is shown in fig. 8.6, and interior detail is shown in fig. 8.7.

The characteristics of a parallel actuator system can be summarised as follows:

(i)   The actuator moves the cockpit controls as well as the control surfaces;
(ii) An artificial feel system is required;
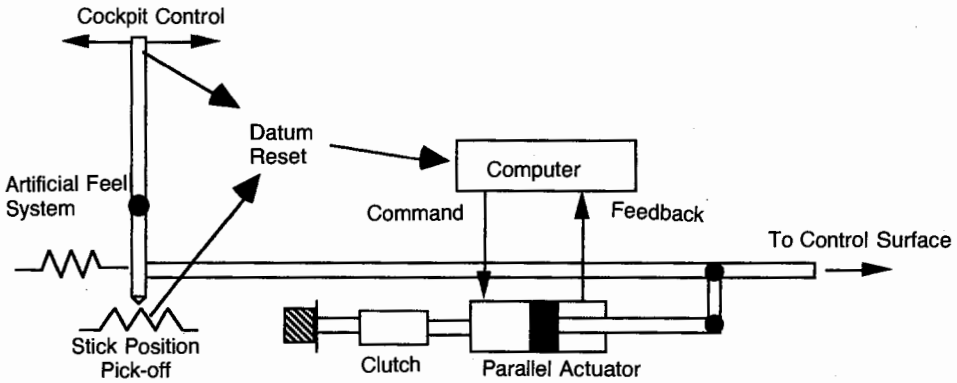
Fig. 8.6 Typical parallel actuator configuration.
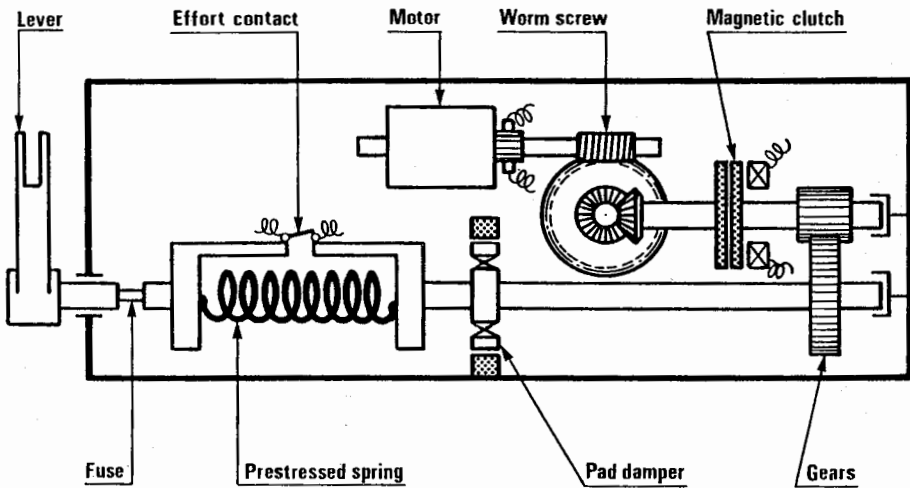
Fig. 8.7 Interior of parallel actuator.

(iii) It can be given full control authority (i.e., be able to move the cockpit control through its full range of travel) as the actuator operates at a low rate, and a failure should be easily detected before a hardover can occur;

(iv) A stick position pick-off is typically required.

In order to save weight and avoid duplication, it is a common practice to integrate an artificial feel system with the parallel actuator as shown in fig. 8.7.
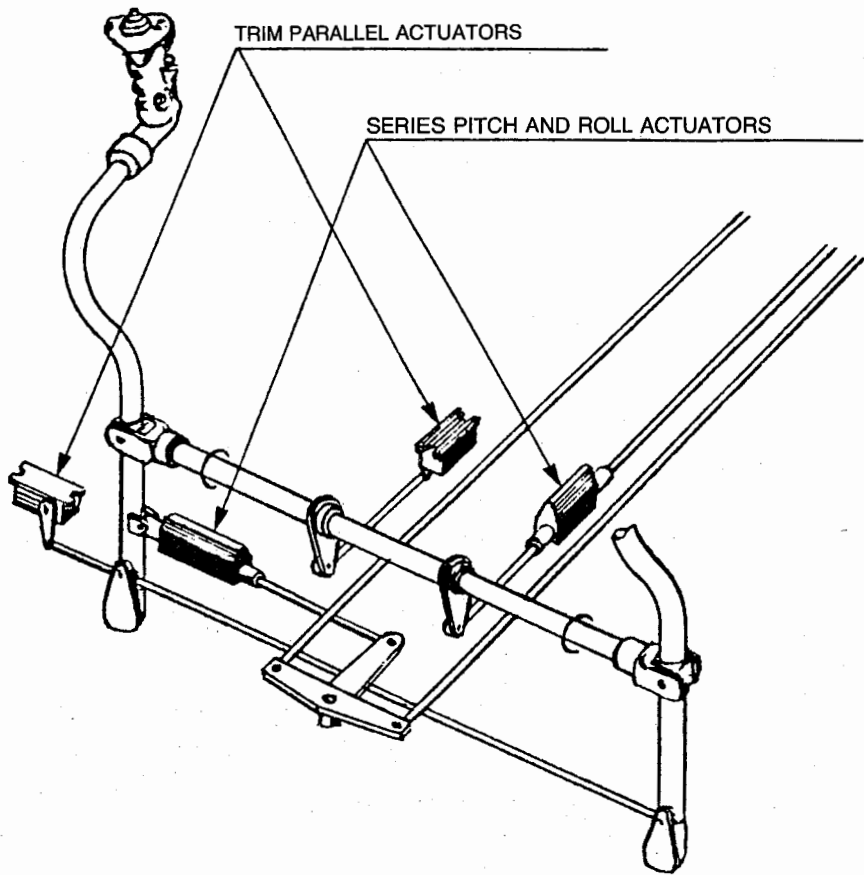
Fig. 8.8 Typical location of actuators.

Combined systems

A combination of series and parallel actuators can be used to provide both good short-term stabilisation, long-term attitude retention and autopilot capabilities. Typically, when the series actuator approaches the limits of its authority, the parallel actuator will move the cockpit controls to re-centre the series actuator. A typical installation is shown in fig. 8.8.

Stability augmentation systems (SAS)

Conventional helicopters are inherently unstable, particularly in pitch and roll, and can be tiring to fly for long periods. To overcome this, simple rate damping systems can be utilised and these will prevent any rates of pitch or
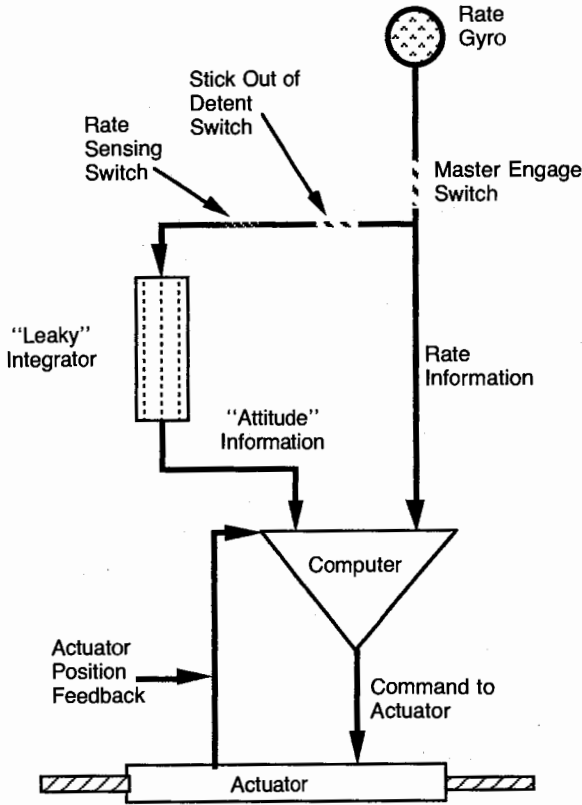
Fig. 8.9 Diagram of a Simple SAS – steady conditions.

roll from developing. However, and for reasons already discussed in chapter 2 (see page 78), these can also cause problems. Such problems were overcome with the development of stability augmentation systems (SAS), which provide corrective control inputs proportional not only to the rate of change of attitude, but also to the deviation from a datum attitude. The rate gyroscope is still the basic sensor, but the signals from it are integrated, using a 'leaky integrator', to simulate an attitude signal that can be used as a datum.

The operation of a simple SAS may be understood from fig. 8.9. With the system in the normal state (i.e. pilot not commanding a manoeuvre), the rate gyro sends a signal directly to the computer; and also sends information through a leaky integrator that produces a 'pseudo' attitude by integrating the rate signal. The present attitude signal is compared to that which existed 20–30 s previously (hence the reason for the term 'leaky' – it does not have long-term memory). If the flight path is disturbed, for example by a gust of wind, the rate signal produced by the gyro is used to

stop rate, and the leaky integrator produces an 'attitude' that is different from the attitude that existed 20–30s previously. Both error signals are sent to the computer, and this results in a correction signal being sent to the actuator. These signals stop the angular rate and then return the helicopter to the datum attitude. The 'leak' in the integtration means that the pseudo attitude will disappear after a short time and the system will regard the continually updated new attitude as the datum. Simply stated, the system will try to maintain the datum to which the helicopter is trimmed, and it will attempt to return to this datum if it is disturbed.

Such a system can overcome the problem of the nose dropping during a turn by feeding a derived bank angle signal to the pitch channel, and also by comparing the pitch attitude to the 'datum' pitch attitude.

Due to the errors in integrating very small signals from the rate gyro, such as in a steady condition with no gusts, the system will drift off the desired condition. Such systems can only be regarded as providing a limited duration 'attitude' hold. The ability of an SAS to provide a long-term attitude hold will largely be determined by (i) the sensitivity of the rate gyro, and (ii) the time constant of the integrator.

Pilot commanded inputs

Both the simple rate damping, and pseudo attitude hold systems, would interpret any movement commanded by the pilot as a disturbance, so a means to permit manoeuvring must be provided. Typically, the cockpit controls are fitted with a stick position pickoff or linear variable displacement transducer (LVDT) sensor. In fig. 8.10, the cockpit controls of an SAS have been moved to command a new flight condition. As soon as the stick is moved, two things happen:

(i) The stick position pickoff (LVDT or stick canceller) disconnects the signal from the rate gyro to the leaky integrator and stops the pseudo attitude signal;

(ii) A rate sensing switch in series with the stick position pickoff opens, also preventing the rate signal from going to the leaky integrator. It will remain open until the stick position pickoff indicates that the stick is in the datum (or trimmed) position, and the rate falls below a preset value, which is normally around 2°/s.

It should be noted that even when the pilot has moved the stick to a new position, rate damping is taking place (the rate gyro signal is going directly to the computer), but there will be no attempt to return to the datum that existed at the beginning of the manoeuvre. The system will only try to maintain an attitude when the helicopter is in a trimmed flight, i.e. manoeuvres are not being commanded. This rate sensing switch will only
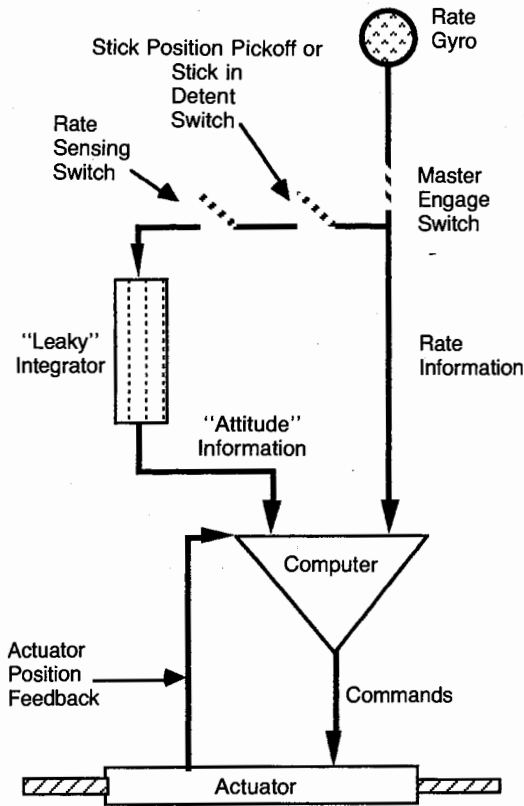
Fig. 8.10 Diagram of a simple SAS – manoeuvre commanded.

open after the stick has been moved. If the stick is in the trimmed position, and the rate sensing switch has closed, then it will remain closed until the stick is next moved out of the detent.

Retrimming

With an SAS system, the helicopter will hold the datum existing when the stick is in the detent, that is, no manoeuvring is being commanded. The helicopter will return to the datum attitude if disturbed by a gust or if the pilot moves the cockpit control and returns it to the same position. This can be accomplished by returning the stick to the zero force point (remember that the SAS needs an artificial force system to prevent the actuator moving the cockpit control instead of the control surface). After the manoeuvre, if the pilot uses the force trim button or beeper trim switch to re-position the stick, then a new datum attitude is established.

Vertical Gyro

Differentiator

q   θ

Computer

Actuator
Position
Feedback

Stick
Position
Pick-Off

Actuator

From Cockpit Control   To Swashplate
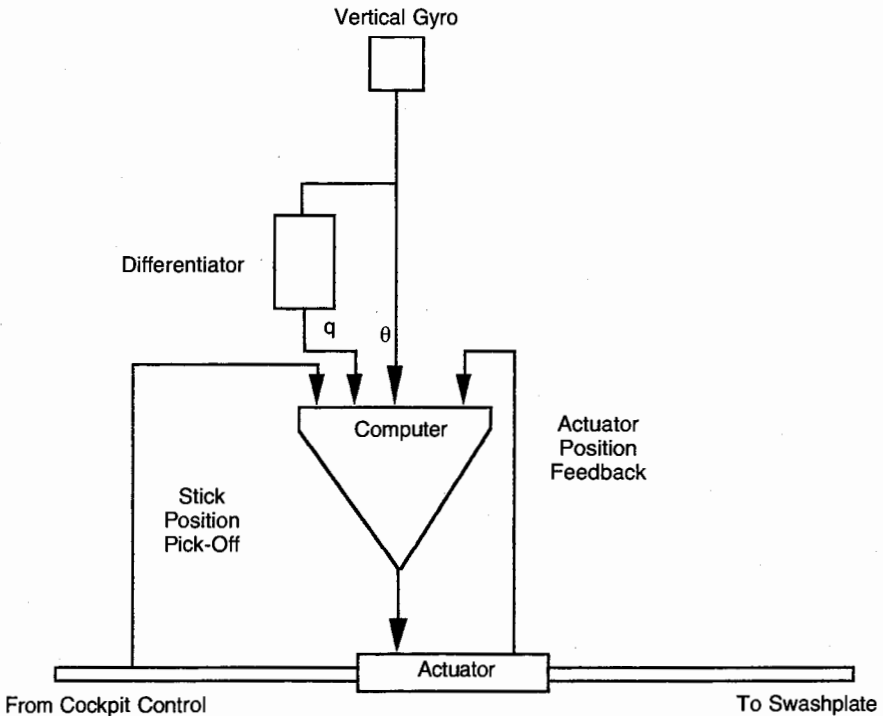
Fig. 8.11 Schematic diagram of a typical attitude hold system.

Stability and control augmentation systems (SCAS)

These systems are similar to an SAS except that instead of the stick position
pick-off cancelling the attitude hold or rate damping signal, they add a
feed forward of the control input to provide a larger initial response. For
example, if the cockpit control is moved, and the aircraft responds, then the
SAS will sense a disturbance and try to correct it; this will make the aircraft
response appear sluggish to the pilot. The 'control augmentation' part of
the SCAS senses the stick movement and feeds it to the actuator to produce
an input. This causes the helicopter to move, which is in turn picked up by
the sensors, and is then damped. The pilot senses a crisp response.

Attitude-based stabilisation systems

Attitude stabilisation systems are a refinement of the SAS system previously
discussed. The major differences are in the use of attitude gyros and
parallel actuators, and they form the basis for most complex AFC and
autopilot systems. It is important to note that with such systems, a means

of rapid damping provided by a series actuator, and SAS type circuits, is still required. The pitch channel of a typical attitude hold stabilisation system is shown in schematic form in fig. 8.11.

The use of a vertical gyro implies that a stable, accurate attitude signal is available. This means that commanded attitudes will be held with great accuracy for a long time, e.g. 1° nose-up will always be 1° nose-up, and not 1° difference from the signal that existed 30 s previously.

The requirement to manoeuvre will still exist, and again requires a cockpit control stick position pick-off or LVDT sensor. When the stick is moved out of the zero force (or datum or detent position), this sensor temporarily disconnects the attitude signal from the vertical gyro. In the same manner as the SAS, it will also open a rate sensing switch.

More definition problems

The effect on the helicopter, once the cockpit control has been returned to the zero force (detent or datum) position, will depend upon the detailed nature of the system. For example, if the system is designed to hold the attitude existing when the stick was in the detent, it will try to return to that condition. If 3° nose-up pitch attitude was the pitch datum, and the pilot desired to make a small, temporary correction, when the stick was returned to the zero force (detent or datum) position, and the pitch rate had fallen below the rate threshold (e.g. 2°/s) the helicopter would return to, and maintain, 3° nose-up. Such a system is considered to be of the 'wing leveller' type of automatic stabilisation equipment.

There are some systems that will attempt to hold the attitude that exists when the stick is returned to the zero force (detent or datum) position and the rate switch has fallen below its threshold. For example, if the commanded (or datum) attitude was 2° nose-down, and the pilot manoeuvred and returned the stick to the original zero force (detent or datum) condition, then when the rate dropped below the threshold (e.g. 1.5°/s), the pitch attitude would be 1° nose-down, and the system would hold 1° nose-down. In the strictly correct definition, such systems are called rate command attitude hold (RCAH) systems.

Attitude datum readjustment

In order to meet the requirement to fine-tune the datum, it is possible to press the force trim button, fly the helicopter to the new condition and release the force trim button to re-establish the control force and attitude datums. This is often inconvenient, especially as the force trim button will remove attitude hold functions in all axes. It may be desired to change the pitch attitude only very slightly, and leave the roll attitude and slip indicator ball as they were. Using the force trim button will cause the

attitude hold in all three axes to be lost, requiring the pilot to hold the roll and yaw controls motionless while adjusting in pitch. It is also possible to use only the beeper trim switch, which may directly re-datum the system in the axis where the switch is moved, and moves the stick. Some AFCSs carry this to an extreme by not moving the stick when the beeper trim is operated.

The problem is further compounded by the use of different control laws in different channels. For example, the pitch channel is often an RCAH system, while the roll channel may be an attitude command/attitude hold (ACAH) system. In practical terms, this means that to maintain an attitude away from trim, it is necessary to hold the control away from trim. In this case, to maintain a bank angle, it is necesssary to hold the cockpit control into the turn. Such systems have found favour in those helicopters that operate at low altitude over the water, in IMC and night conditions. If in a 30° angle of bank turn, the pilot becomes disoriented, he has only to release the stick, and the helicopter will return to 'wings' level attitude. In this case, the stick is used to command a roll attitude (normally it commands a rate of roll) hence the definition 'ACAH' system.

Heading hold and co-ordinated turns

The helicopter has an additional problem – one which centres around its ability to hover. Previous discussions have been about the pitch and roll axis; however, yaw is also an important part of the overall control of the helicopter, and methods of changing from low speed, ground-referenced operation to forward flight, airspeed-referenced operation need consideration.

Hover and low speed

In the hover and low speed regimes (below 40 knots of airspeed) one of the main tasks of the pilot is to maintain heading with the yaw pedals. In a pure (zero airspeed) hover, the helicopter has no stability in yaw – it will stay pointing in any direction, and small changes of power will cause difficulties for the pilot to maintain heading. With a wind (from any direction or at any speed) the aerodynamics, particularly around the tail rotor, become extremely complex, and in some cases, negative gradients of tail rotor (yaw pedal) position vs airspeed are found; this makes manual control of the heading nearly impossible. A yaw channel improves this situation tremendously, even if it only damps out rates of yaw. By preventing yaw rates from developing, the helicopter pilot's workload to maintain heading is reduced.

The only differences between the yaw axis and other axes are the sensors used and the means of disengaging the AFCS to permit manoeuvring. The sensors used in the yaw channel will typically include a flux detector to

sense magnetic heading, as well as rate gyros or attitude gyros. To sense the pilot's demands, various devices can be used. The least obtrusive to the pilot is a pedal force or motion sensor that senses when the pedals have been moved, or when a force greater than a pre-set threshold has been applied. More common is a micro switch on the face of the pedal, which is closed when the pilot places his foot on the pedal, indicating that heading hold is not wanted. Unfortunately, this means that to engage the heading hold, the pilot must take his feet off the pedals – something which few helicopter pilots like to do. Other systems use the force trim button on the cyclic to remove the pedal position signal from the AFCS circuit.

Forward flight

In forward flight, the yaw axis of the helicopter is nearly the same as that in a fixed-wing aircraft. A slight difference is that a fixed-wing aircraft will normally have zero sideslip, whereas a helicopter has some inherent sideslip. This prevents the use of a simple sideslip sensor for the yaw channel, and requires the use of a yaw rate sensor. In forward flight, the yaw channel is used in much the same way as in fixed-wing aircraft, namely to maintain slip indicator ball position.

Various turning modes may be incorporated, ranging from a simple system that provides artificial directional stability (via sideforce or yaw rate sensors), to complex automatic turn facilities geared to specific operational tasks. Perhaps the most interesting area of the yaw axis is in the cross-over from the low airspeed regime to forward flight and vice versa. To blend the two functions as the helicopter accelerates or decelerates between these regimes requires the introduction of an airspeed sensor into the yaw axis. The reason for this is that below 40 knots IAS, most of the flying is referenced to ground speed and alignment with the ground, and above that speed, the flying is referenced to airspeed.

Autopilots

A helicopter autopilot will usually consist of an autostabiliser, providing long-term datum holding, to which other special facilities may be added. There are two main types of autopilot:

Basic autopilot
A basic autopilot holds an external condition, such as airspeed, altitude, track, etc. This differs from the Automatic Stabilization Equipment (ASE) that provides datum holding to a helicopter attitude reference, such as pitch, roll and yaw. This is an academic definition used to differentiate parts of the system, e.g. some AFC systems feature airspeed hold without the option of attitude hold.

Automatic Flight Control

Operational autopilot
The operational autopilot performs manoeuvres, such as an ILS, transitions to and from the hover, and so on.

Facilities available through an autopilot

The facilities that can be provided by an autopilot are limited only by the sensors, computing capabilities and the control authority. Below are some of the more commonly available facilities and their principal characteristics.

(i)   *Altitude or height hold.* Altitude holds use signals derived from a baro-metric capsule, while height holds use a radar altimeter for their signals. Physically both systems could operate through the longitudinal cyclic but more commonly they operate through the collective control. The cyclic channel actuation is the simpler system, but it is only usable above $V_{min\ power}$, and cannot be used simultaneously with an airspeed hold. For those helicopters that require height hold, especially at low speed, the collective based system is the only suitable method.

(ii) *Radio radar altitude hold.* A vertical accelerometer may be used to smooth and shape the vertical response of the height hold channel. A means of averaging the radar altitude signal has also been used to provide a mean sea/ground level reference. This is particularly needed when operating over waves to prevent the radar altimeter from following the waves or getting out of sequence with them.

(iii) *Airspeed or groundspeed hold.* Airspeed hold is provided through the longitudinal cyclic in forward flight. Groundspeed hold and airspeed hold in the low airspeed region will require longitudinal and lateral cyclic control, and a multi-axis sensor. Because of rapid and large changes in sensed signals in this area, it is not uncommon to blend in acceleration signals as well.

(iv) *Programmed manoeuvres.* There are many examples of programmed manoeuvres in modern helicopters covering such things as, auto hover, transition to the hover, transition to forward flight, overshoot/go-around from an automatic approach.

Example of a programmed manoeuvre

An example of a programmed manoeuvre is the transition from forward flight to the hover. These manoeuvres require many signals from a variety of sensors such as ground speed (along and across) from a source (Doppler or other) combined with height (both barometric and radio/radar altimeter), vertical speed (radar altimeter, barometric, acceleration sensed or Doppler-based), airspeed, engine and airframe data. These signals must be blended

in such a way as to take the helicopter from a starting condition to the desired hover condition in terms of height above ground or water, heading and speed with respect to the ground/water and air.
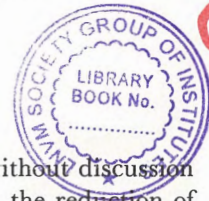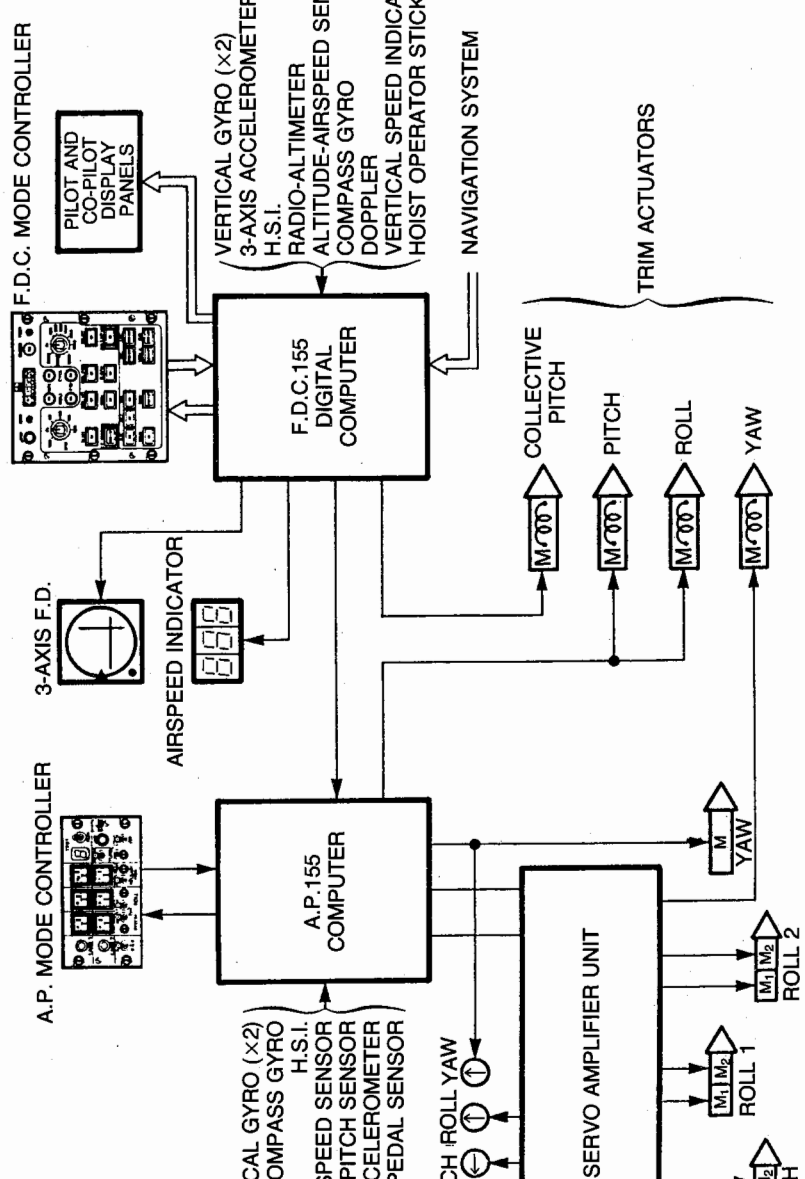
Advanced type of AFCS

The study of a helicopter AFCS would not be complete without discussion of a complex system. Among the functions of an AFCS is the reduction of pilot workload or the performance of tasks that the pilot would find difficult or impossible. The AFCS under consideration falls into both of these categories. Part of the system is designed to reduce the pilot's workload in normal operations by maintaining airspeed or another function, and another part of the system is designed to hover the helicopter without visual references, such as at night over the ocean.

As can be seen from fig. 8.12, the inputs to the system are more complex than the simple AFCS described earlier. The controller and logic requirements are also of greater scope, but the output to the actuators is no more complex than previously discussed. The items labelled as trim actuators are also parallel actuators.

The control panel of the AFCS (see fig. 8.13) has a significant number of modes that permit the pilot to select features appropriate to the flight condition. The 'normal' mode of attitude hold is always selected, and only additional modes are shown on this panel. Several of the modes are mutually exclusive because in using the same axis, they cannot be used simultaneously. For example, the airspeed hold (A/S) mode is controlled by the longitudinal cyclic channel, so it is not possible to also engage the altitude hold (ALT) or vertical speed (V/S) mode at the same time. If the pilot is using a constant collective (power) setting in turbulence, he must choose between altitude hold with variable airspeed or airspeed hold with variable altitude.

In this system, in the A/S mode, the pilot is able to adjust the commanded airspeed by use of the beeper trim switch on the cyclic. When engaged, a display in the normal airspeed indicator (ASI) shows the commanded airspeed, and the pilot is able to adjust it to the desired speed. The computer then corrects the attitude to attain this speed. In ALT mode the AFCS maintains the altitude at the time of engagement. If it is desired to change altitude, then ALT mode must be disabled, either by selecting another mode such as A/S or V/S or de-selecting hold ALT mode and then re-selecting ALT mode at the new altitude. A major limitation on ALT mode is operation through the longitudinal cyclic; it is only useable above $V_{min\ power}$. If the helicopter is driven below the commanded altitude by a gust, the AFCS will attempt to correct by raising the nose. If the helicopter is operating *above* minimum power speed, raising the nose will decrease the airspeed and the helicopter will climb as the power required for level flight
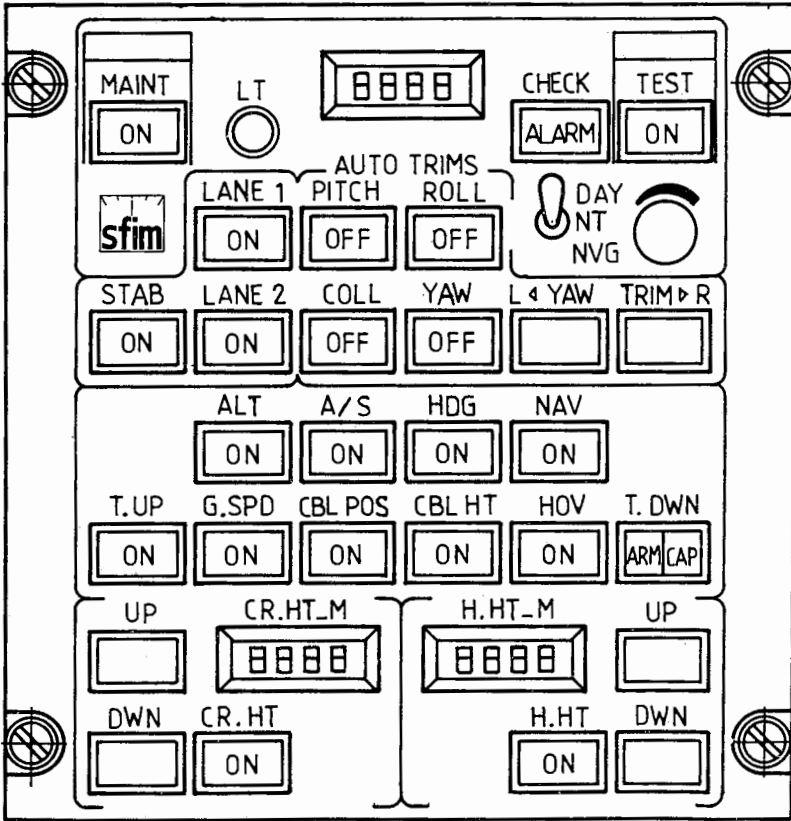
F.D.C. MODE CONTROLLER

PILOT AND CO-PILOT DISPLAY PANELS

VERTICAL GYRO (×2)
3-AXIS ACCELEROMETER
H.S.I.
RADIO-ALTIMETER
ALTITUDE-AIRSPEED SE
COMPASS GYRO
DOPPLER
VERTICAL SPEED INDICA
HOIST OPERATOR STICK

NAVIGATION SYSTEM

TRIM ACTUATORS

F.D.C.155 DIGITAL COMPUTER

COLLECTIVE PITCH

PITCH

ROLL

YAW

3-AXIS F.D.

AIRSPEED INDICATOR

A.P. MODE CONTROLLER

A.P.155 COMPUTER

M YAW

SERVO AMPLIFIER UNIT

M₁M₂ ROLL 2

M₁M₂ ROLL 1

CAL GYRO (×2)
OMPASS GYRO
H.S.I.
SPEED SENSOR
PITCH SENSOR
CELEROMETER
PEDAL SENSOR

CH ROLL YAW
LL YAW

anced type of AFCS.

Fig. 8.13 Typical control panel.

is lower at the new, reduced airspeed. When it reaches the correct altitude, the nose will be lowered and the speed will again increase. If, however, the helicopter is below minimum power speed, raising the nose will further decrease the speed, but the power required to maintain level flight at the slower speed will be greater than before, resulting in a rate of descent. The nose will be raised further to stop the descent and attempt to regain the altitude, making the situation worse. This may continue until the airpeed decreases to zero. For this reason, many such systems incorporate a minimum airspeed switch, or have warnings in the flight manual to prevent the pilot using the system in a region that is inappropriate.

Automatic transition to the hover

There are numerous times when the helicopter must hover over water in conditions of poor visibility and weather. Even in daylight, such hovering

is not ideal as the combination of wind and water motion and rotor downwash often masks the movement of the helicopter with respect to the water. The problem is further complicated by the lack of useful information from the pitot static system below 40 knots. This removes pitot and static information, the two most commonly used items promoting the pilot's situational awareness, at a critical time while hovering and during transition to the hover and back to forward flight. It is possible to replace this information with equally useful knowledge, such as height above water, and groundspeed with respect to surface. A Doppler sensor can provide the groundspeed, and a radar altimeter can provide the height above the surface; however, it is extremely difficult for the pilot to integrate this information to repeatedly arrive in a safe hover in less than ideal conditions. It is far more repeatable and safer to program the AFCS to complete this task.

Transition from cruise to hover

At an appropriate location prior to the desired hover point, the pilot selects the transition down mode (labelled T.DWN on fig. 8.13). This engages the pitch, roll, yaw and collective channels of the AFCS to perform a pre-programmed manoeuvre. The helicopter begins to slow and descend to follow a profile of speed and height to arrive at the height the pilot has previously selected on the hover height controller. As the airspeed falls below a certain pre-set value, the yaw channel converts from balanced flight to heading hold mode, and the roll and pitch channels change from bank angle hold and airspeed hold to lateral and longitudinal groundspeed programs. The helicopter continues to descend and slow down until it reaches the commanded hover height and the *along* and *across* groundspeed velocities are zero.

As the helicopter is now operating in the region of reversed effect with respect to power vs airspeed, it is not possible to control altitude with only the cyclic. As both airspeed and altitude must be controlled, the collective channel as well as the cyclic channel is used. The collective channel is now used to maintain height above the ground or water. Information on the height, speed and rate of descent, as well as power, engine parameters, and position with respect to the start and end points, are all fed into the computer, and the AFCS completes the transition automatically.

If the pilot desires, he can place the helicopter in the hover manually and engage the 'hover' mode when stable. Regardless of whether the transition was done automatically or manually, the pilot can modify the *along* and *across* velocity by use of the beeper trim switch on the cyclic.

At night over water, the transition from the hover to forward flight is also a very complex task in an area with little (if any) visual or suitable instrument flying information, the task is often performed by the AFCS. On the controller in fig. 8.13, for example, pressing the T.UP button will make

the AFCS fly a pre-determined profile to the previously selected cruising height, and airspeed. Both power (collective) and longitudinal cyclic air-speed profiles will be maintained.

Radio coupled operations

Many civil helicopters do not have as complex a system as the one just described, but still have quite complex and capable autopilots. The principles of operation of such systems are no different from those of a fixed-wing aircraft, namely that the sensor detects a signal, compares it to a reference, decides what action is to be taken to correct any errors, and commands the actuator to move in the appropriate direction.

TEST QUESTIONS

1 In what way does the helicopter trim system differ from that of a fixed wing aircraft?
2 What are the two basic types of actuator arrangements?
3 In a stability augmentation system, what is the purpose of the stick out of detent switch?
4 What is the difference between an SAS and an SCAS?
5 What is the difference between an attitude command/attitude hold system and a rate command/attitude hold system?
6 In what way does the helicopter stability augmentation system differ from that of a fixed-wing stability augmentation system?
7 Does an attitude-based stabilisation system require flight control position sensors?
8 In an AFCS that uses the cyclic for altitude hold, why is there a lower speed limit on the use of this mode?
9 In an AFCS which has radar altitude hold, why is an additional signal needed in that channel?
10 In an AFCS which will transition a helicopter from forward flight to the hover, why is it necessary to change the method of operation of the yaw channel part way through the manoeuvre?

9

Flight Director Systems

Flight instrument evolution has followed a pattern of divergent display complexity with advancing technology followed by consolidation of the displays as human capabilities of data interpretation were exceeded. Initially, instrument panel space was devoted to the minimum instrumentation needed for the control of the aircraft, i.e. the turn and slip indicator and the airspeed indicator. Also present were essential engine data displays, and the fundamental attitude data displays provided by the gyro horizon, the barometric type altimeter, vertical speed indicator, and the magnetic compass. Even at this low level of complexity, problems associated with interpretation of displayed data, scanning of instruments and accuracy were manifest. Many diverse panel arrangements came into use, reflecting the vagaries of particular instrument manufacturers, of demands on panel shape and size, and of the experience of particular pilots under instrument flying conditions.

Although the minimum instrumentation referred to above provided the essential data to manoeuvre and control the aircraft, instrument flight was feasible only as an emergency measure for relatively short periods. However, point-to-point navigation, with the precision needed for both military and public transport operations, required the development of new flight instruments and navigational aids. This eventually came about with the progressive development of such radio navigation systems as automatic direction finder (ADF), ILS, VOR, and distance measuring equipment. While it was possible, and practical, to navigate by the earlier developed systems, a high degree of pilot proficiency was nevertheless demanded. This was particularly evident during the terminal manoeuvring and approach-to-landing phase, where the most precise flight path control is required coincident with the performance of many other tasks preparatory to landing.

In order to ease the pilot's workload, and to achieve greater precision of flight guidance and control, the idea of presenting data in the form of control commands was conceived, and this led ultimately to the application of servomechanism design principles to systems known as flight directors.

Flight directors

A pioneer flight director system (FDS) was the Zero Reader (developed by Sperry) which sensed not only flight path deviation, but also such control parameters as aircraft attitude, attitude rate, and deviation rate. Control command signals were computed and presented on an indicator not unlike that of the ILS. Although the pilot no longer had to assemble basic control data from a number of instruments, it was still necessary for him to monitor continuously other instruments as an assurance that all parameters were consistent and compatible with the desired flight objective.

It was logical, therefore, in the next stage of development, to integrate functions of a flight director with those of the complementary instruments, so that essential data could be displayed on fewer instuments thereby reducing scanning time. With small variations from one application to another, and from one aircraft operator to another, this basic integration concept is adopted in a majority of today's aircraft, despite the added demands stemming from the progress in operating under reduced visibility, and in the development of new navigation equipment. The lowering of weather minima (see page 278) directly affects instruments complexity as a result of the need for additional display functions, such as redundant and expanded flight path deviation, airspeed deviation, radio altitude, de-crab and roll-out command data, and integration of turn-and-slip data into the primary display. The advancement to lower weather minima created the need for display integrity; this, in turn, required that nearly all the functions of the instruments be monitored, and that provision be made for the warning of faulty display functions. The demands created by the development of new navigation equipment result from technical innovation and added flexibility. There is a need to assimilate navigational data not previously available or which is now available in a new form. Furthermore, there is a need to indicate which type of system is generating the displayed information, in which mode that system is operating, which of several redundant sets of that type of equipment is connected, and in what co-ordinates the data is displayed.

In the field of automatic control, the development of control systems was also strongly influenced by the demands imposed on FD systems, particularly for control in all phases of flight. The principal reason for this is that much of the basic attitude and navigational data is common to both systems. It was also logical, therefore, to expand on the concept of integration, such that data and servomechanism links could be shared so that an FDS could provide guidance commands for an AFCS and monitor its performance, and be available for reversion to effective manual control if fully automatic control should be lost.

An FDS developed in this manner comprises two principal display units: (i) attitude direction indicator (ADI); and (ii) horizontal situation indicator (HSI). A number of display configurations may be adopted, dependent on
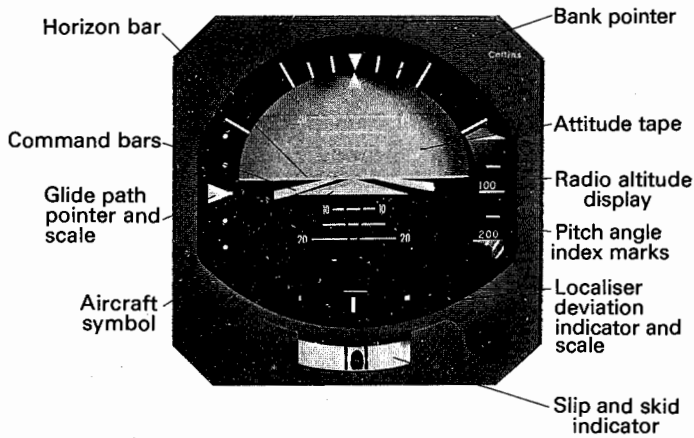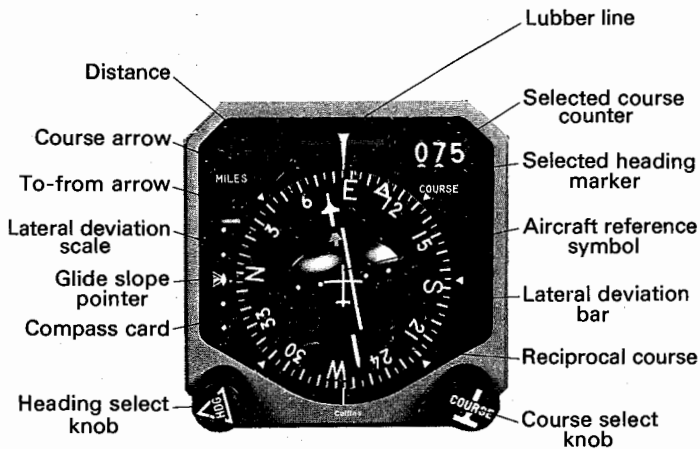
Fig. 9.1 Attitude direction indicator.

Fig. 9.2 Horizontal situation indicator.

a particular manufacturer's design, and on the extent of the functions to be integrated. However, there are certain features which are of a common nature and these may be highlighted by considering the two units shown in figs. 9.1 and 9.2.

Attitude direction indicator (ADI)

This indicator presents aircraft attitude and direction information in the form of a three-dimensional display. Attitude is displayed by the relationship

of a stationary delta-shaped symbol representing the aircraft, with respect to roll and pitch commands displayed by two pointers, or command bars flanking the aircraft symbol, and also by a horizon bar. The command bars form a shallow inverted 'V', and are driven by separate servomotors within the indicator such that they move up and down to command a change in pitch, and rotate clockwise and anti-clockwise to command a change of roll attitude. The outputs of the two servos are combined mechanically so as to provide an integrated pitch and roll command. Sensing with respect to the aircraft symbol is such that the pilot is always directed to 'fly into the V'. When a command has been satisfied, the command bars are aligned with the edges of the aircraft symbol.

The horizon bar is carried on a flexible tape which is also driven by separate pitch and roll servomotors within the indicator. Freedom of tape movement in pitch is ±90°, and 360° in roll. The upper and lower sections of the tape are coloured to represent the sky and ground respectively, and they also have index marks on them to indicate pitch angles. In some types of flight director, the lower section of the moving tape is also marked with lines converging on the centre of the indicator display thereby enhancing its 'forward view' effect. Roll angle is displayed by a pointer which rotates with the flexible tape, and is referenced against a fixed scale. The servomotors are supplied with signals from a vertical gyroscope unit located at a remote point.

Deviations from the ILS glide slope beam (see also page 181) are shown by vertical displacements of a pointer over a scale at the left-hand side of the indicator display. Each of the inner dots on the scale represents a ¼° displacement from the beam centre line, while the outer dots each represents a ½° displacement. The pointer is driven by a d.c. meter movement and, when not in use, is deflected out of view at the top of the scale. A pointer at the lower part of the display indicates deviations from the localiser beam, and is shaped to symbolise a view of a runway during an approach. The reference dots on the localiser or runway scale, indicate approximately 1¼° displacement from the beam centre line. The pointer is also driven by a d.c. meter movement, and, when not in use, is obscured by a black warning flag as shown in fig. 9.1. In some types of indicator, the localiser pointer, or runway symbol is also displaced in response to signals from a radio altimeter so that during the last 200 feet of descent, the pointer moves up to the fixed aircraft symbol thereby presenting a 'rising runway' display of the approach. This radio altitude display concept is also adopted in the indicator shown in fig. 9.1, but, in this case, it is effected by a pointer moving over a fixed altitude scale.

Indications of slip and skid are provided by an inclinometer similar to that adopted in conventional turn and slip indicators. In addition, some flight directors have a rate of turn pointer incorporated in the display, the pointer being actuated by signals from a rate gyroscope sensor unit.

Another command function which may be displayed in some attitude

direction indicators is that related to the speed of an aircraft when executing a go-around manoeuvre following a missed approach. The display comprises a vertical scale and a pointer which is actuated in response to signals corresponding to the difference between indicated airspeed and a predetermined go-around speed obtained from a speed computing system external to the flight director. The scale has several graduations ranging from the computed speed at the centre, to 'fast' and 'slow' at the top and bottom of the scale respectively. In order to achieve the correct go-around speed, engine power is adjusted so as to maintain the pointer at the centre of the scale. The pilot selects the go-around mode by pressing a button switch on the control wheel, the selection being indicated by the illumination of an annunciator light marked 'GA', and by displacement of the flight director command bars to command a wings-level climb attitude. In association with the go-around mode, a second annunciator light marked 'MDA' (minimum decision altitude) is provided. The light illuminates when the aircraft has descended to the preset radio altitude at which the decision whether to land or go-around must be made. If an FDS is supplying guidance commands to an AFCS during an approach, the latter system (with the exception of one having automatic landing capability) is caused to disengage when the 'GA' mode is selected.

Horizontal situation indicator (HSI)

This indicator presents a pictorial display of navigation situation and, as will be noted from fig. 9.2, the situation is shown as a plan view of the aircraft's position and heading with respect to a selected heading and course. In addition to magnetic heading data signals, the indicator is also supplied with signal inputs corresponding to deviations from an ILS localiser beam, and VOR radial, and from a glide path beam. Indication of flight either to or from a VOR station is also provided. Selector knobs at the bottom corners of the indicator permit the setting of a desired magnetic heading and a VOR or localiser course. The aircraft reference symbol is fixed at the centre of the display and it indicates the position and heading of the aircraft in relation to the compass or azimuth card, and the lateral deviation bar. The compass card is synchronous-linked with the aircraft's magnetic heading reference system, and when changes in aircraft heading take place a position error signal is produced in a control transformer synchro within the indicator. After amplification by a servo ampliflier in the instrument ampliflier unit, the signal is supplied to a motor which, by means of a gear train system, drives the compass card to indicate the new heading with reference to a lubber line at the top centre of the indicator. Card movement is damped by means of a rate feedback signal produced by a rate generator which is driven by the motor. Feedback signals are mixed

with position error signals, and the composite signal is amplified and supplied to the control phase of the motor.

At the same time, the motor drives the control transformer synchro rotor to the null position corresponding to the new heading. The lateral deviation bar is a movable centre section of the course arrow, and represents the centre line of the selected localiser course or VOR radial. The bar is deflected to the left or right by a d.c. meter movement to indicate the appropriate commands necessary for beam interception and capture, and it also rotates with the compass card as changes in aircraft heading take place. When operating in the localiser mode initial movement of the deviation bar begins when the aircraft is approximately 4° from the localiser beam centre, and the dots on the deviation scale represent approximately 1¼° and 2½° from beam centre. In the VOR mode, initial movement of the bar begins when the aircraft is approximately 16° from radial centre, and the deviation scale dots then represent approximately 5° and 10° from radial centre.

Selection of a desired localiser course or VOR radial, is carried out by rotating the course selector knob until the course arrow coincides with the desired value on the compass card. The lateral deviation bar and deviation scale also rotate with the course arrow through the gear train system driven by the selector knob. At the same time a digital type of course counter is driven to the corresponding course indication; in fig. 9.2, this is displayed as 075°. Once set, the course arrow rotates with the compass card as aircraft heading changes. The gear train system also positions the rotors of a course resolver synchro associated with the VOR/LOC navigation receiver, and of a course datum control transformer synchro. When the course resolver synchro rotor position is changed, it shifts the phase of the reference 30 Hz signal in a phase shift circuit of the VOR instrumentation unit. The signal is then compared with the variable 30 Hz signal in a phase comparator, the output of which is supplied to the meter movement controlling the lateral deviation bar. When the output is such that it centres the deviation bar the aircraft is on the course selected.

When the aircraft deviates from the selected course, the phase-shifted reference signal is maintained at the angle determined by the resolver synchro rotor, but the variable signal phase received by the VOR navigation receiver is changed. The phase comparator will then produce an output which deflects the deviation bar to the left or right of the selected VOR course. The to-from arrow is positioned by a meter movement which is supplied with the corresponding signals from the radio navigation receiver and via a phase comparator in the instrumentation unit of the FDS. In fig. 9.2 a 'fly to' command is displayed. In the LOC mode of operation, the deviation bar is similarly controlled by changes in resolver synchro rotor position, except that the output to the meter movement results from amplitude comparison of the signals either side of localiser beam centre. The to-from arrows remain out of view since no to-from signals are transmitted in the localiser mode.

Changes in the position of the course datum control transformer synchro rotor produces a position error signal in the stator windings. The signal is proportional to the difference between the selected course and the actual heading of the aircraft and is transmitted to the roll control channel of a steering computer or a flight director computer as a turn command to capture the selected VOR/LOC course. The signal is also transmitted to the AFCS and if this system is coupled to the FDS, it will, of course, turn the aircraft automatically. The output from the appropriate computer is supplied to a roll command servo amplifer contained in an instrument amplifier unit, and after amplification it is fed to the roll servomotor coupled to the command bars of the ADI. Thus, the bars rotate to indicate the direction of roll required to capture the VOR/LOC course. The motor also drives a rate generator which produces a rate feedback signal for the purpose of damping display movements.

The selection of any desired magnetic heading is accomplished by positioning a triangular-shaped heading marker over the compass card, by means of the heading selector knob and its associated shaft and gear train system. At the same time, the rotor of a heading error control transformer synchro is rotated inside its stator, from its null position, and this produces a position error signal proportional to the difference between the selected heading and the aircraft heading sensed by the compass system. In fig. 9.2, the headings displayed are respectively 110° and 085°. The signal is processed in the same manner as that produced by the course datum control transformer synchro, and therefore results in the ADI command bars indicating the direction of roll required to fly on the desired heading.

Warning flags

As noted earlier, provision must be made for the warning of faulty display functions. In practice, warnings are effected by monitoring the command signals produced, so that when they are lost or are too weak to provide reliable information small red flags appear at appropriate parts of the ADI and HSI displays. The flags are actuated by d.c. meter mechanisms which are connected to the relevant signal sources.

In the case of an ADI there are, primarily, three warning flags labelled 'GS', 'GYRO' and 'COMPUTER' and respectively they indicate malfunctions of the glide-slope receiver or signal, and vertical gyroscope and attitude display systems, and the director or steering computer and command display systems. The GS flag, when indicating a malfunction, obscures the glide slope pointer and scale to prevent its use. If the system is not being operated in the glide slope mode, the GS flag and pointer are biased off-scale. Indication of localiser signal malfunction and/or localiser mode not selected, is also provided and generally takes the form of a black shutter which obscures the localiser pointer and scale.

Other warning flags may be provided depending on any aditional func-
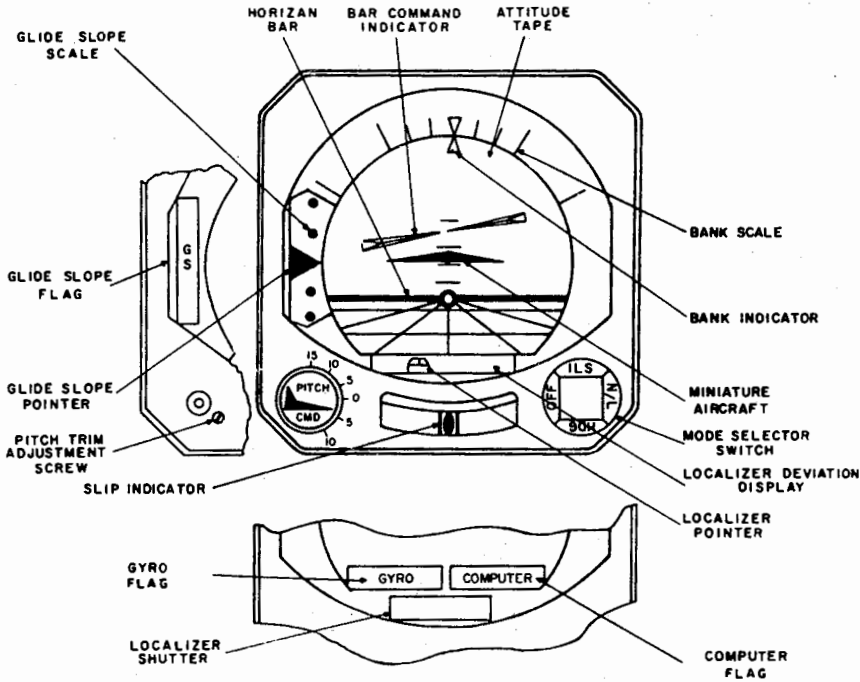
Fig. 9.3 Attitude direction indicator with pitch command and mode select facilities.

tions displayed; for example, a flag is provided to give warning of malfunctions in the circuit of a speed control display associated with a go-around manoeuvre.

In the case of an HSI, there are also three primary warning flags and these are labelled 'GS', 'COMPASS' and 'VOR/LOC'. The GS flag operates in the same manner as that provided in the ADI, while the compass flag indicates malfunctions of the magnetic heading signal circuit of the magnetic heading reference system. The VOR/LOC flag serves the dual function of warning of VOR radial signal and localiser signal malfunction. Warning flag operation is summarised in the table on page 267.

Pitch command facility

In some types of FDS, a pitch command facility is provided which permits the pilot to preselect a fixed climb or descent command under certain modes of operation. Selection is carried out by means of a selector knob which, in some cases, is located in the bottom left-hand corner of an ADI (see fig. 9.3) and in others is located on a separate flight director mode selector panel. The selector knob is mechanically coupled to the rotor of a control transformer synchro, and after the knob is rotated a signal is

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induced in the synchro. After amplification, this signal is transmitted to the pitch servomotor/generator which drives the command bars to the selected position. The aircraft attitude is then changed by 'flying the aircraft symbol into the command bars'. In addition to pitch command, a pitch trim adjustment is provided as a means of altering the position of the ADI horizon bar with respect to the aircraft symbol. The adjustment is purely mechanical in operation and is used for aligning the attitude display during installation of an FDS.

Operating modes

A number of FD systems incorporate facilities for selecting various modes of operation, such facilities being comparable in function to the outer loop control of an AFCS. This being so, it is possible for mode selection to be used on a common basis in cases where an FDS is employed in combination with an AFCS. The number of modes vary dependent on specific aircraft operating requirements, and for a similar reason, the method by which modes are selected can also vary; for example, selection may be effected
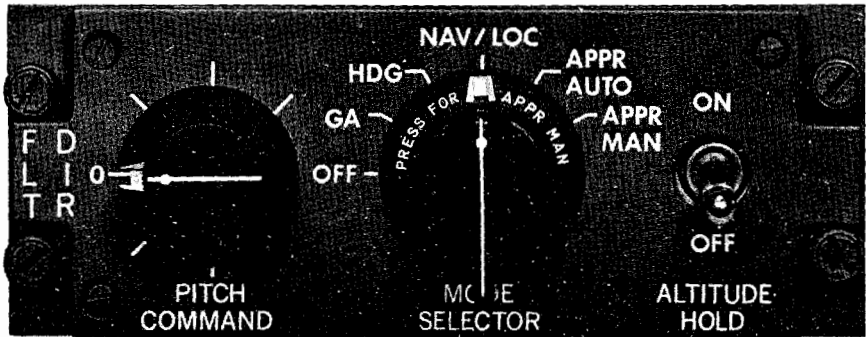
Fig. 9.4 Mode selector panel.
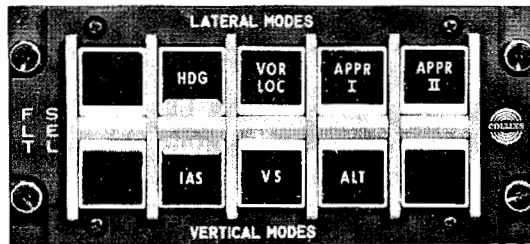
Fig. 9.5 Push-button type mode selector.

by the control knob of a rotary selector switch on an attitude direction indicator (see fig. 9.3) or on a separate mode selector panel (see fig. 9.4).

In some versions of a control panel, shown in fig. 9.5, modes are selected by push buttons which are push-on/push-off solenoid-hold switches. The push buttons illuminate when their corresponding modes are selected and at the same time a mechanically-actuated flag with the word 'ON' appears over a portion of each button engaged. The operating modes which are fundamental to some typical FD systems, are briefly described in Table 9.1, while some appropriate display indications are summarised in the table below. When each mode is selected, signal circuits are completed through the appropriate computer and amplifier sections, the outputs of which are supplied as command signals to the ADI. When an FDS is used in combination with an AFCS, the command signals are of course also utilised by this system for applying control in the sense necessary to satisfy the relevant commands.

Fundamental Operating Modes

OFF	Command bars deflected out of sight, and flight director indicator used as an attitude reference only.
HDG	Command bars provide lateral guidance to achieve and maintain a compass heading, as selected on the course indicator. Vertical guidance is from a preselected pitch attitude.
VOR(NAV)/LOC	Command bars provide lateral guidance to capture and track a VOR radial or localiser beam. Vertical guidance is the same as in HDG mode.
GS	Command bars provide lateral and vertical guidance to capture and track the localiser and glide slope beams respectively. The GS and LOC pointers monitor aircraft deviations of the beam.
GS AUTO	As for GS except that interception and capture of glide slope takes place automatically after the localiser beam has been captured.

ALT	Command bars provide vertical guidance to hold the aircraft at the desired altitude.
APPR I	Selected for capture and tracking of GS and LOC beams on ILS approaches to Category I standards (see page 279). Command bars provide lateral and vertical guidance.
APPR II	As for APPR I but produces tighter tracking of beams to meet higher precision requirements of a Category II ILS approach (see page 279).
GA	Selected for a go-around manoeuvre after a missed approach, and after selecting either one of the approach modes. The command bars command a wings-level, pitch-up attitude. HDG and IAS modes may be selected after go-around power settings and airspeed are established.
IAS	Selected to maintain a particular indicated airspeed during climbout after take-off, and during letdown over a VOR station. The command bars provide vertical guidance.
V/S	Selected to maintain a particular vertical speed, i.e. rate of climb or descent. The command bars provide vertical guidance.
MACH	As for IAS mode but selected at higher altitudes.

Computer and amplifier units

As already noted in the foregoing brief descriptions of ADI and HSI operation, the appropriate command signals are processed by computer

pical warning flag operation

ATTITUDE DIRECTION INDICATOR

ILS CONTROLLING COMMAND V-BARS — PITCH	HORIZON ATTITUDE DISPLAY	GLIDE-SLOPE DEVIATION POINTER	LOCALIZER DEVIATION POINTER	WARNING FLAGS			
				GLIDE-SLOPE	LOCALIZER	GYRO	COMPUTER
OUT OF VIEW		OUT OF VIEW UNLESS TUNED TO ILS – THEN SHOWS GLIDE-SLOPE DEVIATION		OUT OF VIEW UNLESS TUNED TO ILS – THEN MONITORS ILS RADIO			
PITCH COMMAND + PITCH ATTITUDE	SHOWS PITCH & ROLL		COVERED BY FLAG		COVERS DISPLAY	MONITORS GYRO + SERVO POWER + SERVO ERROR	
PITCH ATTITUDE + GLIDE-SLOPE RADIO DEVIATION		INDICATES GLIDE-SLOPE DEVIATION	INDICATES LOCALIZER DEVIATION	MONITORS GLIDE-SLOPE RADIO	MONITORS LOCALIZER RADIO		

HORIZONTAL SITUATION INDICATOR

GLIDE-SLOPE DEVIATION POINTER	VOR/LOC DEVIATION BAR	WARNING FLAGS		
		GLIDE-SLOPE	VOR/LOC	COMPASS
OUT OF VIEW	BIASED OUT OF VIEW			
OUT OF VIEW UNLESS TUNED TO ILS – THEN SHOWS GLIDE-SLOPE DEVIATION			MONITORS COMPASS + GYRO + COMPUTER	
		OUT OF VIEW UNLESS TUNED TO ILS – THEN MONITORS ILS RADIO	MONITORS NAV RADIO	MONITORS COMPASS + SERVO POWER + SERVO ERROR
INDICATES GLIDE-SLOPE DEVIATION	DISPLAYS VOR/LOC DEVIATION	MONITORS GLIDE-SLOPE RADIO		

and instrument amplifier units. The primary function of a computer unit is to provide all the computation necessary for determining any position or attitude errors, and to develop the signals necessary to command position or attitude changes. When an FDS is integrated with an AFCS, the computed signals are also utilised for the application of control. A computer may in some cases be a single unit containing solid-state signal circuits for both lateral and vertical guidance information, while for some director systems separate computer units are utilised. In addition to the signal circuits, a logic network is incorporated, its purpose being to provide correct analogue scaling of signals, and to adjust computer gains and logic to suit specific types of aircraft. All signal and power supply circuits are on printed circuit boards which are arranged as separate functional plug-in modules.

The primary function of an instrument amplifier is to supply servo-actuating power for the display mechanisms of the HSI and ADI. The unit also contains separate plug-in module circuit boards which, as shown in the overall signal flow diagram of a representative system (fig. 9.6) correspond to five servo channels, two signal convertor channels, and three flag alarm circuits. If additional warnings are required the number of alarm circuits is increased accordingly. The converter channels accept d.c. input signals and converts them to 400 Hz signals for use by the pitch and roll command servo channels. The convertors receive pitch and roll steering signals from the computer, position error signal information from the pitch and roll command control transformer synchros in the flight director, and rate feedback signals from the pitch and roll command motor/generators. These signals are mixed and, after filtering, the composite signals provide the input for the appropriate command servo amplifier.

Electronic flight instrument system

This system, commonly referred to as 'EFIS', is a highly sophisticated type of FDS in which flight operating and navigational data is processed by high-storage capacity digital computers, and then presented to the flight crew in video form. EFIS has two display units that correspond to the ADI and the HSI of a conventional FDS, but whereas the latter requires electro-mechanically controlled elements for the presentation of data, EFIS display units utilise colour cathode ray tubes. Movements of horizon lines, pointers, command bars, etc., are effected by circuits which, in response to the appropriate data signals, deflect electron beams within the tubes to appropriate matrix locations on the active surfaces of the tubes.

A single system is provided for each pilot, and as will be noted from fig. 9.7, a system comprises a symbol generator, a control panel, and the two indicators. The symbol generator is comprised of microprocessors, memories, and the circuits necessary for generating the displays of vital
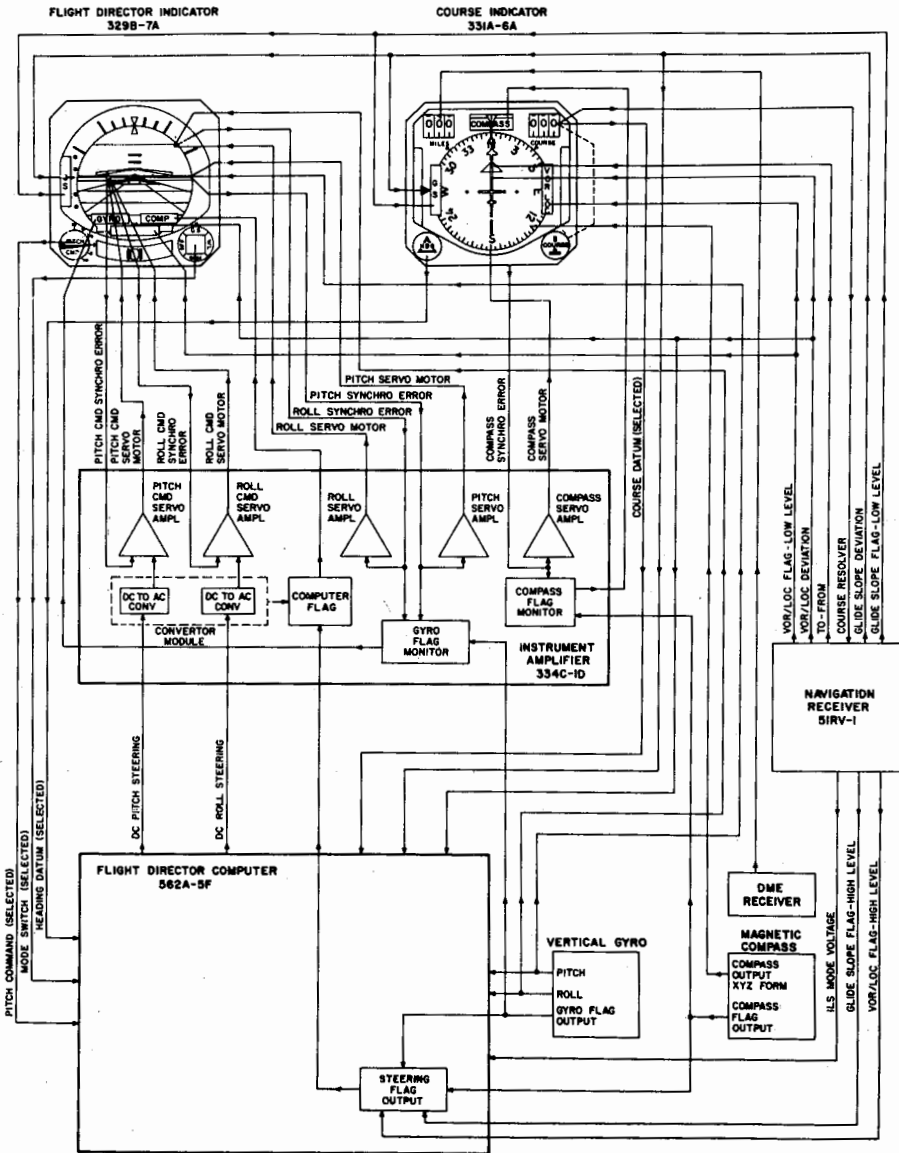
Fig. 9.6 Signal flow diagram of a representative flight director system.

information. It receives data from a data bus which is supplied with signals from all the relevant systems of an aircraft and their computers. The system abbreviations indicated in fig. 9.7 are defined in Appendix 3.

In a complete installation, three symbol generators are required; the third (centre) generator is a 'standby' one so that in the event that either of

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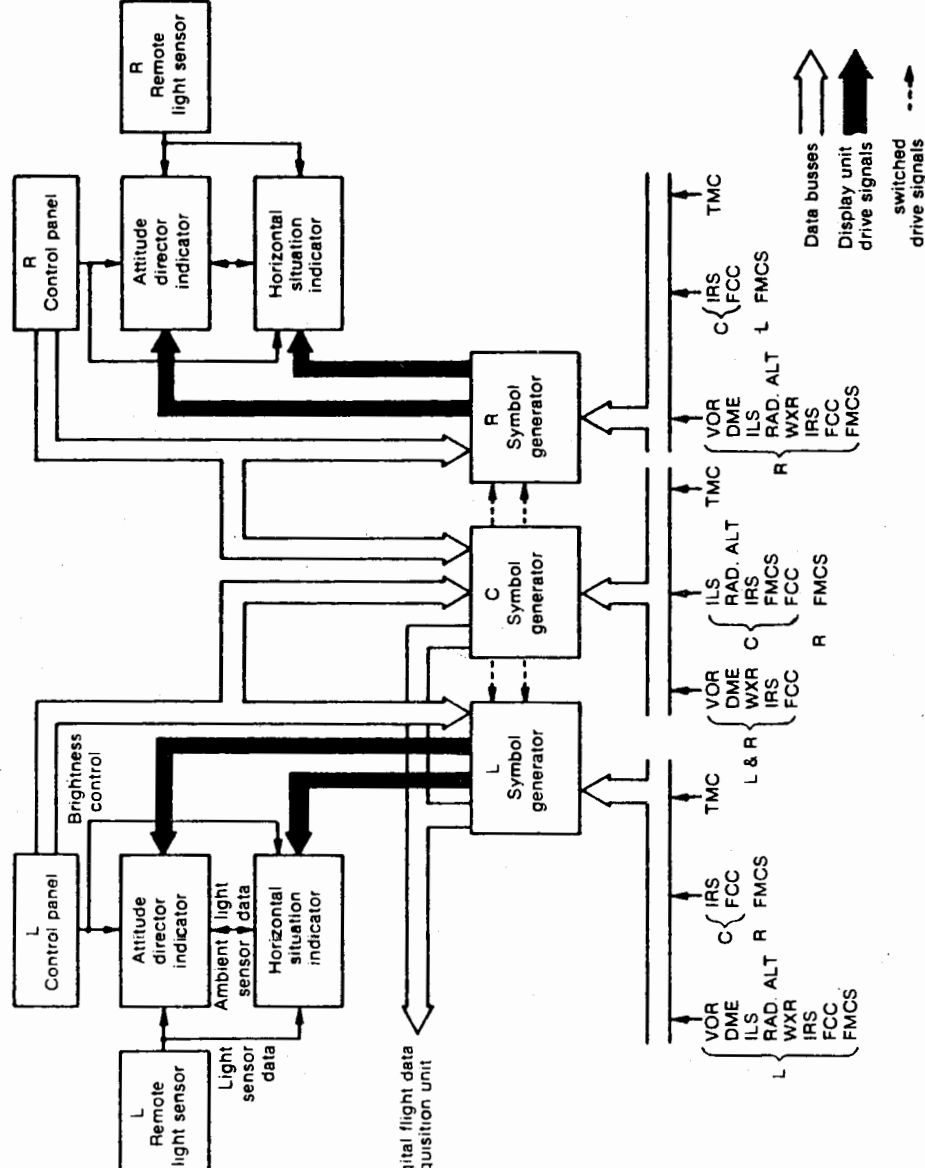
Fig. 9.7 EFIS units and signal interfacing.
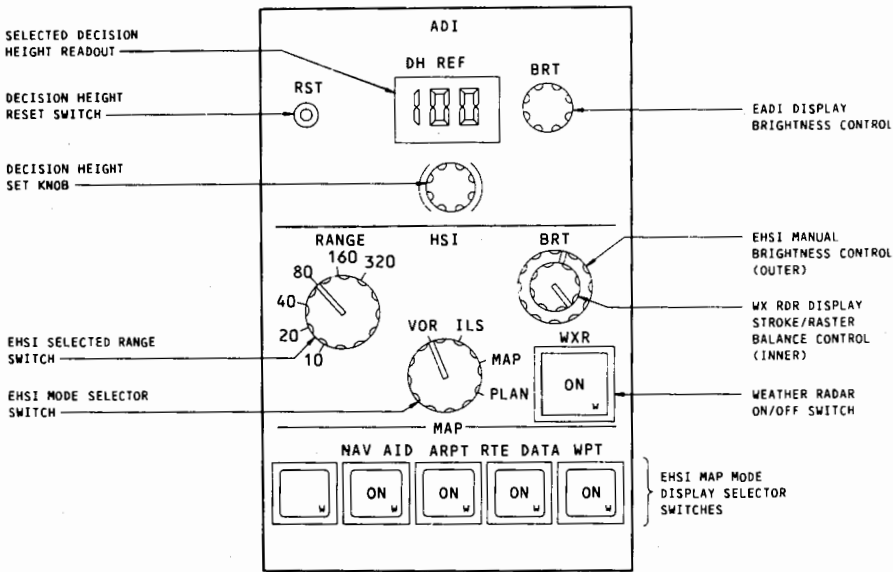
Fig. 9.8 EFIS control panel.

the other two should fail, its drive signals may be switched to the appropriate display units. Switching is done via electromechanical relays that are under the control of selector switches mounted on each of two control panels.

Control panel

Each pilot is provided with a control panel, and as may be seen from fig. 9.8, the control switches are grouped appropriate to each of the display units.

Attitude director indicator (ADI)

This unit displays traditional pitch and roll information, flight director commands, ILS localiser and glide slope deviations, selected airspeed, ground speed, automatic flight control and auto-throttle modes, radio altitude and decision height. As far as pitch and roll attitude information is concerned, the unit is an example of one utilising inertial references (see pages 124 and 190).

All information is displayed on an approximately five-inch square screen, and to convey some idea of the extent of this information and the colours in which it is displayed, fig. 9.9 illustrates the display presentation representing an aircraft making an automatically controlled approach to land. If during

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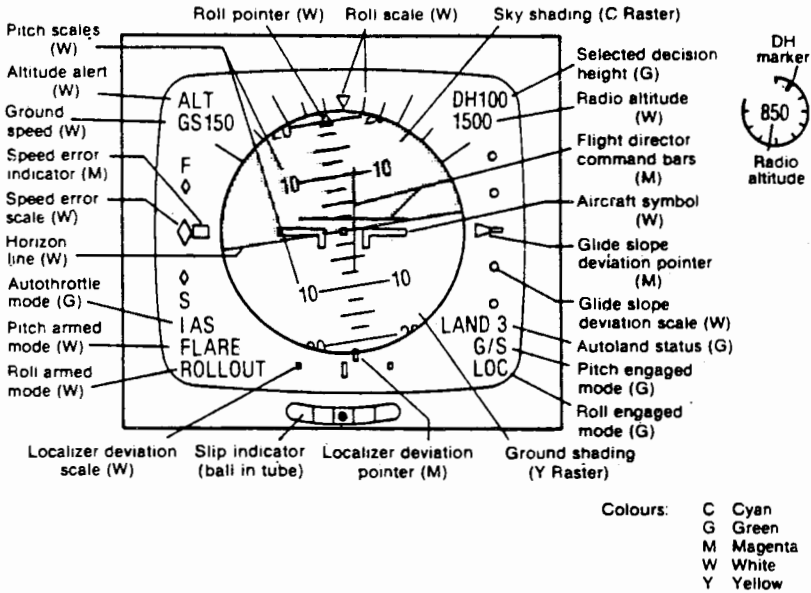
Fig. 9.9 EFIS attitude director indicator.

the approach the aircraft deviates beyond the normal ILS glide slope and/ or localiser limits, the flight crew are alerted by the respective deviation pointers changing colour from white to amber; the pointers also start flashing. The alert condition ceases when the deviations return to within their normal limits.

It will also be noted from fig. 9.9 that pitch and roll information is referenced against a shaded background which, as in the case of the indicator used in conventional flight director systems, represents the sky and the ground. The sky shading is in the colour cyan (light blue) and the ground shading in yellow, both being produced by *raster scanning*; this is the term used to denote the manner in which an electron beam generated within a cathode ray tube is deflected back and forth to produce a solid image on the screen of the tube. All other symbols and characters are produced by a digital stroke pulsing technique known as *stroke scanning*. Each method of scanning is produced by individual generators contained within the symbol generators.

Radio altitude is digitally displayed during an approach, and when the aircraft is 2,500 and 1,000 feet above ground level. Below 1,000 feet the display automatically changes to a white circular scale calibrated in 100-foot increments, and the selected decision height changes from a digital readout to a magenta-coloured marker on the scale. As the aircraft descends, segments of the circular scale are simultaneously 'erased' so that the scale length is continuously decreased in an anti-clockwise direction.

At the pre-selected decision height (plus 50 feet) an aural warning is sounded and continues at an increasing rate until the decision height is reached. At this point, the colour of the circular height scale changes from white to amber; both the scale and marker flash for several seconds. A reset button is provided on the control panel and when pressed the flashing is stopped and the scale and marker are changed back to their normal colours.

During operation, the indicator generates a fair amount of heat, and therefore air from an equipment cooling system is circulated through it at a controlled rate. Temperature-sensing units are provided which are set at low and high values approximately 20°C apart. If the lower value is exceeded, a signal is transmitted to the symbol generator which causes the rastered data (i.e. sky/ground shading) to be turned off, thus alerting the flight crew. The display is restored when the temperature drops 5°C below the 'shutdown' value. If the higher value is exceeded, the whole display is switched off. Normal operation (minus the rastered data) is restored when the temperature drops 6°C below the 'shutdown' value.

Horizontal situation indicator

This unit presents a dynamic display of flight progress and plan view orientation, in four difference display modes which may be selected on the system control panel. The modes are MAP, PLAN, ILS and VOR.

The normally used MAP mode display colours and symbol identification are illustrated in fig. 9.10. In conjunction with a flight plan programmed into the flight management computer (FMC) the indicator displays information against a moving map background (hence the term dynamic display) with all elements to a common scale. The symbol representing the aircraft is at the lower part of the display and an arc of a compass scale is at the upper part. As in the case of pitch and roll information, heading data is supplied from an inertial reference system; it may be selected as either magnetic or true heading.

The tuned VOR/DME stations, airports and their identification letters, and the programmed flight plan, are all correctly oriented with respect to the position and track of the aircraft, and to the range scale (nautical miles/inch) selected on the control panel. Thus, with 80 selected as shown in fig. 9.8, the same number will be displayed alongside the instantaneous track line as indicated in the figure. If the weather radar has been selected on from the control panel, the radar 'returns' are also displayed at the same scale and orientation as the map mode. Indications of other data such as wind speed and direction, lateral and vertical deviations from the selected flight profile, distance to waypoints, estimated time of arrival, etc., are also displayed.

Typical examples of the other modes are shown in fig. 9.11. In the PLAN mode, a static map background with route data corresponding to the FMC flight plan is displayed together with track and heading information. If the

Display orientation (G)
Magnetic track (W)
Aircraft heading pointer (W)
Distance to go (W)
Estimated time of arrival (W)
Heading set bug (M)
Magnetic compass rose (W)
Manually set VOR course (G)
Manually selected course (mag. hdg.) (M)
Range scale (W)
Active flight plan (M)
Range to altitude (G)
Vertical deviation scale (W)
Instantaneous track (W)
Vertical deviation pointer (M)
Wind speed and direction (W)
ILS frequency (G)
Lateral deviation scale (W)
Aircraft symbol (W)
Curved trend vector (W)
Lateral deviation pointer (M)

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Symbols: Waypoints: Active (M) one aircraft currently navigating to.
Inactive (W) a navigation point making up selected active route.

Airports (C)

Navaids (C)

Wind direction (W) with respect to map display orientation and compass reference.

Off-route waypoints (C)

Colours:
C Cyan
G Green
M Magenta
W White

Fig. 9.10 EFIS horizontal situation indicator in 'map' mode.

flight crew wish to make any changes to the route, they first select them on the keyboard of the FMC display unit so that they can be checked as they appear on the screen of the HSI, before entering them into the FMC.

In the VOR and ILS modes heading orientation of the aircraft is displayed against either an expanded or a full compass scale. Selected range, wind information, and the name of the system from which signals are being supplied, are also displayed. If the weather radar has been switched on, radar 'returns' are also displayed when in these modes, but only if an expanded compass scale is presented.

PLAN   VOR

ILS
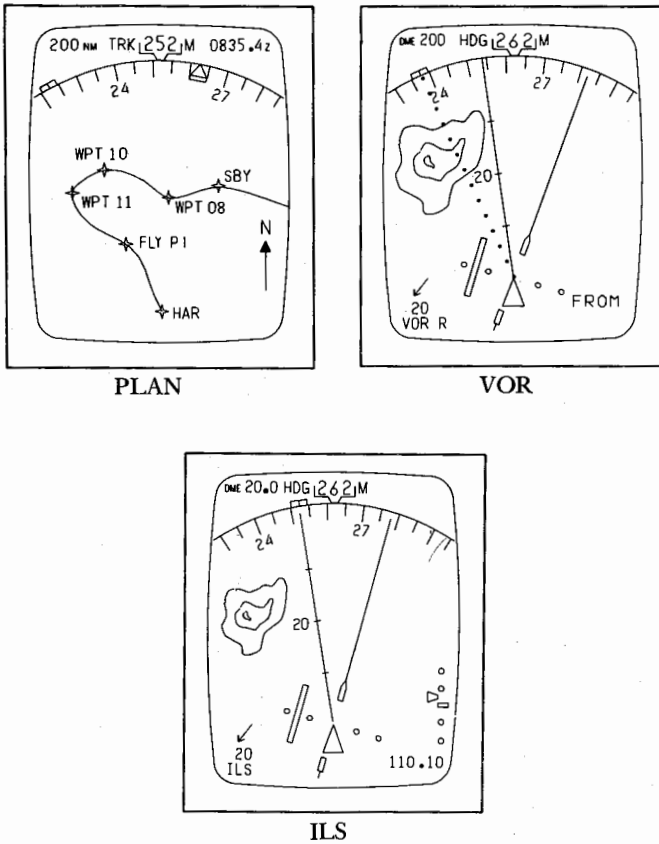
Fig. 9.11 Horizontal situation indicator operating modes.

An HSI display unit is cooled and temperature controlled in the same manner as the ADI.

Light sensors

In order to ensure that the light intensity of the displays is compatible with ambient light conditions in the flight deck, light sensors of the photo-diode type are provided to adjust the intensity automatically. Two sensors are mounted so that they respond directly to light coming in through the flight deck windshields, and there is one in the front of each display unit to respond to light in the area of the instrument panel.

Failure annunciations

As in the case of convential FDS indicators it is also necessary for failure of data signals from such systems as the ILS and radio altimeter to be

annunciated. In EFIS indicators annunciation takes the form of yellow flags 'painted' at specific matrix locations on the display unit screens (see fig. 9.11). In addition, fault messages may also be displayed, e.g. if the associated FMC and weather radar disagree with the control panel range data, the discrepancy message 'WXR/MAP RANGE DISAGREE' appears on the screen of the horizontal situation indicator.

TEST QUESTIONS

1 What information is displayed on a typical ADI? Describe a method of display presentation.
2 What type of signal is applied to the glide slope and localiser meters of FDS indicators?
3 What is the significance of the reference dots on the scales of glide slope and localiser meters?
4 How is the 'go-around' mode selected, and what happens if the FDS is supplying approach guidance commands to the AFCS at the moment of selection?
5 Under what conditions will the 'MDA' light be illuminated?
6 What indications are provided by the lateral deviation bar of an HSI?
7 The TO-FROM indicator of an HSI is in operation when:
    (a) either the VOR or LOC mode is selected.
    (b) only the VOR mode is selected.
    (c) a heading change is selected.
8 To which AFCS channels are FDS signals supplied?
9 How are desired magnetic heading changes selected and transmitted?
10 How are system malfunctions indicated, and to which operating conditions can they be related.
11 What arrangement is provided in some types of FDS to permit the preselection of a fixed climb or descent?
12 Under what conditions would the modes 'GS AUTO' and 'APPR I' be selected?
13 Name the main units that comprise an Electronic Flight Instrument System.
14 What is the function of the third symbol generator in a dual system?
15 Under what flight conditions do the glide slope and localiser deviation pointers of the EFIS ADI change from white to amber?
16 Name the four modes that can be displayed on the EFIS HSI and state how they are selected.
17 Can weather radar 'returns' be displayed in all four modes?
18 When failure of data signals occurs, indication is given on EFIS display units by flags displayed in:
    (a) red (b) magenta (c) yellow

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Automatic Landing and Autothrottle Systems

The fundamental landing requirement

In order to achieve a safe landing, an aircraft has to be so controlled that its wheels make contact with the ground comfortably within the paved surface of the runway, within fairly narrow longitudinal limits along it, and at a sinking speed not greatly in excess of an optimum value of about 1 to 2 feet per second. The speed at touch-down should have been reduced from the approach margin of about 30% above the stall to about half this value, by progressive reduction of engine power during the landing flare. Finally, the wings should have been levelled prior to the actual landing, and the aircraft yawed to bring its longitudinal axis parallel to the runway centre-line to remove any drift angle due to cross-wind; the latter manoeuvre being known as decrabbing, or drift 'kick-off'. Control of the aircraft is needed about all three axes simultaneously, as well as the control of airspeed through engine power changes; it will be appreciated, therefore, that the approach and landing manoeuvre is the most difficult one demanded of the pilot. Add to this the unfortunate fact that in aircraft operations a large percentage of all accidents can be attributed to the approach and landing phase of a flight, and it is self-evident that systems designed to carry out automatic landings under all visibility conditions must provide guidance and control better than that provided by the pilot looking at the outside world. Accident rate statistics figure largely in the formulation of the requirements for automatic landing systems, and this led to the adoption (by the United Kingdom certification authorities) of a minimum reliability value of 1 in 10^7; in other words, that a system should not cause a fatal accident more often than one in ten million landings.

The control function during the approach and landing manoeuvre is required on a highly repetitive basis, and although a number of parameters are to be controlled simultaneously, such control is only necessary for a comparatively short period of time, and is therefore most suited to automatic means.

As a prelude to 'blind landing', automatic landing has always been the ultimate aim of control systems designers and aircraft operators, throughout

the development of automatic flight control systems. The history of such developments, the attendant problems, and the attainment of the requisite high safety levels have been so well documented over the past two decades that even a brief summary would constitute a volume in itself. However, in making a broad analysis of available data, it will be found that the many problems which have had to be solved in the development of systems in current use, and having autolanding capability, can be grouped in the following three main areas:

1. Achieving the highest integrity and reliability of systems bearing in mind that they need to be entrusted with very considerable authority over the controls of an aircraft, including the throttles, and in the presence of the ground.

2. The provision of adequate monitoring information on the progress of the approach and landing manoeuvre, and which will enable the pilot to take over under the most critical conditions of a system malfunction in the presence of the ground.

3. The substitution of the pilot's direct vision with an automatic ground guidance system, having an integrity and reliability of the same high order as that demanded of the 'on board' system.

Weather minima

In low visibility operations, the weather limits for landing are given in the following terms.

1. *Runway visual range* (RVR) which is an instrumentally derived value that represents the range at which high-intensity lights can be seen in the direction of landing along the runway. Its readings are transmitted to the air traffic controller who can inform the pilot of the very latest visibility conditions.

2. *Decision height* is the wheel height above the runway threshold by which a go-round must be initiated by the pilot unless adequate visual reference has been established, and the position and approach path of the aircraft have been visually assessed as satisfactory to safely continue the approach or landing.

Minimum values of these two quantities (known as 'weather minima') are specified by the national licensing authorities for various types of aircraft, and for various airports. When the traffic controller advises that RVR is above the specified minimum the pilot may descend to the specified decision height, and if by then he has sighted a sufficiently large segment of the ground to enable him to be confident of his judgement, he may carry on

and land; otherwise he must overshoot, and either enter the holding pattern pending another approach, or divert to an alternative airport. During the approach, the pilot's line of sight is down the glidepath and not along the runway, and this gives rise to another factor, called 'slant visual range', which a pilot must take into account in order to avoid misinterpretation of visual cues.

ICAO categorisation

The foregoing terms are related in a system of categorisation adopted by ICAO, and which describes low-visibility landing capabilities based on the principle that the probability of having adequate short visual reference, for the range of permitted decision heights, should be as high as possible. The definitions of the main categories are graphically illustrated in fig. 10.1.

The three categories also serve as an indication of the stages through
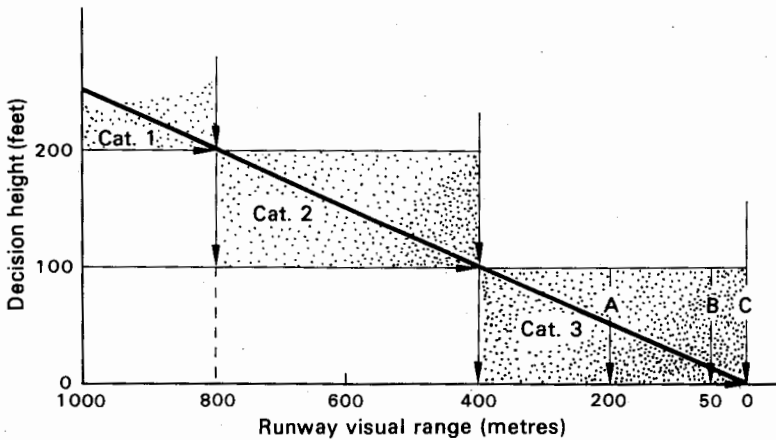
Category 1: Operation down to minima of 200 ft decision height and runway visual range of 800 m with a high probability of approach success.
Category 2: Operation down to minima below 200 ft decision height and runway visual range of 800 m, and to as low as 100 ft decision height and runway visual range of 400 m with a high probability of approach success.
Category 3A: Operation down to and along the surface of the runway, with external visual reference during the final phase of the landing down to runway visual range minima of 200 m.
Category 3B: Operation to and along the surface of the runway and taxiways with visibility sufficient only for visual taxiing comparable to runway visual range value in the order of 50 m.
Category 3C: Operation to and along the surface of the runway and taxiways without external visual reference.

Fig. 10.1 Categories of low-visibility landings.

which automatic approach and automatic landing development progresses, and thereby designate the capabilities of individual automatic flight control systems. In addition, they designate the standards of efficiency of the ground guidance equipment available at airports, namely ILS localiser and glide path, and approach, runway and taxiway lighting.

In connection with automatic landing systems, and in describing low weather minima, the term 'all weather operations' is frequently used; a term which can, and sometimes is, taken to mean that there are no weather conditions that can prevent an aircraft from taking-off and landing success-fully. This is not the case, because no automatic system can, for example, perform the landing task in wind conditions in excess of those for which the aircraft has been certificated, this being primarily governed by the con-trollability characteristics and strength factors of the aircraft. Similarly, no automatic system can land an aircraft on a runway the surface of which, because of water, slush or ice, is not fit for such an operation.

System reliability and redundancy

In chapter 7 (page 226) details were given of the purpose and operation of devices designed to limit the authority of automatic control systems in the event of 'runaway' conditions resulting from malfunctions. While such devices may be incorporated in the more conventional control systems, and thereby be generally effective for the intended purpose down to 'break-off' heights, i.e. approach heights at which a control system is disengaged, this would not satisfy the requirements for systems designed for autolanding. For example, if an aircraft is on the glide path and a 'runaway' occurs in the pitch control channel causing a nose-down attitude then obviously height will be lost, and in using a torque limiting device having a preset value, the aircraft could be well below the glide path before recovery can commence. Thus, there is a minimum altitude to which the device can be used.

The height lost following a malfunction could be reduced by a more severe limiting of control system authority. However, in cases where the flight path may be subject to disturbances resulting from turbulence and wind shear, the situation arises of having to apply rapid correction to the flight path leading to a demand for servo-actuator torque greater than that allowed by the safety devices. Thus, the setting of safety devices is dictated by two conflicting requirements.

1. They must limit the effect of a 'runaway' such that safe recovery can be effected by the pilot;
2. They must allow sufficient authority to the control system so that the required flight path can be followed accurately in the presence of disturbances.

Even with a compromise setting of a safety device, there is the possibility of a height loss under 'runaway conditions', which during an automatic landing would be unacceptable.

A further factor which limits the application of safety devices in the manner of conventional control systems, is their inability to protect against passive failures. While not producing flight path changes directly, these failures would nevertheless mean that the predetermined and accurate flight manoeuvre of automatic landing could not be maintained and so set up an equally dangerous situation.

It follows therefore, that to achieve the objective of automatic landing, the operation of an AFCS must be of such a nature that it will:

1. not disturb the flight path as a result of an active malfunction;
2. have adequate authority for sufficiently accurate control along the required flight path;
3. warn of a passive failure;
4. not fail to complete the intended flight manoeuvre following an active or a passive failure.

In order to resolve the problems which would otherwise have been associated with the application of the more conventional flight control systems, it was considered necessary to adopt the concept of 'system redundancy', i.e. to utilise multiple systems operating in such a manner that a single failure within a system will have an insignificant effect on the aircraft's performance during the approach and landing operation.

In describing failures and the system redundancy concept, it is inevitable that certain terminology must be adopted. It is therefore relevant at this point to review the accepted definitions.

Fail-soft is used to describe the ability of a system to withstand a failure without endangering passenger safety, and without producing excessive deviations from the flight path. An equivalent term adopted in the USA is *fail-passive*.

Fail-operational. This describes a system in which one failure (sometimes more) can occur, but leaves the overall system still functioning, and without causing degradation of performance beyond the limits required for automatic landing and roll-out. Alternative terms are: *fail-active* and *fail-survival*.

Simplex. This term is usually used to define a single automatic control system and its appropriate number of sub-channels. Although various elements of the system may be duplicated, a single failure elsewhere will result in complete unserviceability. In the USA the equivalent term *single (non-redundant)* is used.

Multiplex. This term is applied to a system comprising two or more sensibly independent simplex systems and sub-channels used collectively so that, in

the event of a failure of a system or sub-channel, the remaining systems are alone capable of performing the controlling function. The number of systems and sub-channels adopted is qualified by the terms duplex, triplex and quadruplex as appropriate.

Duplex system is a system of two complete systems or channels which are interconnected, and which together provide continuous control. If comparison monitoring is provided, a duplex system can provide fail-operational capability. The term should not be confused with the terms duplicate-monitored or duplicate-redundancy. An equivalent term adopted in the USA is *dual active with passive monitoring*.

Triplex system is a fail-operational system of three complete systems or channels which are interconnected and which together provide continuous control. In the event of failure of one of the systems or channels, that system or channel is outvoted by the other two and is automatically disengaged; control is therefore continued in duplex. In the event of a further fault in either of the two remaining systems or channels, they will both disconnect, and the aircraft is returned to the pilot in a trimmed and safe attitude. An equivalent term used in the USA is *triple-redundant*.

Duplicate-monitored. This refers to a system comprising two systems in parallel and with separate power supplies. The components of both are designed to be either self-monitoring or to have their outputs checked by parallel comparator circuits. Only one system is engaged at any particular time, the other system being in a follow-up mode, and thereby serving as an active standby. In the event of a fault being shown up by the self-monitors or comparators of either of the systems, control is automatically changed over to the standby system.

Dual-dual. This term is used by some manufactures to define a twin fail-operational control system having twin passive monitoring systems. It should not be considered synonymous to a duplex system, since the control systems may or may not be active simultaneously. In the event of a monitor detecting a failure in its associated system, the second system with its monitor is switched in.

Monitoring. In its strictest sense and, in particular, when applied to multiplex systems, this term defines the process of making comparisons either between two or more outputs (or inputs) or between an output (or input) and a selected datum. The monitoring process can also assume a limiting function; e.g. when it is set up to cause a system to disconnect whenever an output (or input) exceeds a prescribed limit.

Comparison monitor (or Comparator) is one which operates on data supplied from comparable stages in two or more similar systems.

Equaliser. This is a device which adjusts the performance of the sub-systems in multiplex systems to remove differences between sub-system outputs that

may arise other than as a result of fault conditions. Two devices are normally adopted, one called a *gain equaliser* which adjusts the amplitude of response of sub-systems, and the other called a *datum equaliser* which adjusts the steady state output of sub-systems.

Automatic landing sequence

The profile of an automatic approach, flare and landing sequence is illustrated in fig. 10.2 and is based on a system that utilises triple digital flight control computer channels, allowing for redundancy to operate in the fail operational and fail passive conditions already defined. Depending upon the number of channels that are armed and engaged, the system performs what are termed a 'LAND 2' status or 'LAND 3' status autoland. Thus, 'LAND 2' signifies there is dual redundancy of engaged flight control computers, sensors and servos (fail passive operation) while 'LAND 3' signifies triple redundancy of power sources, engaged flight control computers, sensors and servos (fail operational). Each status is displayed on an autoland status annunciator, an example of which is shown in fig. 10.3.

The indicator is of the dual prismatic display type; the upper display indicates the actual autoland capability of the AFCS, while the lower display indicates the degradation from 'LAND 3' capability. Each triangular-shaped prism is actuated by electro-magnetic coils so that the faces of the prisms display the relevant status information as shown. Two test switches are provided; number 1 rotates the upper and lower display faces 'A' into view and number 2 switch rotates the 'B' faces into view. A reset switch (P/RST) can be used to clear the lower display.

During cruise and initial stages of approach to land, the control system operates as a single channel system, controlling the aircraft about its pitch and roll axes and providing the appropriate flight director commands. Since multichannel operation is required for an automatic landing, at a certain stage of the approach, the remaining two channels are armed by pressing an 'APPR' switch on the flight control panel. The operation of this switch also arms the localiser and glide slope modes. Both of the 'off-line' channels are continually supplied with the relevant outer loop control signals and operate on a comparative basis the whole time.

Altitude information essential for vertical guidance to touchdown is always provided by signals from a radio altimeter which becomes effective as soon as the aircraft's altitude is within the altimeter's operating range (typically 2,500 feet).

When the aircraft has descended to 1,500 feet radio altitude, the localiser and glide slope beams are captured, and the armed 'off-line' control channels are then automatically engaged. The localiser and glide slope beam signals control the aircraft about the roll and pitch axes so that any deviations are automatically corrected to maintain alignment with the run-

- Localiser
  Slope Be...
- Off-Line
  engaged.
- Flare Mo...

- Stabiliser trimmed
  nose up.
- Elevator command.

- Flare Mode engaged.
- Elevators bring aircraft
  onto 2 ft/sec descent
  path.
- Throttle retard command.

- Flare Mode disengaged.
- Nose-down command to
  elevators.

ROLLOUT
- ...thrust applied
  ...o-throttle
  ...ged.
- ...ntrol on unit
  ...ged by flight

1500 ft

330 ft

45 ft Gear
Altitude

2 ft/sec
Path

Flare
Path

Glide Slope Beam
Centre Line

Touchdown   Glide Slope
Point   Transmitter

...omatic approach, flare and land.
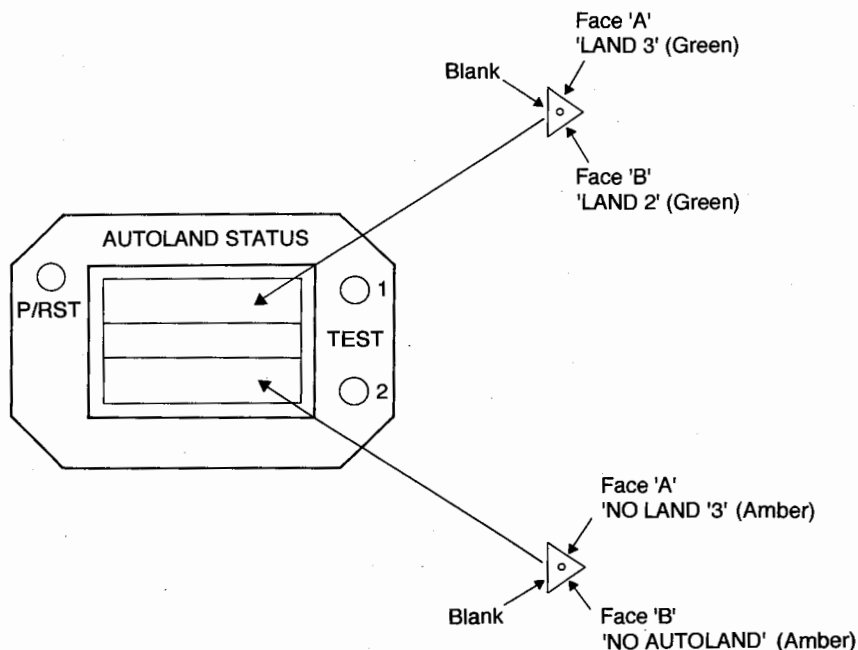
Fig. 10.3

way. At the same time, the autoland status annunciator displays 'LAND 2' or 'LAND 3', depending upon the number of channels 'voted into operation' for landing the aircraft, and computerised control of flare is also armed.

At a radio altitude of 330 feet, the aircraft's horizontal stabiliser is automatically repositioned to begin trimming the aircraft to a nose-up attitude. The elevators are also deflected to counter the trim and to provide subsequent pitch control in the trimmed attitude.

When an altitude is reached at which the landing gear is 45 feet above the ground (referred to as gear altitude) the flare mode is automatically engaged. The gear altitude calculation, which is pre-programmed into the computer, is based upon radio altitude, pitch attitude, and the known distance between the landing gear, the fuselage and the radio altimeter antenna. The flare mode takes over pitch attitude control from the glide slope, and generates a pitch command to bring the aircraft onto a 2 feet/second descent path. At the same time, a 'throttle retard' command signal is supplied to the autothrottle system to reduce engine thrust to the limits compatible with the flare path.

Prior to touchdown, and about 5 feet gear altitude, the flare mode is disengaged and there is transition to the touchdown and roll-out mode. At about 1 foot gear altitude, the pitch attitude of the aircraft is decreased to 2°, and at touchdown, a command signal is supplied to the elevators to

lower the aircraft's nose and so bring the nose landing gear wheels in contact with the runway and hold them there during the rollout. When reverse thrust is applied, the autothrottle system is automatically disengaged. The AFCS remains in control until disengaged by the flight crew.

Autothrottle system

An autothrottle system is of the computer-controlled electro-mechanical type designed to control the thrust of an aircraft's engines within specific engine design parameters. The throttle position of each engine is controlled to maintain a specific value of thrust, in terms of either rotational speed (N1) or engine pressure ratio (EPR) or, a target airspeed, over the full flight regime from take-off to touchdown (see fig. 10.4). It is designed to operate primarily in conjunction with an AFCS to maintain an aircraft's speed and vertical path, and a flight management computer system (FMCS). When an AFCS mode is controlling airspeed, the autothrottle system controls engine thrust to a specific value. When an AFCS mode is controlling the vertical path of an aircraft, the autothrottle maintains airspeed through thrust control. The system computer utilises data from several other interfacing systems as shown in fig. 10.5.

Operating modes

Autothrottle system operation is controlled primarily through the mode control panel (MCP) of the AFCS (see, for example, fig. 2.19 (page 101)) and the indications of the requisite selections necessary for arming the system, and operating it in the relevant modes, are presented on the MCP and on other annunciators. There are basically two modes of operation: take-off and speed control mode.

Take-off mode

This mode is initiated prior to take-off by engaging the FMCS, the computer of which provides the engine rotational speed (N1) limits for each flight profile, and an N1 target speed. The limits and target speeds are displayed by cursors or 'bugs' on the N1 speed indicators. Arming of the autothrottle system for take-off is done by moving the engage switch on the MCP to the 'ARM' position. At the same time, the legened 'ARM' is displayed on appropriate annunciators. Engagement of the system with the servo-actuators controlling the throttle or thrust levers, is done by pressing lever-mounted switches designated as 'take-off/go-around' (TOGA) switches. The servo-actuators then advance the thrust levers at a particular rate to the predicted position to obtain the correct N1 values before a certain take-off speed of the aircraft has been attained. In the system on
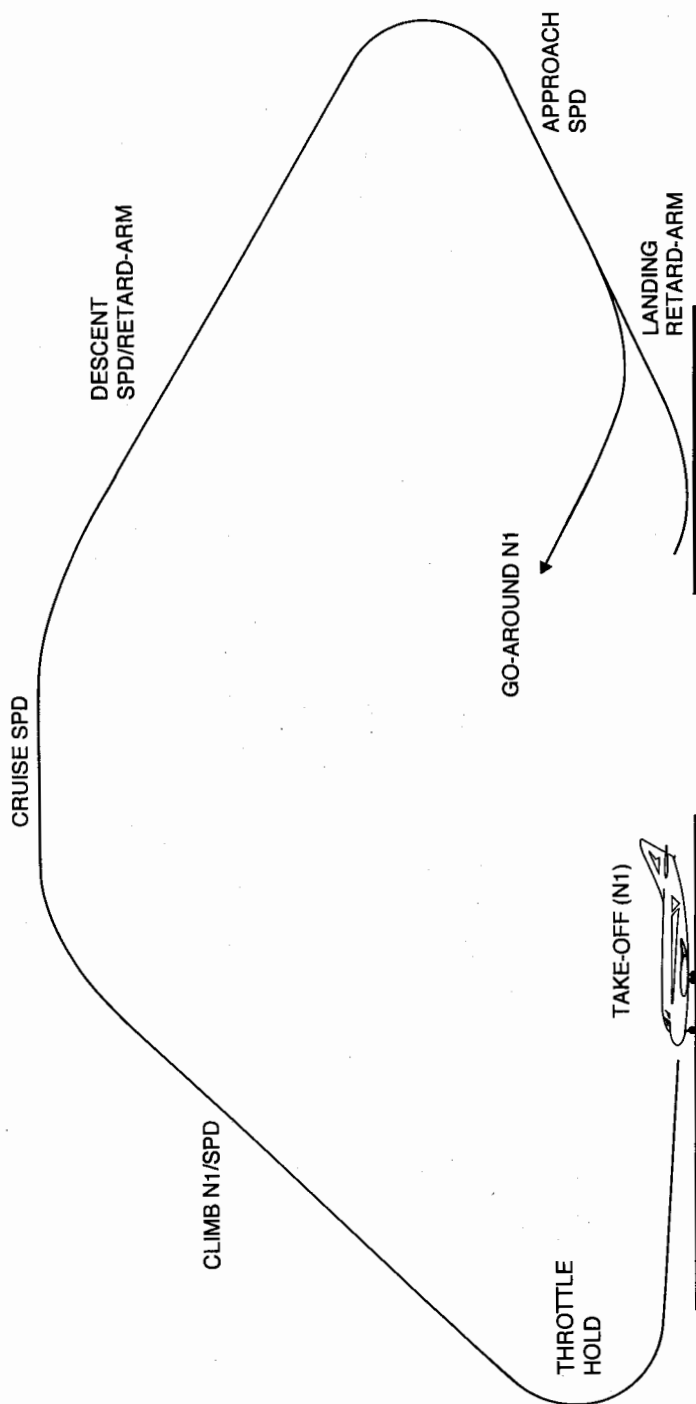
Fig. 10.4 Autothrottle system flight regime.

which fig. 10.5 is based, the advance rate of the thrust levers is 15°/s, and the N1 values are obtained before 60 knots. When this speed is exceeded (in this case, when it reaches 64 knots) an independent speed detector circuit interrupts the operation of the thrust lever servo-amplifiers and actuators. Thus, the autothrottle system enters a 'hold' state and this is indicated by a 'THR HOLD' (throttle hold) annunciator.

The speed detector circuit is also interlocked with microswitches on the main landing gear shock struts, so that in the event of it failing to perform the 'hold' function, the microswitches will do so as a result of their activation after an aircraft lifts off. The circuits of the microswitches also activate timers which normally cause the power supplies to the servo-actuator systems to be re-instated after a specified time from lift-off; in this example, 17.5 s. During this period the aircraft will have climbed to a specific radio altitude (400 feet in this case). If an engine fails and the initial climb is prolonged such that 400 feet is not reached, an additional interlock circuit prevents re-connection of the throttle servo-actuator systems until this altitude is attained. At this stage, the systems are armed to control N1 speeds during the remainder of the climb to the selected cruise altitude.

Speed control mode

This mode can be selected through the MCP of the AFCS, or it is automatically selected by the AFCS when it is not in a speed mode. The speed target set and displayed on the MCP is referred to as the 'MCP SPD'. If vertical navigation (V NAV) control is selected for AFCS operation, the speed target is provided by the flight management computer and is referred to as 'FMC SPD'. The autothrottle system is switched into this speed mode when an aircraft approaches a selected altitude under V NAV control, and will remain in this mode during altitude hold. Airspeed/Mach feedback signals are provided by the air data computers. The autothrottle system limits the airspeed to minimum and maximum safe values independent of the target airspeed values, and it also limits the angle of attack or alpha angle. Minimum airspeed and maximum alpha angle are computed from signals produced by flap position and alpha angle sensors.

When the aircraft begins to descend under V NAV control, the autothrottle system retards the thrust levers to idle, and the legend 'RETARD' is annunciated. The retard rate is nominally two degrees/second, and may be terminated by physically stopping the thrust levers, or allowing them to drive against the idle stop position. The system then goes into the 'ARM' mode which is also annunciated.

When the AFCS captures the glide slope beam, the V NAV mode is disengaged and the autothrottle system switches to the 'MCP SPD' mode, and the speed displayed corresponds to the last value computed by the FMC system. Selection of a desired speed value is made on the AFCS mode
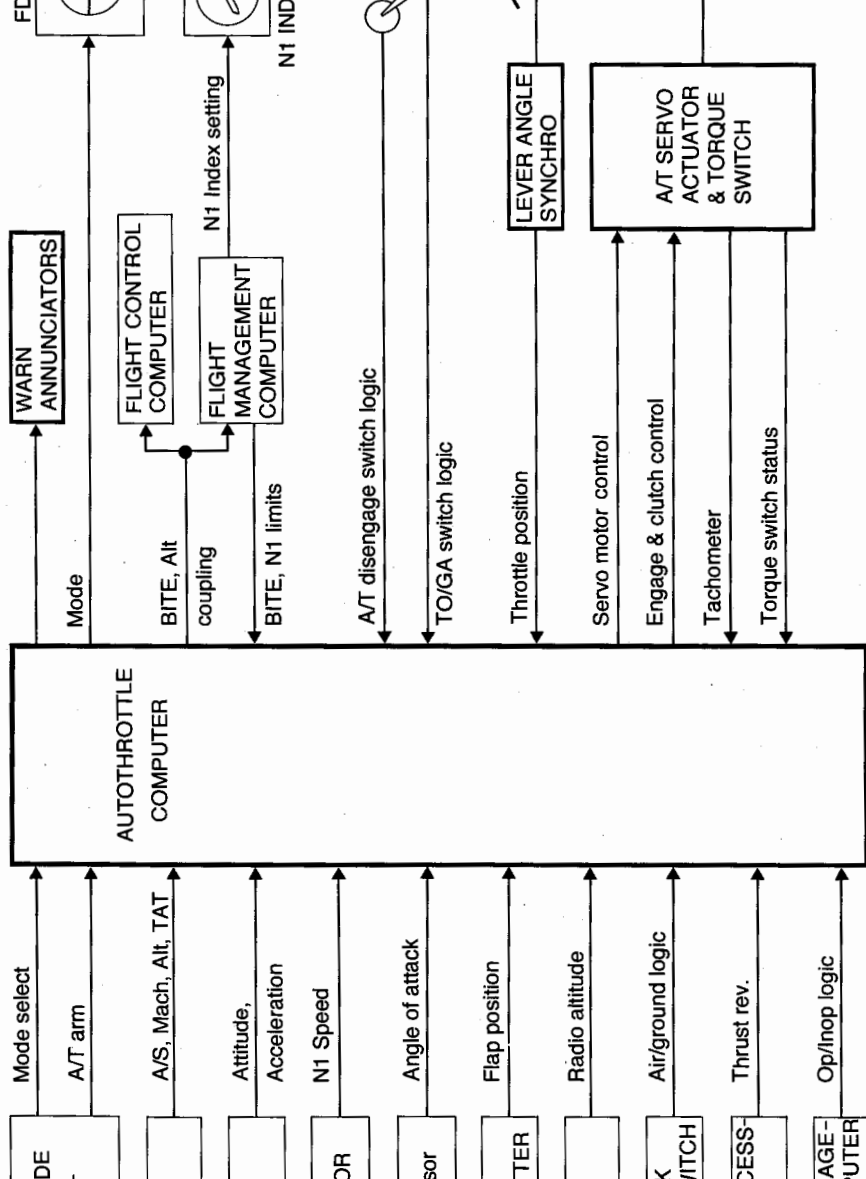
Autothrottle system and signal interfacing.

Labels (top, outputs):
- FD
- N1 INDICATOR / N1 IND
- LEVER ANGLE SYNCHRO
- A/T SERVO ACTUATOR & TORQUE SWITCH
- WARN ANNUNCIATORS
- FLIGHT CONTROL COMPUTER
- FLIGHT MANAGEMENT COMPUTER
- N1 Index setting

AUTOTHROTTLE COMPUTER

Output signals (right side):
- Mode
- BITE, Alt coupling
- BITE, N1 limits
- A/T disengage switch logic
- TO/GA switch logic
- Throttle position
- Servo motor control
- Engage & clutch control
- Tachometer
- Torque switch status

Input signals (bottom):
- Mode select
- A/T arm
- A/S, Mach, Alt, TAT
- Attitude, Acceleration
- N1 Speed
- Angle of attack
- Flap position
- Radio altitude
- Air/ground logic
- Thrust rev.
- Op/Inop logic

control panel. The 'MCP SPD' is annunciated to indicate to the flight crew that pressing the switch will cause the autothrottle system to revert to 'ARM'. In the event of speed mode operation with an engine 'out', the throttles advance together to maintain airspeed, and N1 speed equalisation is replaced by thrust lever equalisation.

Approach gain of the autothrottle system is determined either by glide slope capture or by radio altitude, and flap position. Approach gain provides high gain setting for more precise speed control, and reduced throttle motion during changes of flap position. During an approach in turbulent conditions, the gain tends to cause the system to be high on speed. The degree of overspeed depends on the magnitude and frequency of the turbulence.

During the landing flare manoeuvre, the retard rate of thrust reduction is adjusted so that throttle angle is reduced to idle in 6 s. Retard occurs at 27 feet of radio altitude during an automatic or manual landing. If it is not initiated by radio altitude, it can also occur 1.5 s after an automatic flare. When the aircraft lands and the landing gear shock strut microswitches are activated to their 'ground' position, the thrust levers are moved aft at 8°/s to remove any residual displacement above the idle position. The autothrottle system is automatically disengaged after 2 s.

Go-around mode

This mode can be engaged when the autothrottle system is engaged and the radio altitude is less than 2,000 feet. Initial operation of a 'TOGA' switch on a thrust lever causes the throttles to advance to an angle predicted for a reduced go-around thrust; at the same time, the mode is annunciated by 'GA' on the attitude indicator of the flight director system. The reduced go-around thrust sets the required throttle angle and also the full go-around thrust rating dependent upon the weight of the aircraft and on density altitude. The 'TOGA' switch is then operated a second time causing the engines to 'spool up' to the full thrust rating. The go-around mode also interfaces with the AFCS so that when it is in command at the time of the mode selection, its pitch channel generates a pitch-up command to establish the aircraft in the requisite climb attitude.

TEST QUESTIONS

1 What is meant by the abbreviations RVR and DH?
2 What are the categories of automatic approach and landing?
3 A system in which a failure occurs but leaves the overall system still functioning and without degradation of performance has:
   (a) fail-passive ability.

(b) fail-soft ability.

(c) fail-operational ability.

4 In what manner does a duplicate-monitored system differ from a duplex system?

5 From which source are altitude signals derived during automatic approach and flare?

6 What are the parameters required to be monitored by an autothrottle system?

11
Fly-by-Wire (FBW) Control Systems

Introduction

The potential benefits of fly-by-wire (FBW) control of civil transport air-craft have long been recognised, and some of the first aircraft to which the technique was applied were the Concorde (elevon control) and the Boeing 767 for the control of its wing spoiler system. It was only with the design and development of the Airbus A320, however, that benefits have been fully exploited whereby FBW/digital computer systems have complete control over all flight control surfaces. Another significant feature of this aircraft is that each pilot has a side-stick type of controller (see also chapter 1, page 38) in place of the more conventional control column.

An FBW system has a number of advantages, the most notable of which may be highlighted as follows:

- *Weight saving*. The elimination of the conventional connecting rods, cables and pulleys, together with the mechanical adjustment, feel and intercon-nection units, saves considerable weight even after allowing for the weight of the computers and cable systems which are introduced as replacements.

- *Reduced maintenance times*. Conventional control systems can be very man-power consuming where maintenance is concerned because of the need to make many adjustments to the mechanical control runs *in situ*. More-over, it can take numerous flight tests to correct some conventional control system adjustment problems. Maintenance down-times can be reduced significantly through the greater use of line replaceable units which is possible in FBW control systems and digital flight control systems, once developed fully, should need little or no physical main-tenance action.

- *Gust load alleviation*. Another advantage of the use of FBW, which has been utilised for some years, is that it may be used to control the ailerons in a manner which will alleviate the effects of wind gusts. Normally, when an aircraft encounters an up-gust of wind the immediate effect is

for the lift generated by the wing to increase, causing the aircraft to climb, often quite violently. This has the effect not only of causing discomfort to the passengers but also of putting considerable stress on the aircraft structure. Fast-acting computers can sense the initial increase in lift and deflect the ailerons on both sides of the aircraft upwards, reducing the wing-generated lift and opposing the effects of the gust. This reduces the discomfort of the passengers, reduces the strength requirements of the wings and reduces the fatigue endured by the structure.

- *Automatic manoeuvre envelope protection.* In most commercial types of aircraft, the control systems are such that manoeuvres can be initiated which would be beyond the flight manoevuvre limits these aircraft require in the normal course of their operation. With FBW controlled-aircraft, however, the pilot demands requisite control surface deflections from computers that are programmed such that the command signals generated and distributed, are tailored to match the flight criteria established for the particular type of aircraft. In other words, the computers limit the response of the controls and thereby ensure that flight manoeuvre envelope limits of bank, pitch, yaw, speed, angle of attack and 'g' forces are not exceeded.

- *Improved handling.* From the pilot's viewpoint perhaps the most obvious feature of FBW systems is the improvement in handling characteristics which they can bring. The many small deficiencies in handling of even modern airliners can be eliminated through the efficient use of computers which can make the aircraft responses to control inputs match exactly what the pilot would want them to be. However, given that all modern airliners exhibited acceptable handling already, the achievement of improvements in handling qualities was not in itself a good commercial reason for the introduction of FBW.

- *Fuel saving.* A more compelling commercial reason for the implementation of FBW was the potential for fuel savings to be made if the aircraft could be flown at centre of gravity (CG) positions which would normally have made the aircraft unacceptably unstable. Although there is currently perceived to be a need to allow civil aircraft to be flown manually in the event of complete computer or electrics failure, and thus for the aircraft always to exhibit normal stability, it is likely that at some time in the future, when confidence in FBW systems is complete, flight with relaxed stability will be permitted. Such flight could only be safely achieved through the use of FBW and, once it has been allowed, significant fuel savings could be made. Moreover, the acceptance of 'relaxed stability' would allow major weight savings to be made due to the ability of the designer to plan smaller tail surfaces on such aircraft, perhaps giving weight savings of up to 10% on a typical airliner type aircraft. Thus,

although the generation of aircraft represented by the A320 does not yet capitalise on the fuel saving advantage offered by FBW, future generations of aircraft, e.g. the A330 and A340, might do so, and to the extent of utilising a degree of stability relaxation in the cruise.

In this chapter we shall concern ourselves only with the architecture, control laws, and alternate operation of the control system as applied to the A320 which is currently in worldwide commercial service.

Advanced FBW concepts

The introduction of FBW to an aircraft could simply provide a computer link between the pilot's controls and the control surfaces; in other words, a movement of the pilot's controls would cause a corresponding and proportionate movement of the control surfaces. Such a level of development would provide the weight savings promised by FBW but would do little to improve the handling of the aircraft, and would not advance the technology very far towards allowing aircraft with relaxed stability to be flown. In order to achieve either of these two goals the computer must be made to do a little more and, typically, this would be to cause the aircraft to respond in a certain manner to the pilot's inputs by driving the controls as appropriate. The pilot could be enabled, for example, to demand changes in the pitch rate or the flightpath of the aircraft rather than demand simple control surface movements. Such an FBW system is often called an 'Active Control' system because the control system itself is more than a passive conveyor of instructions.

If active control systems are to be effective then both the computers and the actuators employed in them must be fast-acting to minimise the destabilising effects of control delays. The speeds of reaction required will be dependent to an extent on the natural handling characteristics of the aircraft, an unstable aircraft requiring a much faster acting system than one with benign handling. The general layout of the components of an active control system (the number of computers, their functions, the manner of cross-voting) can be called its *architecture* whereas the manner in which the computers are expected to create control surface and thus aircraft movement can be called the *control laws*. In all aircraft systems design allowance must be made for partial or total failure of elements in the system, and provision must be made for *alternate* operation.

Architecture

In order to provide some redundancy and to improve safety by allowing comparisons to be made of the output demands of more than one com-

ELAC	SEC	FAC
Elevator/Aileron Computer	Spoiler/Elevator Computer	Flight Augmentation Computer
Elevators Ailerons Trimmable Horizontal Stabiliser	Spoilers Elevators Trimmable Horizontal Stabiliser	Yaw Damping Rudder Travel Limits Rudder Trim
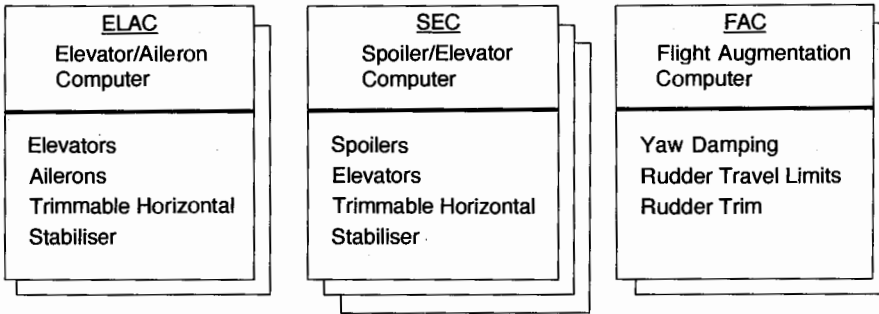
ELAC   Elevator/Aileron Computers (2)

SEC   Spoiler/Elevator Computers (3)

FAC   Flight Augmentation Computers (2)

Fig. 11.1 Computer arrangement. It can be seen that the computer arrangement is such that neither the ELACs nor the SECs provide the only control in either the pitch or the roll axes and this is designed to decrease the risk of a common design fault having an uncontained effect on the aircraft. Furthermore, redundancy and safety are increased through the use of different microprocessor types, different suppliers, physical segregation of signalling lanes and the division of each computer into two physically separated units. The power supplies are also segregated and, as with most other aircraft, the individual control surfaces are signalled by different lanes and powered by different hydraulic systems. The pitch control is shown in fig. 11.2.

putation, it is normal for an active control system to comprise several different computers. Each one of these computers will have specific functions and no single computer will be permitted to exercise control without its commands being monitored by at least one other computer. Ideally the computers should be constructed separately and their programs written independently in order to avoid the possibility of a design fault or software error being common to them all. The A320 Electronic Flight Control System (EFCS) employs seven principal flight control computers as shown in fig. 11.1 and communication is by an ARINC 429 Data Bus.

Control laws

Regardless of the architecture of the flight control system, control laws must be designed which determine how the pilot's control demands are to be translated into control surface movements. As was stated earlier, an active control system will do more than simply pass on control surface movements in direct proportion to the amount of movement made by the pilot controls. Instead, the flight control system will be programmed to provide a particular form of aircraft response as the result of the pilot's input. Control in

295
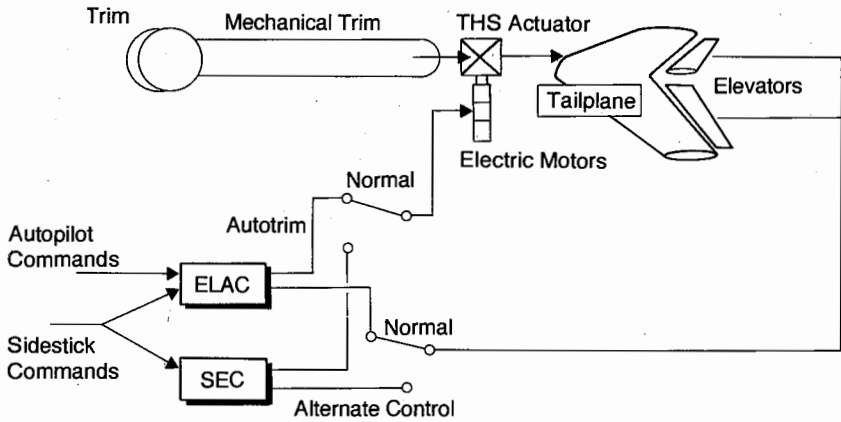
Fig. 11.2 Pitch control. It can be seen that the ELACs control the aircraft in pitch in the so-called *normal* control law and that they do so by sending commands to the left and right hand elevators and also by sending longer term trim commands to the trimmable horizontal stabiliser (THS). In the event that the ELACs are unserviceable or unavailable due to failures in their supplies, two of the three SECs, Nos 1 and 2, will take over their role to control the aircraft in pitch; the so-called *alternate* control law. Under alternate control the aircraft should handle almost exactly as in normal control but many of the envelope protection features are unavailable. These include the high angle of attack protection and pitch attitude protection which form a part of the normal control laws. A further degradation requiring, for example, the loss of all three inertial reference systems (IRSs) would cause the selection of the *pitch direct law* in which movement of the sidestick controller in pitch is translated directly into movement of the elevator control surfaces, the only modification to the action being that the total elevator movement available is determined by the position of the CG, and of the flaps. A complete failure of the EFCS in pitch, requiring failure of both ELACs and both SEC 1 and 2, would require that the aircraft is flown in pitch through the use of the pitch trim wheel in a condition known as *mechanical* pitch back up. Fig. 11.3 shows the levels of redundancy available in the pitch control system.
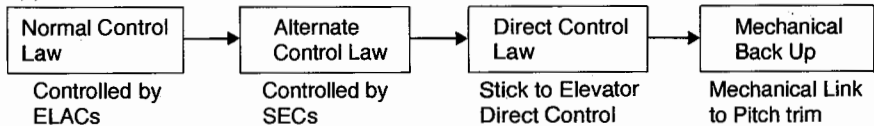
Fig. 11.3 Pitch control redundancy. Roll control is provided by both the ELACs and the SECs, the former controlling the ailerons and the latter controlling the spoilers. In *Normal* control law both types of computer contribute to roll control, but in the event of a failure of one channel the other can assume total authority, albeit with different control laws. The roll control architecture is shown in fig. 11.4.
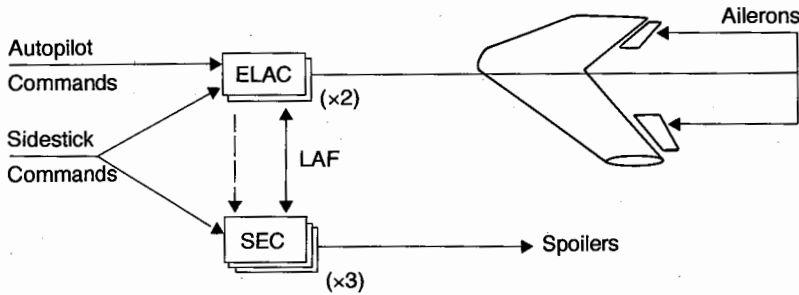
Fig. 11.4 Roll control. Yaw control is achieved through signalling from the FACs to the rudder actuators although the FACs themselves receive their input signals from the ELACs and the autopilot. A mechanical connection is retained between the rudder pedals and the rudder actuators to allow for control of the aircraft in roll (through the secondary effect of yaw) in the event of a complete failure of the EFCS or of the electrical supplies. Total mechanical backup is thus available through the use of the pitch trim wheel and the rudders.

the pitching plane is the most complex of the three and this is considered in most detail here.

Pitch control

Before considering the control requirements of an airliner design, it is worth considering the normal stability and control characteristics of a conventional aircraft.

Conventional stability and control characteristics
A conventionally stable aircraft is 'angle of attack stable', i.e. if the aircraft is disturbed from its trimmed angle of attack, for example by a gust of wind, it will tend naturally to return again to the trimmed angle of attack. Thus an upgust, causing an increase in angle of attack and an increase in lift, will be met by a natural tendency for the aircraft to pitch down to regain the original angle of attack and, coincidentally, to a reduction in the height gain. It is the relationship between the position of the CG and the wing's centre of pressure, together with the design of the horizontal stabiliser which cause this to happen. With this form of aircraft, movement of the elevator control by the pilot causes a change in the trimmed angle of attack and so causes the aircraft to pitch up or down in order to achieve the new angle. When accelerating in level flight it becomes necessary for the pilot to trim forwards not just because if he did not then the aircraft would start to climb (increased airspeed = increased lift), but also because the aircraft would tend to pitch up as speed increased due to the natural stability of the aircraft forcing it to remain at the original angle of attack (climbing would otherwise tend to reduce the angle of attack).

If so desired, an aircraft with active controls could be made to respond in the same manner as a naturally stable aircraft either by providing only a direct link between the control column and the control surfaces or by introducing an angle of attack feedback into the control loop. In the former case, where a direct control link was provided, the aircraft's handling would remain unimproved on its natural form but in the latter case the control loop could be manipulated if necessary to improve the aircraft's handling whilst retaining a natural angle of attack stability and the consequent 'natural feel'. Although angle of attack feedback control systems lack many of the advantages of other, more complex systems, as seen below, they do provide the pilot with the natural pitch changes which he knows should accompany changes in speed.

Pitch rate demand/attitude hold

This is a control system in which the pilot demands changes in the pitch rate through control column inputs. In the short term, the response of the aircraft to the pilot's inputs is similar to that of a conventional aircraft, but in the longer term the effects of his inputs are quite different. If a certain control column position demanded zero pitch rate and the pilot moved it rearwards then he would be demanding a positive rate of pitch, which is a similar response to that of the conventional aircraft. This *constant* rate would, however, be maintained as long as the control input was retained, even though the consequently reducing airspeed would normally, in a conventional aircraft, lead to a reduction in the pitch rate, eventually to zero once again. Considering the effects of external disturbances; if an aircraft fitted with a pitch rate demand control system was flying level, in trim, at a certain pitch angle and at zero pitch *rate* and it was hit by a gust then, although the increase in angle of attack and lift might cause the aircraft to climb, the natural tendency for it to pitch down would be resisted by the control system. This is because no change in pitch rate (from zero) has been demanded by the pilot. This aircraft will tend to 'heave' in turbulence but will tend not to pitch unless demanded to do so by the pilot. Changes in flightpath may thus occur without being demanded, and undemanded changes in airspeed will require a pilot input if the flightpath is to remain unchanged. This system has the advantage that a certain amount of feedback is provided to the pilot of changes in airspeed (although not as much as in the natural aircraft) and yet a certain pitch stability in gusts is achieved. It has the disadvantage that the required flightpath is not maintained automatically and so continual pilot inputs may be required, for example on the approach.

'g' demand/flightpath hold

If we wished, the control system could be designed to maintain a constant, trimmed, 'g' level rather than a constant pitch rate, as this is the parameter which the pilot is usually attempting to command (through changes in

pitch, achieved by changes in angle of attack). Constant 1 'g' flight is of course the same as a steady, level flightpath and so 'g' demand can be considered to be the same. In this case the control system's response to undemanded changes in angle of attack or airspeed, brought on by wind gusts, for example, would be to pitch the aircraft up or down as required in order to restore the aircraft's flightpath to its original value. Similarly the pilot, in moving the controls, would be demanding changes to the aircraft's flightpath and not to its angle of attack. This is achieved by introducing a flightpath feedback to the flight control loop; in simple terms the pilot demands a certain rate of change of flightpath through stick movement and the achieved rate of change of flightpath is fed back to the control system which continues to output elevator demands until the required rate is achieved. Note that rate of change of flightpath is effectively normal acceleration or 'g' and so this can be described as a 'g'-demand system.

The advantage of this type of control system is that once the pilot has demanded a steady flightpath, e.g. level flight or alternatively a steady descent towards the runway, he should not need to make further inputs in order to retain that flightpath. The flight control system will make the necessary elevator inputs to achieve the constant flightpath demanded and will even trim the tailplane over a period of time to remove the need for constant elevator deflections to be made. A disadvantage of such a system is that it removes from the pilot the natural feedback of changes from the trimmed airspeed that an angle of attack–stable system would have (i.e. the need to push forwards on the control stick as speed increases from the trimmed value in order to prevent the aircraft from pitching up and climbing). Another potential problem with the use of a flightpath demand control system comes from the fact that there is often a noticeable delay between the achievement of a new pitch rate and the consequent achievement of a change in the flightpath. This is due to the inertia of the aircraft and is most noticeable at lower airspeeds as an apparent 'lag' in the aircraft's response. Furthermore, because this type of control system would be attempting to achieve changes in the 'lagging' flightpath through earlier changes in pitch rate, it can tend to create too much activity in pitch, especially in turbulence.

The C* control law

A form of pitch control law which has gained acceptance as being suitable for the public transport role is the so-called C* control law (pronounced C-star). This is principally a flightpath command control law which, at low airspeeds, also includes an element of pitch rate command. In general terms the pilot is able to select a particular flightpath angle, e.g. level flight, and, once the aircraft has stabilised at that angle, if he released the controls the control system would ensure that the flightpath angle was retained. The introduction of an element of pitch rate command at lower airspeeds helps to overcome the problems of lag and over-activity in pitch in the achieve-

ment of flightpath changes referred to above; it makes the control system feel more 'rigid'. It has been found to be rather difficult to land aircraft employing flightpath command control systems, partly because the pilot might have to 'push down' to land following the initial flare, and this is an unnatural sensation. As a result, it is not unusual for these control laws to be modified for the landing phase of flight such that the pilot is required to pull back progressively during the flare. A way of achieving this is to cause the aircraft automatically to pitch down slightly as it approaches the ground, requiring the pilot to apply back stick in order to oppose the motion.

The A320 uses a form of C* control law as its Normal Law in pitch. It is modified in the flare by a law which memorises the pitch attitude at 50 ft above the ground and subsequently decreases the attitude by a fixed amount over a period of 8 s, starting at 30 ft above the ground. This pitch down can be overcome by the pilot and it provides the natural 'flare' feeling referred to above. In turns, the control laws are designed such that the pilot does not need to apply any additional back pressure in order to maintain level flight unless the bank angle is greater than 33° (this despite the fact that in turning level flight the required 'g' level is greater than in straight level flight). Protection is provided against overspeeding, excessive angle of attack, excessive pitch angles (+30° to −15°) and excessive manoeuvre (g).

Roll control

The control laws required for control in roll are much less complex than those used in the pitching plane, at least in concept. A conventional aircraft is required to exhibit a degree of lateral stability whereby, if bank is applied and the controls are subsequently released then the aircraft will slowly return to wings level. The principle reason for this requirement is to ensure that the reverse does not happen, i.e. that the aircraft does not progressively roll further into the turn when the pilot's attention is distracted elsewhere. As a result it is normal for the pilot to have to hold a small amount of aileron towards the direction of turn in order to maintain a bank angle. With an FBW aircraft, where it can be demonstrated that the control system will not allow the bank angle to increase uncommanded, the requirement for positive lateral stability can be relaxed and control systems can be designed which will allow a selected bank angle to be maintained without the pilot's having to keep some control deflection applied.

The A320 employs roll rate command with bank angle stability up to 33° of bank. Thus zero stick deflection demands zero roll rate and no change in bank angle, and increasing stick deflection causes an increasing amount of roll rate to be selected. The bank angle stability allows the bank angle to be maintained without control inputs being required for bank angles up to 33°. Beyond that angle, and up to the limiting angle of bank achievable, 67° normally, increasing amounts of control deflection are required from the pilot.

Control inceptors

Naturally, if an FBW system is to function it must be informed of the pilot's control intentions and so pick-offs must be attached to the pilot's controls (inceptors). Airbus have dispensed with the conventional control columns and yokes and have replaced them with sidesticks. These are carefully designed so that the force gradient is consistent with the pilot's ability to apply force to a side-mounted stick, the maximum force required being 10 daN (decaNewtons) in pitch and 3 daN in roll. It is interesting to note that pilots are able to apply a greater force inwards than they can apply outwards with a sidestick and this has been recognised by the provision of a shallower force gradient outboard than that provided inboard (maximum of 2 daN vs 3 daN). Also the fore/aft axes of the centresticks are angled inboard in recognition of the natural orientation of the pilot's wrist. The inceptors in the A320, including the thrust levers, are not back-driven by the autopilot and, as the sidesticks are not interconnected, they do not move in response to movements imparted by the other pilot.

Appendix 1: Fixed-wing Aircraft/ AFCS Combinations

Aerostar 601	King KRC 200; Century IV; Bendix FCS810
Airbus A300; 310	SFENA
A320	SFENA
A330	
340	
BAC 1-11 201 AC/300 series	Bendix PB20; Elliot E2000/E2000
400 series/500 series	Bendix PB2000/E2000,E2100,E2200
Beechcraft Baron 55,58	Bendix FCS810; Century IV; Century III; King KFC 200, KFC 250; Collins AP107
95-B55	Century III; Century IV; Beech B-5, H14
Queenair 65-80/70/A65	Brittain B4; Beech H14/Bendix M4/ Bendix M4
Duchess 76	Century II; Century IV
Duke 60	Century IV
King Air 90	Sperry SPZ200; Century IV; Collins AP106; Bendix M4 series
King Air A100/200	Collins AP105; Beech H14/Collins APS80, Collins AP105; King KFC 300
Bonanza A36	Bendix FCS810; Century I
C23	Beech Navomatic
F33A	Century III; King KFC 200
Boeing 707 series/720 series	Bendix PB20/PB120
727 100 series/200 series	Sperry SP50/SP150
737/-236	Sperry SP77/SP177
737-300	
400	
747	Sperry SPZ1
747-400	Collins
757⎫	Collins FCS-700
767⎭	
Britten-Norman Islander/Trislander	Brittain B5; Collins AP107; Century I/Collins AP107
Cessna 172 series, 177, 182 series,	Navomatic 200, 300 series; Century I, II

series 210, 310, 330	Navomatic 400 series
series 340	Navomatic 400 series; King KFC 200
series 400/500	Navomatic 800, 1000 series; King KFC 200/Sperry SPZ200, SPZ500
DHC Twin Otter/Dash 7	Collins AP106/Sperry SPZ700
Dash 8	Honeywell Sperry
Douglas DC-3/DC-6/DC-8	Sperry A-3/A12/SP30
DC-9/DC-10	Sperry SP50G/Bendix PB100
MD-11	
Embraer Bandeirante	Bendix M4 Series
Brazilia	King KFC300
Xingu	Sperry SPZ200
Falcon 20E/20F/50	Collins AP104, 109/Collins 105/Sperry SPZ600
Fokker F27 200 and 400/F28 4000	Smiths SEP2/SEP6
F50	Sperry EDZ 801
F100	Rockwell Collins
Grumman AA/Traveller	Century I
5A Cheetah	Century I, II
5B Tiger	
GA7 Cougar	Century III
G159 Gulfstream I	Collins AP104, 105
G1159 Gulfstream II	Sperry SP50G
Gulfstream IV	
HP Jetstream – I/200/300	Bendix M4 series; Smiths SEP10/ Bendix M4 series/Sperry SPZ500
Jetstream 31	
748-1/-2A/-2B	Smiths SEP2/SEP2, SEP6; Sperry SPZ500/SEP2
125-1, 1B, 3B	Collins AP103, 104
−400 series	
−600 series/700 series	Collins AP104/AP103; APS180
146	Smiths SEP10
Learjet 35A, 36A	Bendix FCS200
Lockheed – 1011 Tristar	Collins FCS-110
Mooney M20 Series	Brittain B2C, B5; Century 21; King KFC200
Partenavia P68B/P68C	Century I; King KFC200/Century III
Piper PA23-160/-250	Century I, Altimatic II, III, V; Bendix FCS810
24-180	Century I
28 series	Altimatic III; Century I, 21; Mitchell Autocontrol II, III; Piper Autoflite II
30/31 series/32 series	Altimatic III; Century II; Autocontrol III/Altimatic III, V, X; Bendix FCS

Appendix 1

34 series	810, 870; King KAP 200, KFC 200, KFC 250, Century 41; King KAP 200, KFC 200; Autocontrol III
39/42	Altimatic III; King KAP 200, KFC 200 Altimatic III/King KFC 300
Rallye TB10/235	Century I/Badin Crouzet
Rockwell 112/114/500	Century I, II, III/Century I; Altimatic III; King KFC 200/Century III
685/690/840	Bendix FCS810; M4/Bendix M4; Collins AP106/Collins AP106
Sabre 65	Collins APS 80
Shorts SD-330 SD-360	
Socata TB10	Century I, III
Swearingen SA 226T (B)	Collins APS 80
Viscount 700 series/800 series	Bendix PB10; Collins AP101/PB10; Smiths SEP2

Appendix 2: Helicopter/AFCS Combinations

Airframe manufacturer	Type	AFCS available
Augusta	A109	Sextant Avionique AFCS 95 Honeywell SPZ-7000
Aerospatiale (Eurocopter)	Gazelle	SFIM 85 SFENA
	SA 365 (Dauphin)	SFIM 155D Honeywell SPZ-7000
	Super Puma (AS 330/332)	SFIM AFCS 165
	BO-105 BK 117	Honeywell SPZ-7000
	AS 350 (Ecreuil/Squirrel)	SFIM 85T31
	AS 355	SFIM 85T31
Bell	206 series	Bendix King KAP 150H Ferranti FAS 2 Collins APS 841H Sextant Avionique FCS 200B
	212	Smiths Industries – Newmark LN450 Honeywell SPZ 7000 Sextant Avionique AFCS 95
	412	Honeywell SPZ-7600
	222/230 series	Honeywell SPZ-7000 Sextant Avionique FCS 200L/AFCS 90
	214ST	Sperry Helipilot 7600
Boeing Vertol	234LR	Hamilton Standard
Hindustan Aeronautics	ALH	SFIM 166

Appendix 2

Airframe manufacturer	Type	AFCS available
MacDonnell Douglas	MD 500 series	Astronautics 131500
	NOTAR	Astronautics 168670
Sikorsky	S61 series	Smiths Industries – Newmark LN400/450
	S76	Hamilton Standard SPZ-7000 SPZ-7600
	S70 (Blackhawk series)	Hamilton Standard
Westland	Wessex 60/1 (military)	Smiths Industries – Newmark Mk 19
	WG 30	Smiths Industries – Newmark Mk 31
	EH-101	Smiths Industries – Newmark
	Sea King	Smiths Industries – Newmark Mk 31 AFCS/SN500

Appendix 3: Acronyms and Abbreviations Associated with AFCS, Equipment and Controlling Signal Functions

ACCEL	ACCELerometer
ACQ	ACQuire (prefixed by a condition, e.g. ALT ACQ)
A/D	Analogue to Digital
ADC	Air Data Computer
ADI	Attitude Director Indicator
AFCS	Automatic Flight Control System
AFS	Automatic Flight System
AGC	Automatic Gain Control
AGS	Automatic Gain Stabilisation
AHRS	Attitude and Heading Reference System
ALG ARM	Align Arm
ANN	ANNunciator
AOSS	After Over Station Sensor
AP, A/P	AutoPilot (suffixed by condition, e.g. ENG, DISC)
APFDS	AutoPilot and Flight Director System
APMS	Automatic Performance and Management System
APPR OC	APProach On Course
APS	Altitude PreSelect
APSB	APS Bracket
ARINC	Aeronautical Radio INCorporated
ARM	ARMed (prefixed by condition, e.g. LOC ARM, VOR ARM)
AS, A/S	Airspeed
ASA	Autoland Status Annunciator
AT	AutoThrottle
ATS	AutoThrottle System
AT/SC	AutoThrottle/Speed Control
ATT	ATTitude (may be followed by condition, e.g. ATT HOLD)
ATT ERR	ATTitude ERRor
AUTO APPR	AUTOmatic APProach
B/A	Bank Angle

Appendix 3

BARO	BAROmetric
BB	Bar Bias
B/B	Back Beam
B/C, BC, B/CRS	Back Course
B/D	Bottom of Descent
BITE	Built-In Test Equipment
BRG	BeaRinG
CADC	Central Air Data Computer
CAP	CAPture (prefixed by a condition, e.g. LOC CAP, NAV CAP)
CBB	Collective Bar Bias
CDU	Control and Display Unit
CE	Course Error
CLK	CLocK
CMD	CoMmanD (prefixed by another abbreviation, e.g. FD CMD)
CP	Control Panel
CPL	CouPLed (prefixed by condition, e.g. ROLL, PITCH, APPR)
CRS	CouRSe
CT	Control Transformer
CW	Caution and Warning
CWS	Control Wheel Steering
D/A	Digital to Analogue
DADC	Digital Air Data Computer
DEVN	DEViatioN
DES	DESired (suffixed by condition, e.g. DES TRK, DES CRS)
DG	Directional Gyroscope
DH	Decision Height
DIFCS	Digital Integrated Flight Control System
DISC	DISConnect
DISP	DISPlacement
DLC	Direct Lift Control
DME	Distance Measuring Equipment
DMUX	DeMUltipleXer
DSR TK	DeSiRed TracK
DTG	Distance-To-Go
DU	Display Unit
EADI	Electronic Attitude Direction Indicator
EFCU	Electronic Flight Control Unit
EFIS	Electronic Flight Instrument System

EHSI	Electronic Horizontal Situation Indicator
EHSV	Electro-Hydraulic Servo Valve
ENG	ENGage
EO	Easy-On
EX LOX	EXpanded LOCaliser
FAC	Flight Augmentation Computer
FAWP	Final Approach WayPoint
FCC	Flight Control Computer
FCEU	Flight Control Electronic Unit
FCU	Flight Control Unit
FCES	Flight Control Electronic System
FD, F/D	Flight Director
FGS	Flight Guidance System
FIS	Flight Instrument System
FL CH	Flight Level CHange
FMA	Flight Mode Annunciator
FMC	Flight Management Computer
FMCS	Flight Management Computer System
FMS	Flight Management System
FODTS	Fibre-Optic Data Transmission System
FS	Fast Slew
FTR	Force Trim Release
GA, G/A	Go-Around
GS, G/S	Glide Slope
HARS	Heading and Attitude Reference System
HDG	HeaDinG (can be suffixed by condition, e.g. HDG HOLD, HDG SELect)
HLD	HoLD
HSI	Horizontal Situation Indicator
IAS	Indicated AirSpeed
IAWP	Initial Approach WayPoint
ICU	Instrument Comparator Unit
ILS OC	Instrument Landing System On Course
INS	Inertial Navigation System
INTLK	INTerLocK
INWP	INtermediate WayPoint
IRS	Inertial Reference System
IRU	Inertial Reference Unit
IVS	Instantaneous Vertical Speed
IVV	Instantaneous Vertical Velocity

LAU	Linear Accelerometer Unit
LBS	Lateral Beam Sensor
LNAV	Lateral NAVigation
LOC	LOCaliser
LRRA	Low-Range Rader Altimeter
LRU	Line Replaceable Unit
LSSAS	Longitudinal Static Stability Augmentation System
LSU	Logic Switching Unit
LVDT	Linear Voltage Differential (also Displacement) Transformer (Transmitter)
MADGE	Microwave Aircraft Digital Guidance Equipment
MALU	Mode Aunnunciation Logic Unit
MAN	Manual
MAP	Mode Annunciator Panel
MAWP	Missed Approach WayPoint
MCDP	Maintenance Control Display Panel
MCP	Mode Control Panel
MDA	Minimum Descent Altitude
MM	Middle Marker
MPU	Microprocessor Unit
MSU	Mode Selector Unit
MUX	Multiplexer
NCD	No Computed Data
NCU	Navigation Computer Unit
ND	Navigation Display
NDB	Non Directional Beacon
NOC	NAV On Course
OC,O/C	On Course
OD	Out of Detent (may be prefixed, e.g. CWS OD)
OM	Outer Marker
OSS	Over Station Sensor
P ATT	Pitch ATTitude
PBB	Pitch Bar Bias
PCA	Power Control Actuator
PCPL	Pitch CouPLed
PCWS	Pitch Control Wheel Steering
PECO	Pitch Erection Cut-Off
PFD	Primary Flight Display
P HOLD	Pitch HOLD
PSAS	Pitch Stability Augmentation System
PSM	Power Supply Module

PSO	Phase Shift Oscillator
P SYNC	Pitch SYNChronisation
RA, R/A	Radio (Radar) Altimeter
RBB	Roll Bar Bias
RCPL	Roll CouPLed
RCWS	Roll Control Wheel Steering
REF	REFerence
REV/C	REVerse Course (sames as Back Course)
RG	Raster Generator
R/HOLD	Roll HOLD
RLS	Remote Light Sensor
RMI	Radio Magnetic Indicator
RN, RNAV	Area NAVigation
RN/APPR	Area Navigation APPRoach
RSAS	Roll Stability Augmentation System
RTE DATA	RouTE Data
RVDT	Rotary Voltage Differential Transmitter (Transformer)
SAM	Stabiliser Aileron Module
SAS	Stability Augmentation System
SBY	StandBY
S/C	Step Climb
SCAT	Speed Command of Altitude and Thrust
SCM	Spoiler/Speedbrake Control Module
SEL	Select
SFCC	Slat/Flap Control Computer
SID	Standard Instrument Departure
SG	Symbol Generator (Stroke Generator)
SGU	Symbol Generator Unit
SPD	SPeeD (Airspeed or Mach hold)
SRP	Selected Reference Point
SS	Slow Slew
SSEC	Static Source Error Correction
STAR	Standard Terminal Arrival Route
STS	STatuS (prefixed by a function, e.g. TRACK STS)
STCM	Stabiliser Trim Control Module
TACAN	TACtical Air Navigation
TAS	True Air Speed
T/C	Top of Climb
TCC	Thrust Control Computer
TCS	Touch Control Steering
T/D	Top of Descent

Appendix 3

TKE	TracK angle Error
TK CH	TracK CHange
TMC	Thrust Management Computer
TMS	Thrust Management System
TMSP	Thrust Mode Select Panel
TRP	Thrust Rating Panel
TTG	Time To Go
TTL	Tuned To Localiser
TURB	TURBulence
VAR	VARiable
VBS	Vertical Beam Sensor
VGU	Vertical Gyro Unit
VLD	VaLiD (usually suffixing a condition, e.g. VG VLD, FLAG VLD)
VNAV	Vertical NAVigation
VOR	Very-high-frequency Omnidirectional Range
VOR APPR	VOR APPRoach
VOR OC	VOR On Course
VORTAC	VOR TACtical (Air navigation)
VS	Vertical Speed
WO, W/O	WashOut
WPT	WayPoinT
WXR	Weather Radar transceiver
XTK DEV	Cross TracK DEViation
XTR	TRansmitter
YD, Y/D	Yaw Damper
YDM	Yaw Damper Module

Appendix 4: Logic Circuits

The operation of the majority of units comprising automatic flight control systems is based on the application of solid-state circuit technology, i.e. components such as resistors, rectifiers, and internal switches that are normally interconnected as separate discrete components, are all 'embedded' in micro-size sections of semiconductor material. Apart from the vast reduction in dimensions, this form of integration also makes possible the production of circuit 'packs' capable of performing a vast number of individual dedicated functions. Thus, in knowing the operating parameters of a system overall, and the functions that constituent units are required to perform, the complete circuitry of a system is built up by interconnecting selected functional packs. The packs consist of basic decision-making elements referred to as *logic gates*, each performing combinational operations on their inputs and so determining the state of their outputs.

As far as diagrammatic presentation of the foregoing circuits is concerned, greater use is made of a schematic form depicting interconnected blocks and a variety of special logic symbols, each representing a specific circuit network 'hidden away' in the semiconductor material. The study of a system's operation is therefore based more on the interpretation of symbols and the logic state of signal functions at the various interconnections of the circuit, rather than tracing through diagrams that depict all internal circuit details in pure theoretical form.

Logic gates and symbols

Logic gates are of a binary nature, i.e. the inputs and the outputs are in one of two states expressed by the digital notation 1 or 0. Other corresponding expressions are also frequently used, as follows:

　　1 – on; true; high (H); closed; engaged
　　0 – off; false; low (L); open; disengaged

The 1 and 0 state designations are arbitrary. For example, if the states are represented by voltage levels, one may be positive and the other 0 V; one

may be negative and the other 0 V; one may be positive and the other negative; both may be positive, or both may be negative. The applications of logic to a system or device may therefore be further defined as follow:

1 Positive logic, when the more positive potential (high) is consistently selected as the 1 state.
2 Negative logic, when the less positive potential (low) is consistently selected as the 1 state.
3 Hybrid or mixed logic, when both positive or both negative logic is used.

The circuits to which digital logic is applied are combinations of three basic gates performing functions referred to as 'AND', 'OR' and 'NOT': the latter being an inverting function and giving rise to two other gates referred to as 'NAND' and 'NOR'.

Gate circuits are designed so that switching is carried out by either junction diodes, or transistors, or by a combination of both. In order to simplify diagrams as much as possible, the internal circuit arrangements are omitted, and the gates are represented by corresponding distinctively-shaped symbols which conform to accepted standards. These are shown in Table A.1 opposite, together with their *truth tables* which indicate all possible combinations of input and output logic states in terms of the binary digits (bits) 0 and 1.

Variations in the symbol shapes adopted will be found in some literature, but those shown here are adopted in the majority of manuals relating to aircraft automatic flight control systems.

Logic gates and truth tables.

OR

A	B	C
0	0	0
0	1	1
1	0	1
1	1	1

NAND

A	B	C
0	0	1
0	1	1
1	0	1
1	1	0

NOR

A	B	C
0	0	1
0	1	0
1	0	0
1	1	0

INVERTER

A	B
0	1

A	B
1	0

NEGATED OR ($\bar{A}$)

$\bar{A}$	B	C
1	0	1
1	1	1
0	0	0
0	1	1

NEGATED OR ($\bar{B}$)

A	$\bar{B}$	C
0	1	1
0	0	0
1	1	1
1	0	1

EXCLUSIVE GATES

OR

A	B	C
0	0	0
0	1	1
1	0	1
1	1	0

NOR

A	B	C
0	0	1
0	1	0
1	0	0
1	1	1

NEGATE

AND

OR

WIRED G

Appendix 5: Solutions to Multi-choice Questions

Chapter 1   4(b), 9(b), 14(b), 21(b), 27(c), 28(a)

Chapter 5   1(c), 2(b)

Chapter 6   8(c), 10(b), 14(c)

Chapter 7   1(c)

Chapter 9   7(b), 18(c)

Chapter 10   3(c)

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