1

Flight Control Systems

1.1 Introduction

Flight controls have advanced considerably throughout the years. In the earliest biplanes flown by the pioneers flight control was achieved by warping wings and control surfaces by means of wires attached to the flying controls in the cockpit. Figure 1.1 clearly shows the multiplicity of rigging and control wires on an early monoplane. Such a means of exercising control was clearly rudimentary and usually barely adequate for the task in hand. The use of articulated flight control surfaces followed soon after but the use of wires and pulleys to connect the flight control surfaces to the pilot's controls persisted for many years until advances in aircraft performance rendered the technique inadequate for all but the simplest aircraft.

Figure 1.1 Morane Saulnier Monoplane refuelling before the 1913 Aerial Derby (Courtesy of the Royal Aero Club)

Aircraft Systems: Mechanical, electrical, and avionics subsystems integration. 3rd Edition Ian Moir and Allan Seabridge

When top speeds advanced into the transonic region the need for more complex and more sophisticated methods became obvious. They were needed first for high-speed fighter aircraft and then with larger aircraft when jet propulsion became more widespread. The higher speeds resulted in higher loads on the flight control surfaces which made the aircraft very difficult to fly physically. The Spitfire experienced high control forces and a control reversal which was not initially understood. To overcome the higher loadings, powered surfaces began to be used with hydraulically powered actuators boosting the efforts of the pilot to reduce the physical effort required. This brought another problem: that of 'feel'. By divorcing the pilot from the true effort required to fly the aircraft it became possible to undertake manoeuvres which could overstress the aircraft. Thereafter it was necessary to provide artificial feel so that the pilot was given feedback representative of the demands he was imposing on the aircraft. The need to provide artificial means of trimming the aircraft was required as Mach trim devices were developed.

A further complication of increasing top speeds was aerodynamically related effects. The tendency of many high performance aircraft to experience roll/yaw coupled oscillations – commonly called Dutch roll – led to the introduction of yaw dampers and other auto-stabilisation systems. For a transport aircraft these were required for passenger comfort whereas on military aircraft it became necessary for target tracking and weapon aiming reasons.

The implementation of yaw dampers and auto-stabilisation systems introduced electronics into flight control. Autopilots had used both electrical and air driven means to provide an automatic capability of flying the aircraft, thereby reducing crew workload. The electronics used to perform the control functions comprised analogue sensor and actuator devices which became capable of executing complex control laws and undertaking high integrity control tasks with multiple lanes to guard against equipment failures. The crowning glory of this technology was the Category III autoland system manufactured by Smiths Industries and fitted to the Trident and Belfast aircraft.

The technology advanced to the point where it was possible to remove the mechanical linkage between the pilot and flight control actuators and rely totally on electrical and electronic means to control the aircraft. Early systems were hybrid, using analogue computing with discrete control logic. The Control and Stability Augmentation System (CSAS) fitted to Tornado was an example of this type of system though the Tornado retained some mechanical reversion capability in the event of total system failure. However the rapid development and maturity of digital electronics soon led to digital 'fly-by-wire' systems. These developments placed a considerable demand on the primary flight control actuators which have to be able to accommodate multiple channel inputs and also possess the necessary failure logic to detect and isolate failures (see Figure 1.2).

Most modern fighter aircraft of any sophistication now possess a fly-by-wire system due to the weight savings and considerable improvements in handling characteristics which may be achieved. Indeed many such aircraft are totally unstable and would not be able to fly otherwise. In recent years this technology

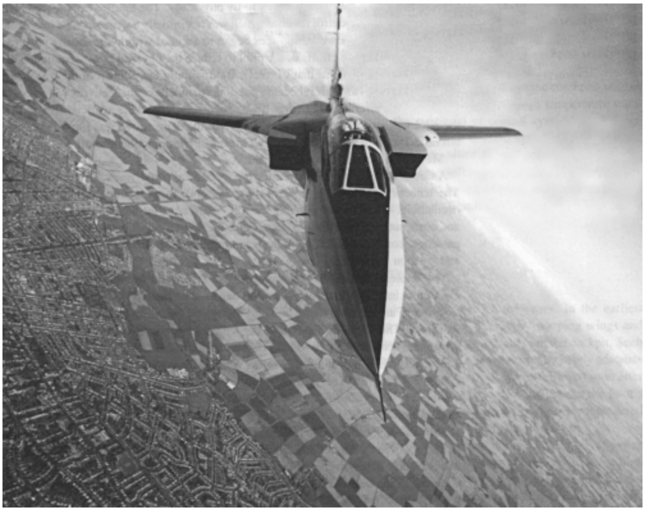

Figure 1.2 Tornado ADV (F 3) Prototype (Courtesy of BAE Systems)

has been applied to civil transports: initially with the relaxed stability system fitted to the Airbus A320 family and A330/A340. The Boeing 777 airliner also has a digital fly-by-wire system, the first Boeing aircraft to do so.

1.2 Principles of Flight Control

All aircraft are governed by the same basic principles of flight control, whether the vehicle is the most sophisticated high-performance fighter or the simplest model aircraft.

The motion of an aircraft is defined in relation to translational motion and rotational motion around a fixed set of defined axes. Translational motion is that by which a vehicle travels from one point to another in space. For an orthodox aircraft the direction in which translational motion occurs is in the direction in which the aircraft is flying, which is also the direction in which it is pointing. The rotational motion relates to the motion of the aircraft around three defined axes: pitch, roll and yaw. See Figure 1.3.

This figure shows the direction of the aircraft velocity in relation to the pitch, roll and yaw axes. For most of the flight an aircraft will be flying straight

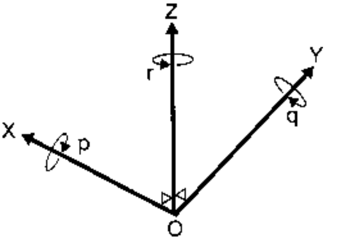

Figure 1.3 Definition of flight control axes

and level and the velocity vector will be parallel with the surface of the earth and proceeding upon a heading that the pilot has chosen. If the pilot wishes to climb, the flight control system is required to rotate the aircraft around the pitch axis (Ox) in a nose-up sense to achieve a climb angle. Upon reaching the new desired altitude the aircraft will be rotated in a nose-down sense until the aircraft is once again straight and level.

In most fixed wing aircraft, if the pilot wishes to alter the aircraft heading then he will need to execute a turn to align the aircraft with the new heading. During a turn the aircraft wings are rotated around the roll axis (Oy) until a certain bank is attained. In a properly balanced turn the angle of roll when maintained will result in an accompanying change of heading while the roll angle (often called the bank angle) is maintained. This change in heading is actually a rotation around the yaw axis (Oz). The difference between the climb (or descent) and the turn is that the climb only involves rotation around one axis whereas the turn involves simultaneous coordination of two axes. In a properly coordinated turn, a component of aircraft lift acts in the direction of the turn, thereby reducing the vertical component of lift. If nothing were done to correct this situation, the aircraft would begin to descend; therefore in a prolonged turning manoeuvre the pilot has to raise the nose to compensate for this loss of lift. At certain times during flight the pilot may in fact be rotating the aircraft around all three axes, for example during a climbing or descending turning manoeuvre.

The aircraft flight control system enables the pilot to exercise control over the aircraft during all portions of flight. The system provides control surfaces that allow the aircraft to manoeuvre in pitch, roll and yaw. The system has also to be designed so that it provides stable control for all parts of the aircraft flight envelope; this requires a thorough understanding of the aerodynamics and dynamic motion of the aircraft. As will be seen, additional control surfaces are required for the specific purposes of controlling the high lift devices required during approach and landing phases of flight. The flight control system has to give the pilot considerable physical assistance to overcome the enormous aerodynamic forces on the flight control surfaces. This in turn leads to the need to provide the aircraft controls with 'artificial feel' so that he does not inadvertently overstress the aircraft. These 'feel' systems need to provide the pilot with progressive and well-harmonised controls that make the aircraft safe and pleasant to handle. A typical term that is commonly used today to describe this requirement is 'carefree handling'. Many aircraft embody automatic flight control systems to ease the burden of flying the aircraft and to reduce pilot workload.

1.3 Flight Control Surfaces

The requirements for flight control surfaces vary greatly between one aircraft and another, depending upon the role, range and agility needs of the vehicle. These varying requirements may best be summarised by giving examples of two differing types of aircraft: an agile fighter aircraft and a typical modern airliner.

The (Experimental Aircraft Programme) EAP aircraft is shown in Figure 1.4 and represented the state of the art fighter aircraft as defined by European Manufacturers at the beginning of the 1990s. The EAP was the forerunner to the European Fighter Aircraft (EFA) or Eurofighter Typhoon developed by the four nation consortium comprising Alenia (Italy), British Aerospace (UK), CASA (Spain) and DASA (Germany).

1.4 Primary Flight Control

Primary flight control in pitch, roll and yaw is provided by the control surfaces described below.

Pitch control is provided by the moving canard surfaces, or foreplanes, as they are sometimes called, located either side of the cockpit. These surfaces provide the very powerful pitch control authority required by an agile high performance aircraft. The position of the canards in relation to the wings renders the aircraft unstable. Without the benefit of an active computer-driven control system the aircraft would be uncontrollable and would crash in a matter of seconds. While this may appear to be a fairly drastic implementation, the benefits in terms of improved manoeuvrability enjoyed by the pilot outweigh the engineering required to provide the computer-controlled or 'active' flight control system.

Roll control is provided by the differential motion of the foreplanes, augmented to a degree by the flaperons. In order to roll to the right, the left foreplane leading edge is raised relative to the airflow generating greater lift than before. Conversely, the right foreplane moves downwards by a corresponding amount relative to the airflow thereby reducing the lift generated. The resulting differential forces cause the aircraft to roll rapidly to the right. To some extent roll control is also provided by differential action of the wing trailing edge flaperons (sometimes called elevons). However, most of the roll control is provided by the foreplanes.

Yaw control is provided by the single rudder section. For high performance aircraft yaw control is generally less important than for conventional aircraft due to the high levels of excess power. There are nevertheless certain parts of the flight envelope where control of yaw (or sideslip) is vital to prevent roll–yaw divergence.

1.5 Secondary Flight Control

High lift control is provided by a combination of flaperons and leading edge slats. The flaperons may be lowered during the landing approach to increase the wing camber and improve the aerodynamic characteristics of the wing. The leading edge slats are typically extended during combat to further increase wing camber and lift. The control of these high lift devices during combat may occur automatically under the control of an active flight control system. The

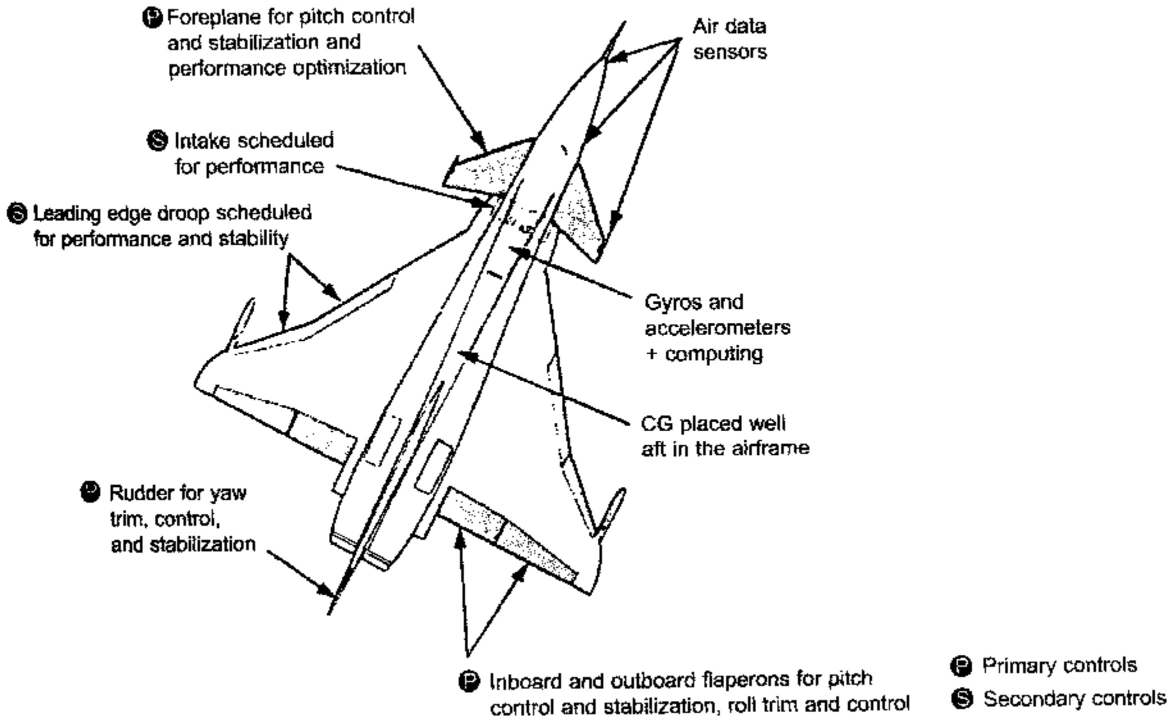

Figure 1.4 Example of flight control surfaces – EAP (Courtesy of BAE Systems)

penalty for using these high lift devices is increased drag, but the high levels of thrust generated by a fighter aircraft usually minimises this drawback.

The Eurofighter Typhoon has airbrakes located on the upper rear fuselage. They extend to an angle of around 50 degrees, thereby quickly increasing the aircraft drag. The airbrakes are deployed when the pilot needs to reduce speed quickly in the air; they are also often extended during the landing run to enhance the aerodynamic brake effect and reduce wheel brake wear.

1.6 Commercial Aircraft

1.6.1 Primary Flight Control

An example of flight control surfaces of a typical commercial airliner is shown in Figure 1.5. Although the example is for the Airbus Industrie A320 it holds good for similar airliners produced by Boeing. The controls used by this type of aircraft are described below.

Pitch control is exercised by four elevators located on the trailing edge of the tailplane (or horizontal stabiliser in US parlance). Each elevator section is independently powered by a dedicated flight control actuator, powered in turn by one of several aircraft hydraulic power systems. This arrangement is dictated by the high integrity requirements placed upon flight control systems. The entire tailplane section itself is powered by two or more actuators in order to trim the aircraft in pitch. In a dire emergency this facility could be used to control the aircraft, but the rates of movement and associated authority are insufficient for normal control purposes.

Roll control is provided by two aileron sections located on the outboard third of the trailing edge of each wing. Each aileron section is powered by a dedicated actuator powered in turn from one of the aircraft hydraulic systems. At low airspeeds the roll control provided by the ailerons is augmented by differential use of the wing spoilers mounted on the upper surface of the wing. During a right turn the spoilers on the inside wing of the turn, that is the right wing, will be extended. This reduces the lift of the right wing causing it to drop, hence enhancing the desired roll demand.

Yaw control is provided by three independent rudder sections located on the trailing edge of the fin (or vertical stabiliser). These sections are powered in a similar fashion to the elevator and ailerons. On a civil airliner these controls are associated with the aircraft yaw dampers. These damp out unpleasant ‘Dutch roll’ oscillations which can occur during flight and which can be extremely uncomfortable for the passengers, particularly those seated at the rear of the aircraft.

1.6.2 Secondary Flight Control

Flap control is effected by several flap sections located on the inboard two-thirds of the wing trailing edges. Deployment of the flaps during take-off or landing extends the flap sections rearwards and downwards to increase wing

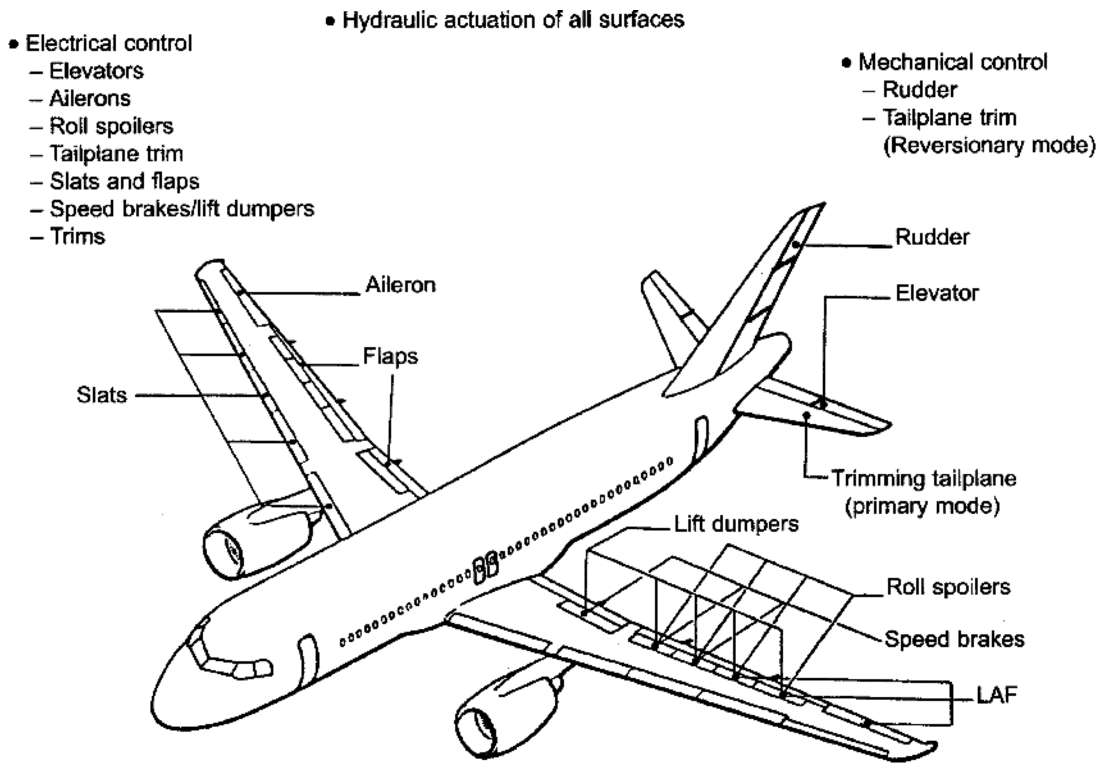

Figure 1.5 Example of flight control surfaces – commercial airliner (A320)
(Courtesy of Airbus (UK))

area and camber, thereby greatly increasing lift for a given speed. The number of flap sections may vary from type to type; typically for this size of aircraft there would be about five per wing, giving a total of ten in all.

Slat control is provided by several leading edge slats, which extend forwards and outwards from the wing leading edge. In a similar fashion to the flaps described above, this has the effect of increasing wing area and camber and therefore overall lift. A typical aircraft may have five slat sections per wing, giving a total of ten in all.

Speed-brakes are deployed when all of the over-wing spoilers are extended together which has the effect of reducing lift as well as increasing drag. The effect is similar to the use of air-brakes in the fighter, increasing drag so that the pilot may adjust his airspeed rapidly; most airbrakes are located on rear fuselage upper or lower sections and may have a pitch moment associated with their deployment. In most cases compensation for this pitch moment would be automatically applied within the flight control system.

While there are many identical features between the fighter and commercial airliner examples given above, there are also many key differences. The greatest difference relates to the size of the control surfaces in relation to the overall size of the vehicle. The fighter control surfaces are much greater than the corresponding control surfaces on an airliner. This reflects its prime requirements of manoeuvrability and high performance at virtually any cost. The commercial airliner has much more modest control requirements; it spends a far greater proportion of flying time in the cruise mode so fuel economy rather than ultimate performance is prime target. Passenger comfort and safety are strong drivers that do not apply to the same degree for a military aircraft.

1.7 Flight Control Linkage Systems

The pilot's manual inputs to the flight controls are made by moving the cockpit control column or rudder pedals in accordance with the universal convention:

- Pitch control is exercised by moving the control column fore and aft; pushing the column forward causes the aircraft to pitch down, and pulling the column aft results in a pitch up
- Roll control is achieved by moving the control column from side to side or rotating the control yoke; pushing the stick to the right drops the right wing and vice versa
- Yaw is controlled by the rudder pedals; pushing the left pedal will yaw the aircraft to the left while pushing the right pedal will have the reverse effect

There are presently two main methods of connecting the pilot's controls to the rest of the flight control system. These are:

- Push-pull control rod systems
- Cable and pulley systems

An example of each of these types will be described and used as a means of introducing some of the major components which are essential for the flight control function. A typical high lift control system for the actuation of slats and flaps will also be explained as this introduces differing control and actuation requirements.

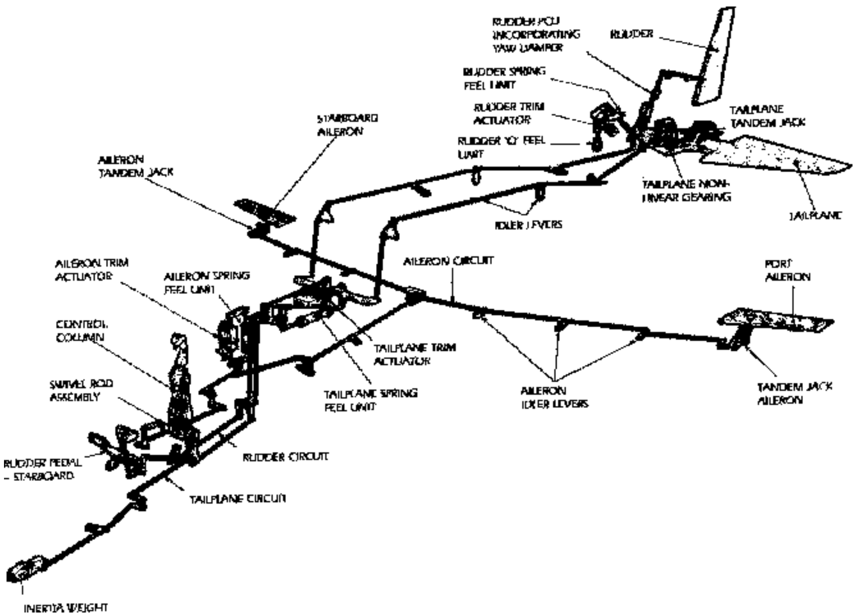

Figure 1.6 Hawk 200 push-pull control rod system (Courtesy of BAE Systems)

1.7.1 Push-Pull Control Rod System

The example chosen for the push-pull control rod system is the relatively simple yet high performance BAE Hawk 200 aircraft. Figure 1.6 shows a simplified three-dimensional schematic of the Hawk 200 flight control which is typical of the technique widely used for combat aircraft. This example is taken from British Aerospace publicity information relating to the Hawk 200 see reference [1]. The system splits logically into pitch–yaw (tailplane and rudder) and roll (aileron) control runs respectively.

The pitch control input is fed from the left hand or starboard side (looking forward) of the control column to a bell-crank lever behind the cockpit. This connects in turn via a near vertical control rod to another bell-crank lever which returns the control input to the horizontal. Bell-crank levers are used to alter the direction of the control runs as they are routed through a densely packed aircraft. The horizontal control rod runs parallel to a tailplane trim actuator/tailplane spring feel unit parallel combination. The output from these units is fed upwards into the aircraft spine before once again being translated by another bell-crank lever. The control run passes down the left side of the fuselage to the rear of the aircraft via several idler levers before entering a nonlinear gearing mechanism leading to the tandem jack tailplane power control unit (PCU). The idler levers are simple lever mechanisms which help to support the control run at convenient points in the airframe. The hydraulically powered PCU drives the tailplane in response to the pilot inputs and the aircraft manoeuvres accordingly.

The yaw input from the rudder pedals is fed to a bell-crank lever using the same pivot points as the pitch control run and runs vertically to another bell-crank which translates the yaw control rod to run alongside the tailplane trim/feel units. A further two bell-cranks place the control linkage running down the right-hand side of the rear fuselage via a set of idler levers to the aircraft empennage. At this point the control linkage accommodates inputs from the rudder trim actuator, spring feel unit and 'Q' feel unit.The resulting control demand is fed to the rudder hydraulically powered PCU which in turn drives the rudder to the desired position. In this case the PCU has a yawdamper incorporated which damps out undesirable 'Dutch roll' oscillations.

The roll demand is fed via a swivel rod assembly from the right hand of port side (looking forward) of the control column and runs via a pair of bell-crank levers to a location behind the cockpit. At this point a linkage connects the aileron trim actuator and the aileron spring feel unit. The control rod runs aft via a further bell-crank lever and an idler lever to the centre fuselage. A further bell-crank lever splits the aileron demand to the left and right wings. The wing control runs are fed outboard by means of a series of idler levers to points in the outboard section of the wings adjacent to the ailerons. Further bell-cranks feed the left and right aileron demands into the tandem jacks and therefore provide the necessary aileron control surface actuation.

Although a simple example, this illustrates some of the considerations which need to be borne in mind when designing a flight control system. The interconnecting linkage needs to be strong, rigid and well supported; otherwise fuselage flexing could introduce 'nuisance' or unwanted control demands into the system. A further point is that there is no easy way or route through the airframe; therefore an extensive system of bell-cranks and idler levers is required to support the control rods. This example has also introduced some of the major components which are required to enable a flight control system to work while providing safe and pleasant handling characteristics to the pilot. These are:

- Trim actuators in tailplane (pitch), rudder (yaw) and aileron (roll) control systems
- Spring feel units in tailplane (pitch), rudder (yaw) and aileron (roll) control systems
- 'Q' feel unit in the rudder (yaw) control system
- Power control units (PCUs) for tailplane, rudder and aileron actuation

1.7.2 Cable and Pulley System

The cable and pulley system is widely used for commercial aircraft; sometimes used in conjunction with push-pull control rods. It is not the intention to attempt to describe a complete aircraft system routing in this chapter. Specific examples will be outlined which make specific points in relation to the larger aircraft (see Figure 1.7).

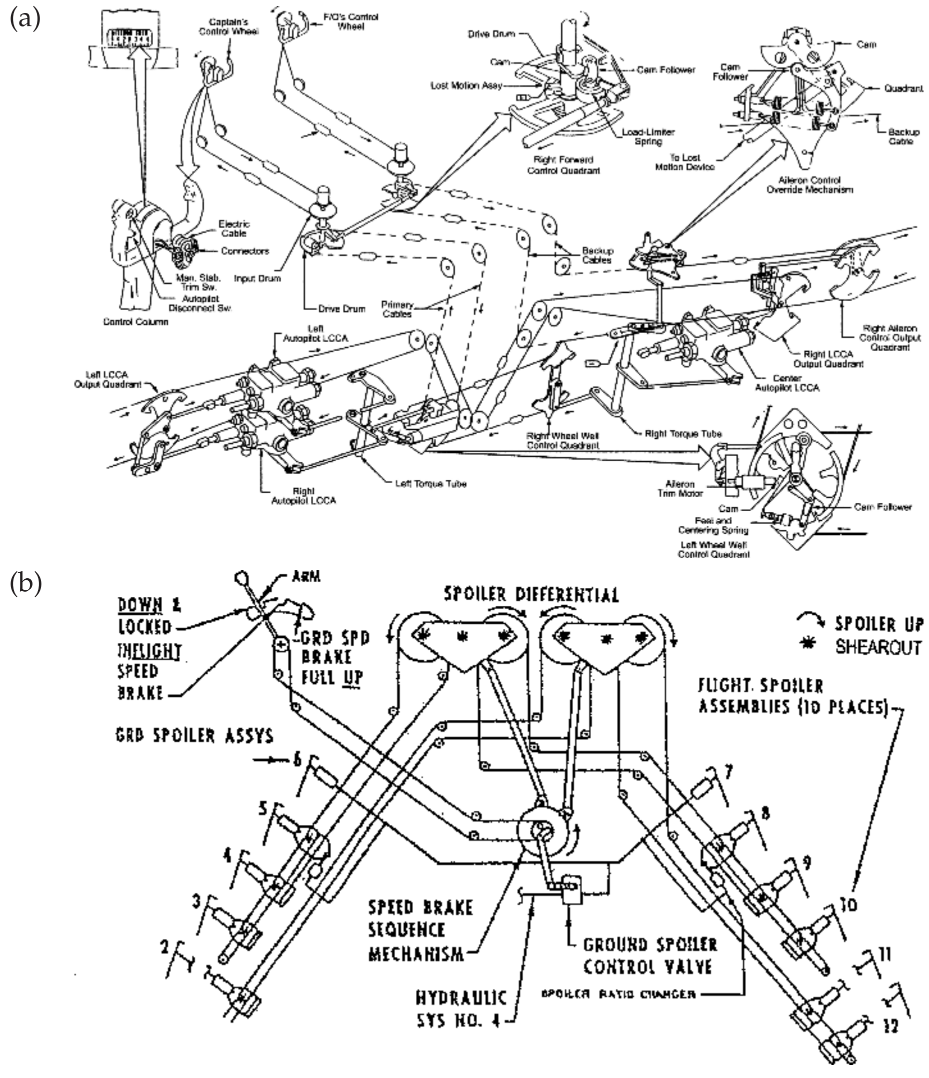

Figure 1.7 Examples of wire and pulley aileron control system (Courtesy of Boeing)

Figure 1.7a shows a typical aileron control system. Manual control inputs are routed via cables and a set of pulleys from both captain's and first officer's control wheels to a consolidation area in the centre section of the aircraft. At this point aileron and spoiler runs are split both left/right and into separate aileron/spoiler control runs. Both control column/control wheels are synchronised. A breakout device is included which operates at a predetermined force in the event that one of the cable runs fails or becomes jammed.

Control cable runs are fed through the aircraft by a series of pulleys, idler pulleys, quadrants and control linkages in a similar fashion to the push-pull rod system already described. Tensiometer/lost motion devices situated throughout the control system ensure that cable tensions are correctly maintained and lost motion eliminated. Differing sized pulleys and pivot/lever arrangements allow for the necessary gearing changes throughout the control runs. Figure 1.7a also shows a typical arrangement for control signalling in the wing. Figure 1.7b shows a typical arrangement for interconnecting wing spoiler and speedbrake controls. Trim units, feel units and PCUs are connected at strategic points throughout the control runs as for the push-pull rod system.

1.8 High Lift Control Systems

The example chosen to illustrate flap control is the system used on the BAE 146 aircraft. This aircraft does not utilise leading edge slats. Instead the aircraft relies upon single section Fowler flaps which extend across 78% of the inner wing trailing edge. Each flap is supported in tracks and driven by recirculating ballscrews at two locations on each wing. The ballscrews are driven by transmission shafts which run along the rear wing spar. The shafting is driven by two hydraulic motors which drive into a differential gearbox such that the failure of one motor does not inhibit the drive capability of the other. See Figure 1.8 for a diagram of the BAE 146 flap operating system.

As well as the flap drive motors and flap actuation, the system includes a flap position selector switch and an electronic control unit. The electronic control unit comprises: dual identical microprocessor based position control channels; two position control analogue safety channels; a single microprocessor based safety channel for monitoring mechanical failures. For an excellent system description refer to the technical paper on the subject prepared by Dowty Rotol/TI Group reference [2].

The slat system or leading edge flap example chosen is that used for the Boeing 747-400. Figure 1.9 depicts the left wing leading edge slat systems. There is a total of 28 flaps, 14 on each wing. These flaps are further divided into groups A and B. Group A flaps are those six sections outside the outboard engines; group B flaps include the five sections between inboard and outboard engines and the three sections inside the inboard engines. The inboard ones are Kreuger flaps which are flat in the extended position, the remainder are of variable camber which provide an aerodynamically shaped surface when extended. The flaps are powered by power drive units (PDUs); six of these drive the group A flaps and two the group B flaps. The motive power is pneumatic with electrical backup. Gearboxes reduce and transfer motion from the PDUs to rotary actuators which operate the drive linkages for each leading edge flap section. Angular position is extensively monitored throughout the system by rotary variable differential transformers (RVDTs).

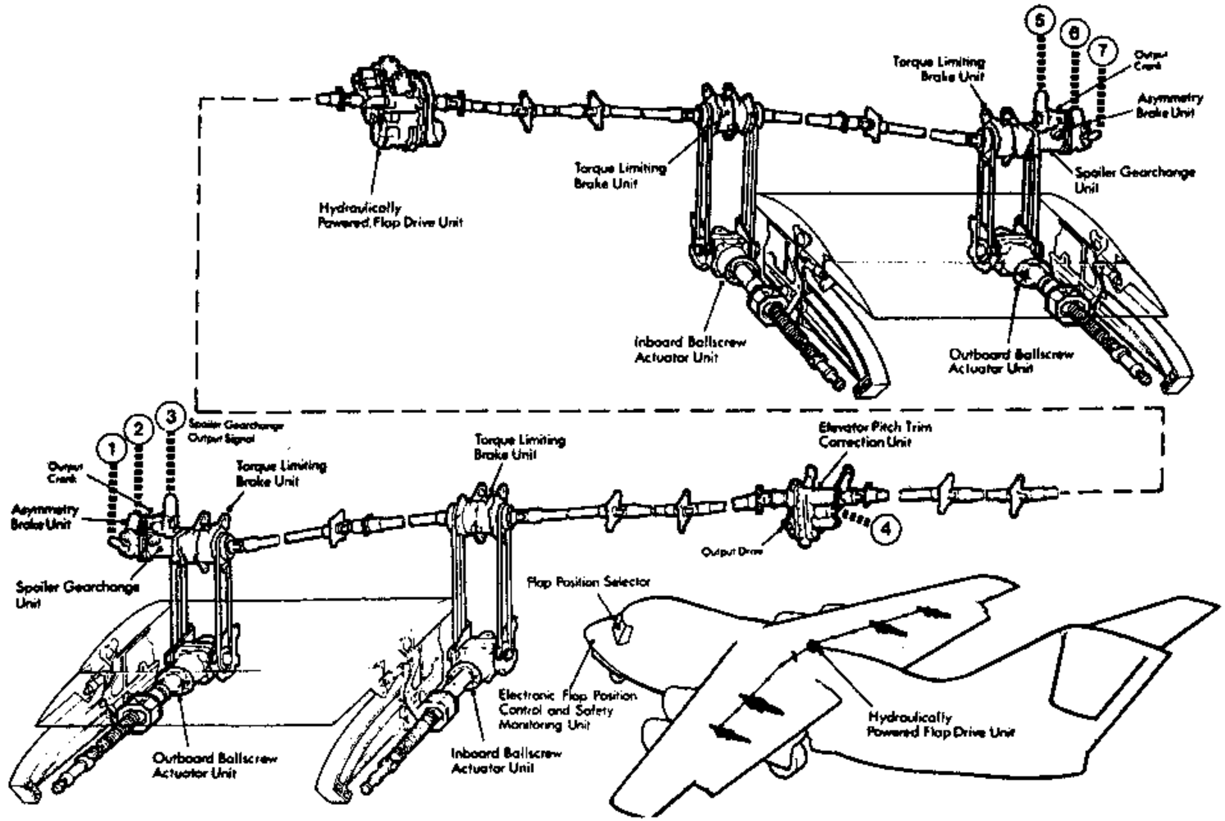

Figure 1.8 BAE 146 flap operating system (Courtesy of Smiths Group – now GE Aviation)

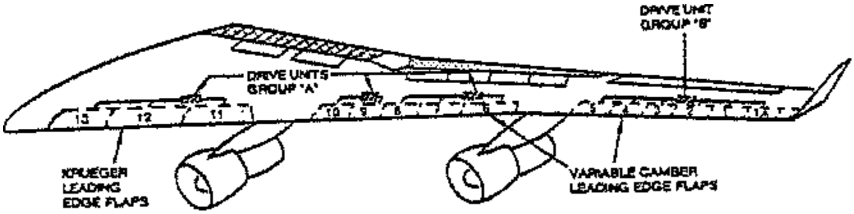

Figure 1.9 Boeing 747-400 leading edge flap system (Courtesy of Boeing)

1.9 Trim and Feel

The rod and pulley example for the BAE Hawk 200 aircraft showed the interconnection between the pilot's control columns and rudder bars and the hydraulically powered actuators which one would expect. However the diagram also revealed a surprising number of units associated with aircraft trim and feel. These additional units are essential in providing consistent handling characteristics for the aircraft in all configurations throughout the flight envelope.

1.9.1 Trim

The need for trim actuation may be explained by recourse to a simple explanation of the aerodynamic forces which act upon the aircraft in flight. Figure 1.10 shows a simplified diagram of the pitch forces which act upon a stable aircraft trimmed for level flight.

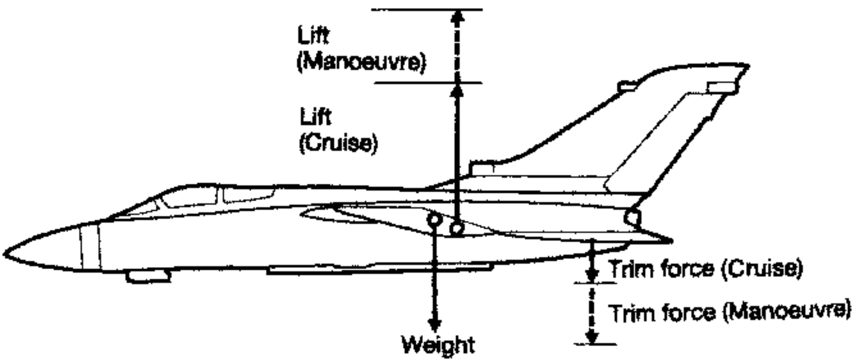

Figure 1.10 Pitch forces acting in level flight

The aircraft weight usually represented by the symbol W, acts downwards at the aircraft centre-of-gravity or CG. As the aircraft is stable the CG is ahead of the centre of pressure where the lift force acts (often denoted by the symbol L) and all aerodynamic perturbations should be naturally damped. The distance between the CG and the centre of pressure is a measure of how stable and also how manoeuvrable the aircraft is in pitch. The closer the CG and centre of pressure, the less stable and more manoeuvrable the aircraft. The converse is true when the CG and centre of pressure are further apart.

Examining the forces acting about the aircraft CG it can be seen that there is a counter-clockwise moment exerted by a large lift force acting quite close to the pivot point. If the aircraft is not to pitch nose-down this should be counterbalanced by a clockwise force provided by the tailplane. This will be a relatively small force acting with a large moment. If the relative positions of the aircraft CG and centre of pressure were to remain constant throughout all conditions of flight then the pilot could set up the trim and no further control inputs would be required.

In practice the CG positions may vary due to changes in the aircraft fuel load and the stores or cargo and passengers the aircraft may be carrying. Variations in the position of the aircraft CG position are allowed within carefully prescribed limits. These limits are called the forward and aft CG limits and they determine how nose heavy or tail heavy the aircraft may become and still be capable of safe and controllable flight. The aerodynamic centre of pressure similarly does not remain in a constant position as the aircraft flight conditions vary. If the centre of pressure moves aft then the downward force required of the tailplane will increase and the tailplane angle of incidence will need to be increased. This requires a movement of the pitch control run equivalent to a small nose-up pitch demand. It is inconvenient for the pilot constantly to apply the necessary backward pressure on the control column, so a pitch actuator is provided to alter the pitch control run position and effectively apply this nose-up bias. Forward movement of the centre of pressure relative to the CG would require a corresponding nose-down bias to be applied. These nose-up and nose-down biases are in fact called nose-up and nose-down trim respectively.

Pitch trim changes may occur for a variety of reasons: increase in engine power, change in airspeed, alteration of the fuel disposition, deployment of flaps or airbrakes and so on. The desired trim demands may be easily input to the flight control system by the pilot. In the case of the Hawk the pilot has a four-way trim button located on the stick top; this allows fore and aft (pitch) and lateral (roll) trim demands to be applied without moving his hand from the control column.

The example described above outlines the operation of the pitch trim system as part of overall pitch control. Roll or aileron trim is accomplished in a very similar way to pitch trim by applying trim biases to the aileron control run by means of an aileron trim actuator. Yaw or rudder trim is introduced by the separate trim actuator provided; in the Hawk this is located in the rear of the aircraft. The three trim systems therefore allow the pilot to offload variations in load forces on the aircraft controls as the conditions of flight vary.

1.9.2 Feel

The provision of artificial 'feel' became necessary when aircraft performance increased to the point where it was no longer physically possible for the pilot to apply the high forces needed to move the flight control surfaces. Initially with servo-boosting systems, and later with powered flying controls, it became necessary to provide powered assistance to attain the high control forces required. This was accentuated as the aircraft wing thickness to chord ratio became much smaller for performance reasons and the hinge moment available was correspondingly reduced. However, a drawback with a pure power assisted system is that the pilot may not be aware of the stresses being imposed on the aircraft. Furthermore, a uniform feel from the control system is not a pleasant characteristic; pilots are not alone in this regard; we are all used to handling machinery where the response and feel are sensibly related. The two types of feel commonly used in aircraft flight control systems are spring feel and 'Q' feel.

Typically the goal is to provide a fairly constant 'Stick force per g' over the full flight envelope. In this regard, the feel system is further complicated with variable geometry aircraft such as the Tornado since aircraft response in pitch and roll varies dramatically with wing sweep. The feel system must therefore take into account both Q and wing sweep.

Spring feel, as the name suggests, is achieved by loading the movement of the flight control run against a spring of a predetermined stiffness. Therefore when the aircraft controls are moved, the pilot encounters an increasing force proportional to the spring stiffness. According to the physical laws spring stiffness is a constant and therefore spring feel is linear unless the physical geometry of the control runs impose any nonlinearities. In the Hawk 200, spring feel units are provided in the tailplane, aileron and rudder control runs. The disadvantage of spring feel units is that they only impose feel proportional to control demand and take no account of the pertaining flight conditions.

'Q' feel is a little more complicated and is more directly related to the aerodynamics and precise flight conditions that apply at the time of the control demand. As the aircraft speed increases the aerodynamic load increases in a mathematical relationship proportional to the air density and the square of velocity. The air density is relatively unimportant; the squared velocity term has a much greater effect, particularly at high speed. Therefore it is necessary to take account of this aerodynamic equation; that is the purpose of 'Q' feel. A 'Q' feel unit receives air data information from the aircraft pitot-static system. In fact the signal applied is the difference between pitot and static pressure, (known as Pt-Ps) and this signal is used to modulate the control mechanism within the 'Q' feel unit and operate a hydraulic load jack which is connected into the flight control run.

In this way the pilot is given feel which is directly related to the aircraft speed and which will greatly increase with increasing airspeed. It is usual to use 'Q' feel in the tailplane or rudder control runs; where this method of feel is used depends upon the aircraft aerodynamics and the desired handling or safety features. The disadvantage of ‘Q’ feel is that it is more complex and only becomes of real use at high speed. Figure 1.11 is a photograph of a ‘Q’ feel unit supplied by Dowty for the BAE Harrier GR5 and McDonnell Douglas AV-8B aircraft. This unit is fitted with an electrical solenoid so that the active part of the system may be disconnected if required. This unit is designed to operate with an aircraft 20.7 MN/sq m (3000 psi) hydraulic system pressure.

Figure 1.11 ‘Q’ feel unit for GR5/AV8B (Courtesy of Smiths Group – now GE Aviation)

The rudder control run on Hawk 200 shown in Figure 1.6 uses both spring and ‘Q’ feel. It is likely that these two methods have been designed to complement each other. The spring feel will dominate at low speed and for high deflection control demands. The ‘Q’ feel will dominate at high speeds and low control deflections.

1.10 Flight Control Actuation

The key element in the flight control system, increasingly so with the advent of fly-by-wire and active control units, is the power actuation. Actuation has always been important to the ability of the flight control system to attain its specified performance. The development of analogue and digital multiple control lane technology has put the actuation central to performance and integrity issues. Addressing actuation in ascending order of complexity leads to the following categories:

- Simple mechanical actuation, hydraulically powered
- Mechanical actuation with simple electromechanical features
- Multiple redundant electromechanical actuation with analogue control inputs and feedback

The examination of these crudely defined categories leads more deeply into systems integration areas where boundaries between mechanical, electronic, systems and software engineering become progressively blurred.

1.10.1 Simple Mechanical/Hydraulic Actuation

Conventional Linear Actuator

The conventional linear actuator used in powered flight controls would be of the type show in Figure 1.12. This type of actuator would usually be powered by one of the aircraft hydraulic systems – in this case the blue channel is shown. In functionally critical applications a dual hydraulic supply from another aircraft hydraulic system may be used. A mechanically operated Servo Valve (SV) directs the hydraulic supply to the appropriate side of the piston ram.

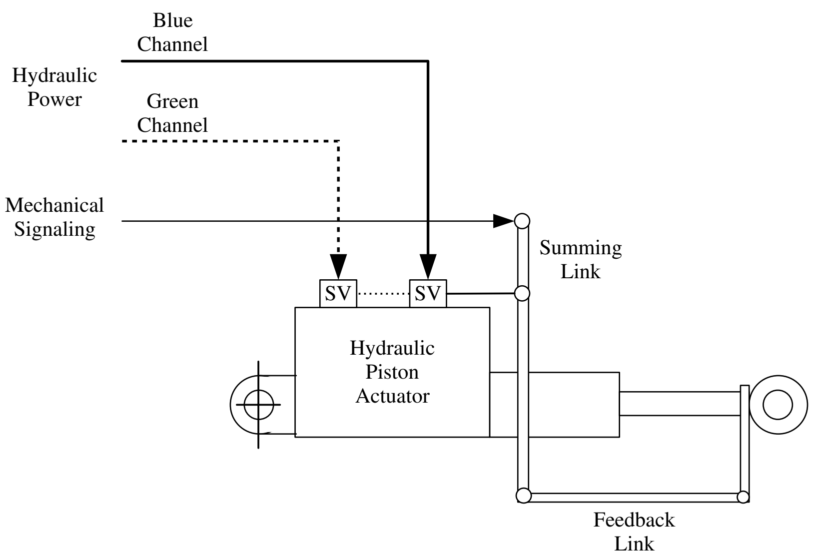

Figure 1.12 Conventional linear actuator

As the pilot feeds a mechanical input to the flight control actuator, the summing link will rotate about the bottom pivot, thus applying an input to the servo valve. Hydraulic fluid will then flow into one side of the ram while exiting the opposite side resulting in movement of the ram in a direction dependent upon the direction of the pilot's command. As the ram moves, the feedback link will rotate the summing link about the upper pivot returning the servo valve input to the null position as the commanded position is achieved.

The attributes of mechanical actuation are straightforward; the system demands a control movement and the actuator satisfies that demand with a power assisted mechanical response. The BAE Hawk 200 is a good example of a system where straightforward mechanical actuation is used for most of the flight control surfaces. For most applications the mechanical actuator is able to accept hydraulic power from two identical/redundant hydraulic systems. The obvious benefit of this arrangement is that full control is retained following loss of fluid or a failure in either hydraulic system. This is important even in a simple system as the loss of one or more actuators and associated control surfaces can severely affect aircraft handling. The actuators themselves have a simple reversion mode following failure, that is to centre automatically under the influence of aerodynamic forces. This reversion mode is called aerodynamic centring and is generally preferred for obvious reasons over a control surface freezing or locking at some intermediate point in its travel. In some systems 'freezing' the flight control system may be an acceptable solution depending upon control authority and reversionary modes that the flight control system possesses.The decision to implement either of these philosophies will be a design decision based upon the system safety analysis.

Mechanical actuation may also be used for spoilers where these are mechanically rather than electrically controlled. In this case the failure mode is aerodynamic closure, that is the airflow forces the control surface to the closed position where it can subsequently have no adverse effect upon aircraft handling. Figure 1.13 illustrates the mechanical spoiler actuator supplied by

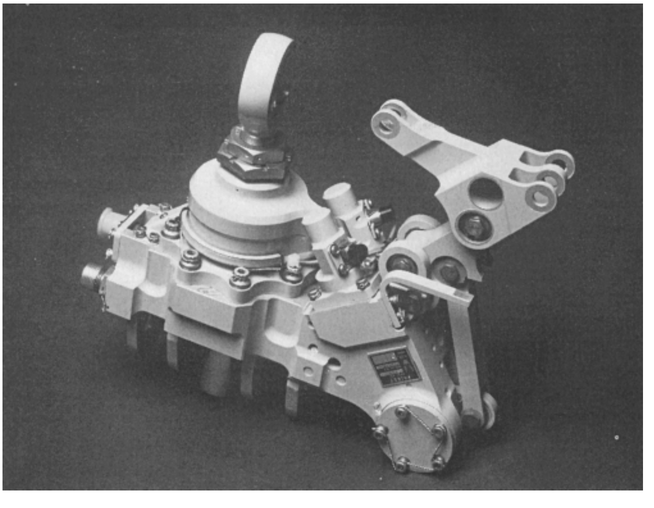

Figure 1.13 BAE 146 spoiler actuator (Courtesy of Claverham/Hamilton Sundstrand)

Claverham for the BAE 146. This unit is simplex in operation. It produces thrust of 59.9 kN (13 460 lb) over a working stroke of 15 mm (0.6 inch). It has a length of 22.4 mm (8.8 inch) and weighs 8.3 kg (18.2 lb). The unit accepts hydraulic pressure at 20.7 MN/sqm (3000 psi).

1.10.2 Mechanical Actuation with Electrical Signalling

The use of mechanical actuation has already been described and is appropriate for a wide range of applications. However the majority of modern aircraft use electrical signalling and hydraulically powered (electro-hydraulic) actuators for a wide range of applications with varying degrees of redundancy. The demands for electro-hydraulic actuators fall into two categories: simple demand signals or autostabilisation inputs.

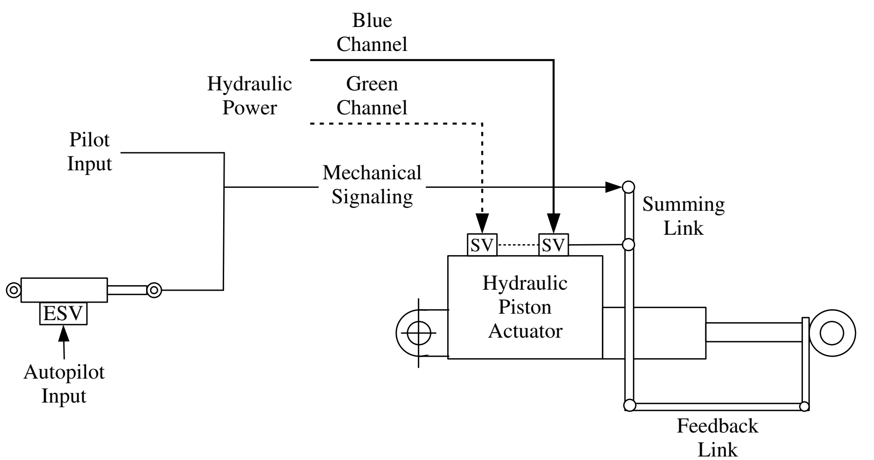

Figure 1.14 Conventional linear actuator with autopilot interface

As aircraft acquired autopilots to reduce pilot work load then it became necessary to couple electrical as well as mechanical inputs to the actuator as shown in Figure 1.14. The manual (pilot) input to the actuator acts as before when the pilot is exercising manual control. When the autopilot is engaged electrical demands from the autopilot computer drive an electrical input which takes precedence over the pilot's demand. The actuator itself operates in an identical fashion as before with the mechanical inputs to the summing link causing the Servo-Valve (SV) to move. When the pilot retrieves control by disengaging the autopilot the normal mechanical link to the pilot through the aircraft control run is restored.

Simple electrical demand signals are inputs from the pilots that are signalled by electrical means. For certain noncritical flight control surfaces it may be easier, cheaper and lighter to utilise an electrical link. An example of this is the airbrake actuator used on the BAE 146; simplex electrical signalling is used and in the case of failure the reversion mode is aerodynamic closure.

In most cases where electrical signalling is used this will at least be duplex in implementation and for fly-by-wire systems signalling is likely to be quadruplex; these more complex actuators will be addressed later. An example of duplex electrical signalling with a simplex hydraulic supply is the spoiler actuators on Tornado. There are four actuators fitted on the aircraft, two per wing, which are used for roll augmentation.

In general, those systems which extensively use simplex electrical signalling do so for autostabilisation. In these systems the electrical demand is a stabilisation signal derived within a computer unit. The simplest form of autostabilisation is the yaw damper which damps out the cyclic cross-coupled oscillations which occur in roll and yaw known as 'Dutch roll'. The Hawk 200 illustrated this implementation. Aircraft which require a stable platform for weapon aiming may have simplex autostabilisation in pitch, roll and yaw; an example of this type of system is the Harrier/AV-8A. A similar system on the Jaguar uses simplex autostabilisation in pitch and roll.

1.10.3 Multiple Redundancy Actuation

Modern flight control systems are increasingly adopting fly-by-wire solutions as the benefits to be realised by using such a system are considerable. These benefits include a reduction in weight, improvement in handling performance and crew/passenger comfort. Concorde was the first aircraft to pioneer these techniques in the civil field using a flight control system jointly developed by GEC (now Finmeccanica) and SFENA.[3] The Tornado, fly-by-wire Jaguar and EAP have extended the use of these techniques; the latter two were development programmes into the regime of the totally unstable aircraft. In the civil field the Airbus A320 and the Boeing 777 introduced modern state-of-the-art systems into service. For obvious reasons, a great deal of care is taken during the definition, specification, design, development and certification of these systems. Multiple redundant architectures for the aircraft hydraulic and electrical systems must be considered as well as multiple redundant lanes or channels of computing and actuation for control purposes. The implications of the redundancy and integrity of the other aircraft systems will be addressed. For the present, attention will be confined to the issues affecting multiple redundant electro-hydraulic actuation.

A simplified block schematic diagram of a multiple redundant electro-hydraulic actuator is shown in Figure 1.15. For reasons of simplicity only one lane or channel is shown; in practice the implementation is likely to be quadruplex, i.e. four identical lanes. The solenoid valve is energised to supply hydraulic power to the actuator, often from two of the aircraft hydraulic systems. Control demands from the flight control computers are fed to the servo valves. The servo valves control the position of the first-stage valves that are mechanically summed before applying demands to the control valves. The control valves modulate the position of the control ram. Linear variable

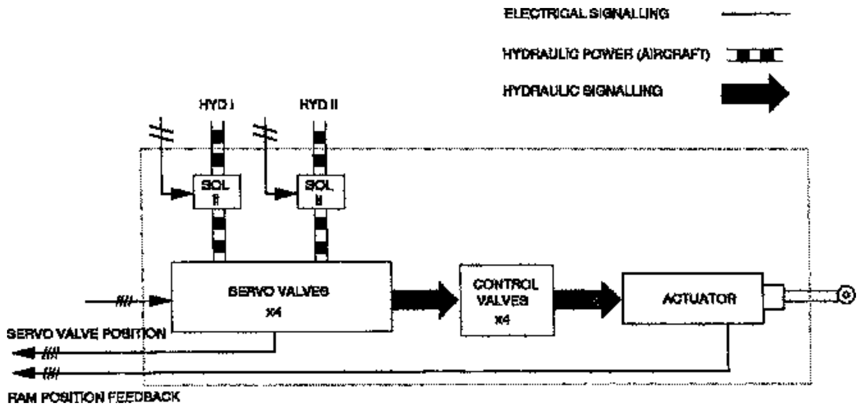

Figure 1.15 Simplified block schematic diagram of a multiple redundant electrically signalled hydraulic actuator

differential transformers (LVDTs) measure the position of the first-stage actuator and output ram positions of each lane and these signals are fed back to the flight control computers, thereby closing the loop. Two examples of this quadruplex actuation system are given below: the Tornado quadruplex taileron and rudder actuators associated with the Control Stability Augmentation System (CSAS) and the EAP flight control system. Both of these systems are outlined at system level in reference [1]. The description given here will be confined to that part of the flight control system directly relevant to the actuator drives.

The Tornado CSAS flight control computation is provided by pitch and lateral computers supplied by GEC (now part of Finmeccanica) and Bodenseewerk (now Thales). The pitch computer predominantly handles pitch control computations and the lateral computer roll and yaw computations though there are interconnections between the two (see Figure 1.16a). There are three computing lanes; computing is analogue in nature and there are a number of voter-monitors within the system to vote out lanes operating outside specification. The combined pitch/roll output to the taileron actuators is consolidated from three lanes to four within the pitch computer so the feed to the taileron actuators is quadruplex. The quadruplex taileron actuator is provided by Fairey Hydraulics (now Hamilton Sundstrand) and is shown in Figure 1.16b. This actuator provides a thrust of 339.3 kN (76 291 lb) over a working stroke of 178 mm. The actuator is 940 mm (37.0 in) long and weighs 51.0 kg and operates with the two aircraft 4000 psi hydraulic systems. The rudder actuator similarly receives a quadruplex rudder demand from the lateral computer, also shown in Figure 1.14b. The rudder actuator is somewhat smaller than the taileron actuator delivering a thrust of 80.1 kN. The CSAS is designed so that following a second critical failure it is possible to revert to a mechanical link for pitch and roll. In these circumstances the rudder is locked in the central position.

The Tornado example given relates to the analogue system that comprises the CSAS. The EAP flight control system (FCS) is a quadruplex digital computing

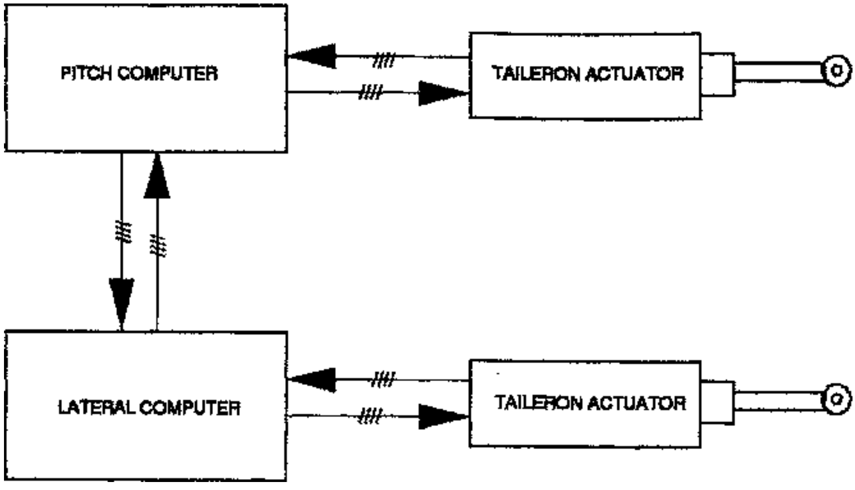

Figure 1.16a Tornado Taileron/Rudder CSAS drive interface

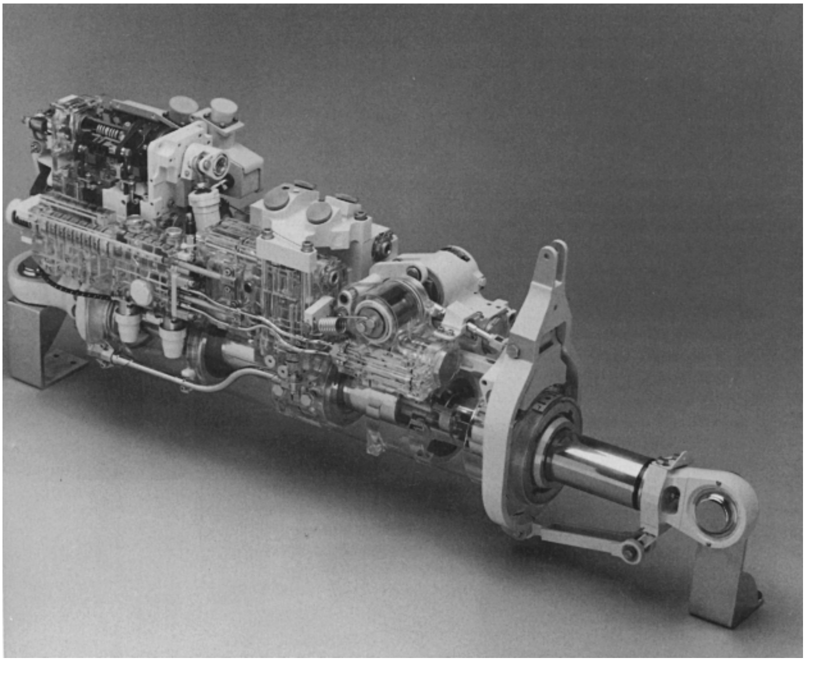

Figure 1.16b Tornado taileron and rudder actuators (Courtesy of Claverham/Hamilton Sundstrand)

system in which control computations are undertaken in all four computing lanes. The system is quadruplex rather than triplex as a much higher level of integrity is required. As has been mentioned earlier the EAP was an unstable aircraft and the FCS has to be able to survive two critical failures. Figure 1.17a shows the relationship between the flight control computers

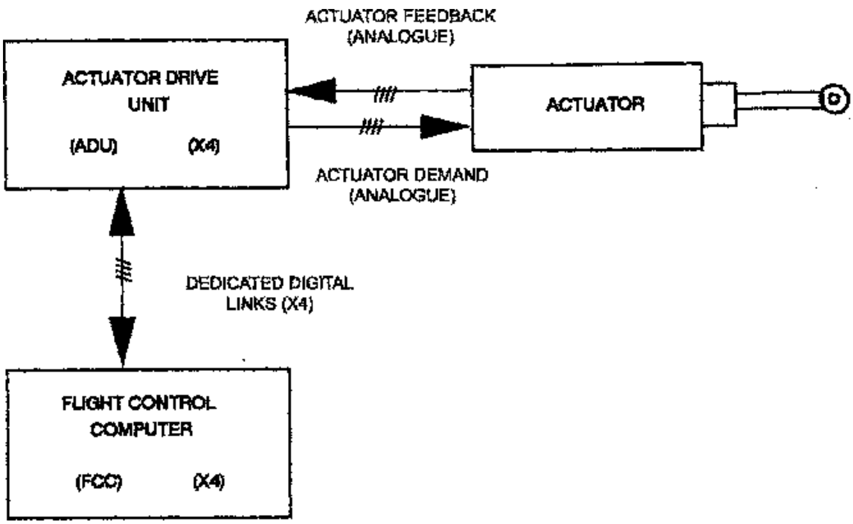

Figure 1.17a EAP actuator drive configuration

(FCCs), Actuator Drive Units (ADUs) and the actuators. The foreplane actuators are fed quadruplex analogue demands from the quadruplex digital FCCs. Demands for the left and right, inboard and outboard flaperons and the rudder are fed in quadruplex analogue form from the four ADUs. The ADUs receive the pitch, roll and yaw demands from the FCCs via dedicated serial digital links and the digital to analogue conversion is carried out within the ADUs. The total complement of actuators supplied by Dowty (now GE Aviation) for the EAP is as follows:

- Quadruplex electrohydraulic foreplane actuators: 2
- Quadruplex electrohydraulic flaperon actuators:
  - outboard flaperons – 100 mm working stroke: 2
  - inboard flaperons – 165 mm working stroke: 2
- Quadruplex electrohydraulic rudder actuators – 100 mm working stroke: 1 (Figure 1.17b.)

All seven actuators are fed from two independent hydraulic systems.

The EAP flight control system represented the forefront of such technology of its time and the aircraft continued to exceed expectations following the first flight in August 1986 until the completion of the programme. Further detail regarding the EAP system and the preceding Jaguar fly-by-wire programme may be found in a number of technical papers which have been given in recent years references [3–8]. Most of these papers are presented from an engineering perspective. The paper by Chris Yeo, Deputy Chief Test Pilot at British Aerospace at the time of the fly-by-wire programme, includes an overview of the aircraft control laws reference [5].

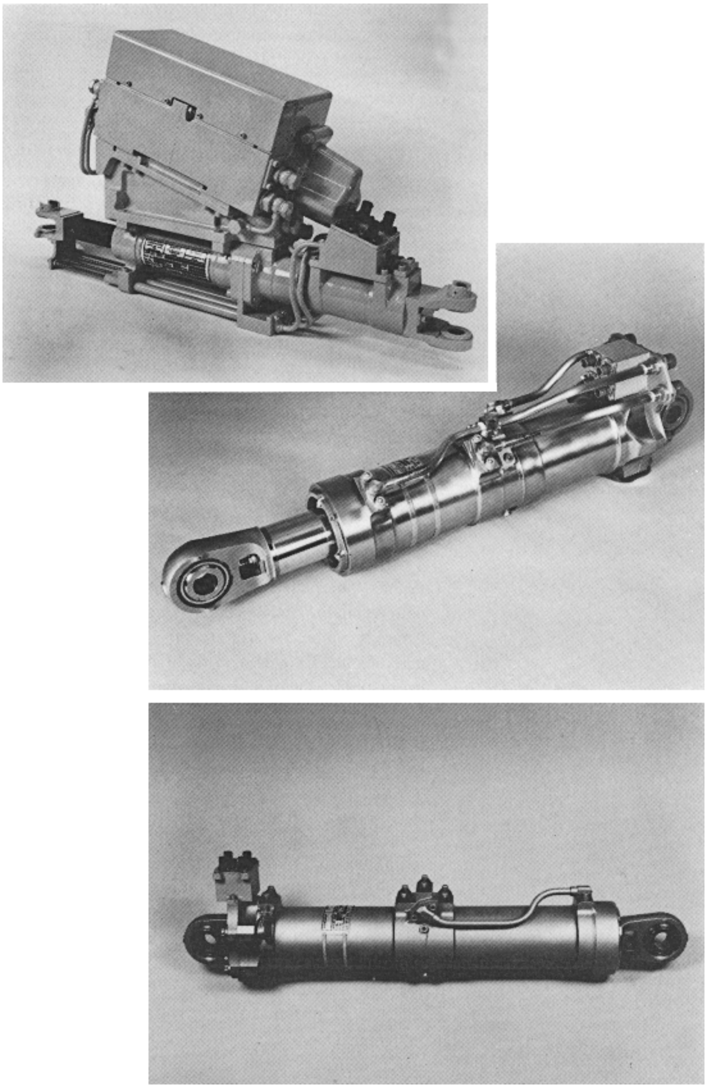

Figure 1.17b EAP foreplane, flaperon and rudder actuators (Courtesy of Smiths Group – now GE Aviation)

1.10.4 Mechanical Screwjack Actuator

The linear actuators described so far are commonly used to power aileron, elevator and rudder control surfaces where a rapid response is required but the aerodynamic loads are reasonably light.There are other applications where a relatively low speed of response may be tolerated but the ability to apply or withstand large loads is paramount. In these situations a mechanical screwjack is used to provide a slow response with a large mechanical advantage. This is employed to drive the Tailplane Horizontal Stabilator or Stabiliser (THS), otherwise known years ago as a 'moving tailplane'. The THS is used to trim an aircraft in pitch as airspeed varies; being a large surface it moves slowly over small angular movements but has to withstand huge loads. The mechanical screwjack shown in Figure 1.18 often has one or two aircraft hydraulic system supplies and a summing link that causes SVs to move in response to the mechanical inputs. In this case the SVs moderate the pressure to hydraulic motor(s) which in turn drive the screwjack through a mechanical gearbox.As before the left-hand portion of the jack is fixed to aircraft structure and movement of the screwjack ram satisfies the pilot's demands, causing the tailplane to move, altering tailplane lift and trimming the aircraft in pitch. As in previous descriptions, movement of the ram causes the feedback link to null the original demand, whereupon the actuator reaches the demanded position.

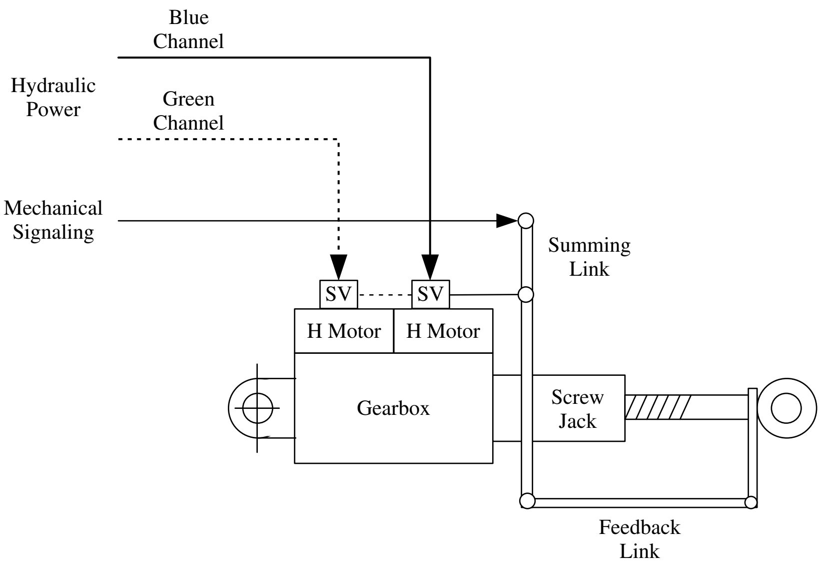

Figure 1.18 Mechanical screwjack actuator

1.10.5 Integrated Actuator Package (IAP)

In the UK, the introduction of powerful new AC electrical systems paved the way for the introduction of electrically powered power flying controls. Four channel AC electrical systems utilised on the Avro Vulcan B2 and Handley Page Victor V-Bombers and the Vickers VC10 transport aircraft utilised flight control actuators powered by the aircraft AC electrical system rather than centralised aircraft hydraulic systems.

Figure 1.19 shows the concept of operation of this form of actuator known as an Integrated Actuator Package (IAP). The operation of demand, summing and feedback linkage is similar to the conventional linear actuator already described. The actuator power or 'muscle' is provided by a three-phase constant speed electrical motor driving a variable displacement hydraulic

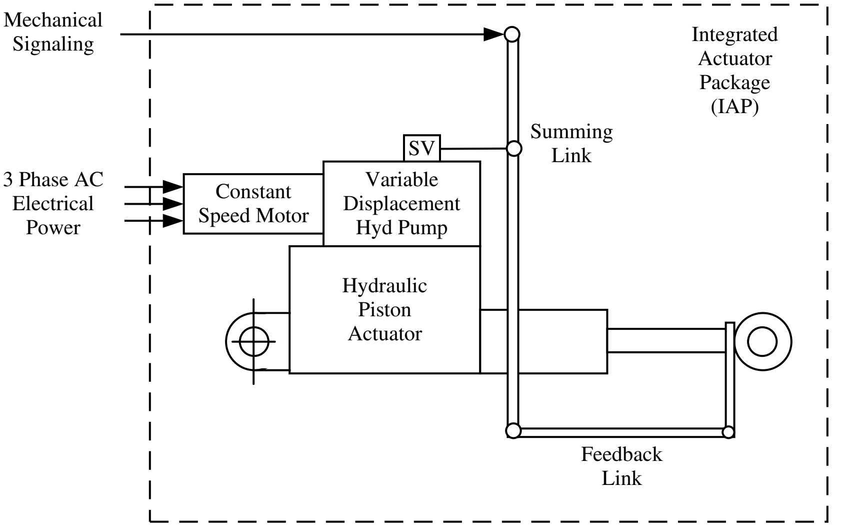

Figure 1.19 Integrated actuator package

pump. The hydraulic pump and associated system provides hydraulic pressure to power the actuator ram.

The variable displacement hydraulic pump is the hydraulic pressure source for the actuator. A bi-directional displacement mechanism which is controlled via a servo valve determines the pumps flow and hence actuator velocity. As with the linear actuator, a feedback mechanism nulls off the input to the servo valve as the desired output position is achieved.

Therefore when the actuator is in steady state, the pump displacement is set to the null position but the pump continues to rotate at a constant speed imposing a significant 'windage' power loss which is a significant disadvantage with this design. The more modern integrated actuator designs, specifically the Electro-Hydrostatic Actuator (described later) eliminates this problem.

Figure 1.20 depicts an overview of a typical IAP used on the Vickers VC-10 flight control system. A total of 11 such units were used in the VC-10 system to powr each of the following flight control surfaces:

- Ailerons: 4 sections
- Elevators: 4 sections
- Rudder: 3 sections

The power consumption of each of the IAPs is in the region of 2.75 kVA and are still flying today in the Royal Air Force's VC-10 Tanker fleet. The units are powered by a constant frequency, split-parallel, 115 VAC three-phase electrical system.

The Avro Vulcan B-2 also used IAP to power the primary flight control surfaces. Being a large delta aircraft this system had an unusual configuration comprising eight elevons powered by IAPs located on the trailing edge of the delta wing plus two on the aircraft rudder. The elevons provided a combined elevator and aileron function to control the aircraft in pitch and roll. Figure 1.21

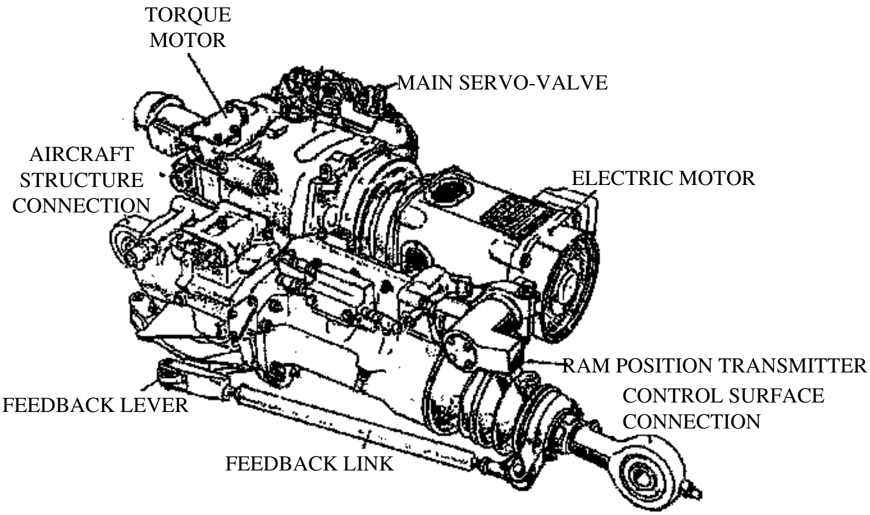

Figure 1.20 Integrated actuator package (VC-10)

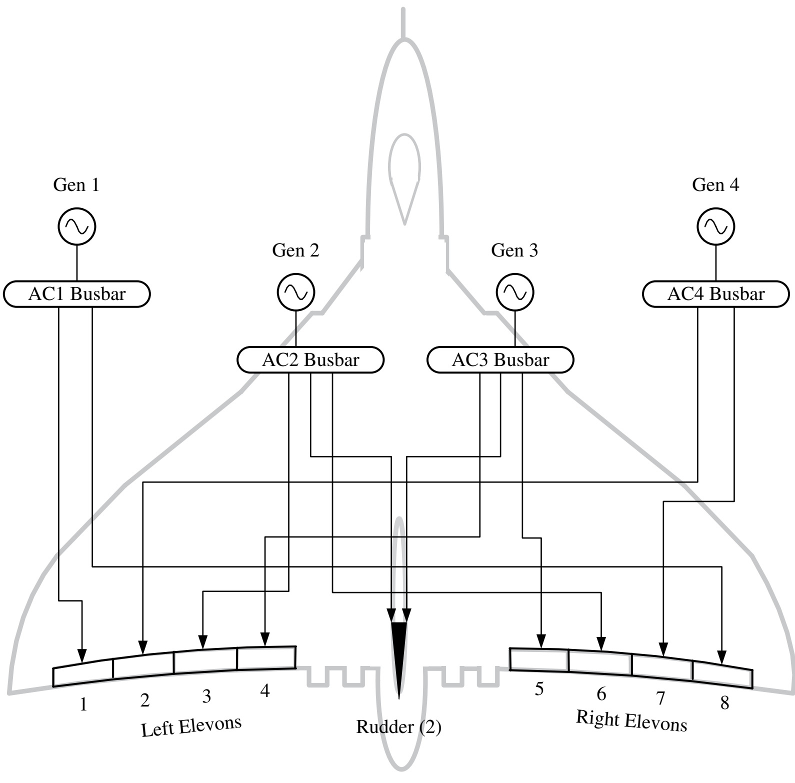

Figure 1.21 Avro Vulcan B-2 FCS architecture using IAPs

illustrates how the total complement of ten power flight control units were powered by the four aircraft AC buses.

1.10.6 Advanced Actuation Implementations

The actuation implementations described so far have all been mechanical or electro-hydraulic in function using servo valves. There are a number of recent developments that may supplant the existing electro-hydraulic actuator. These newer types of actuation are listed below and have found application in aircraft over the past 10–15 years:

- Direct drive actuation
- Fly-by-Wire (FBW) actuation
- Electro-Hydrostatic Actuator (EHA)
- Electro-Mechanical Actuator (EMA)

Direct Drive Actuation

In the electro-hydraulic actuator a servo valve requires a relatively small electrical drive signal, typically in the order of 10–15 mA. The reason such low drive currents are possible is that the control signal is effectively amplified within the hydraulic section of the actuator. In the direct drive actuator the aim is to use an electrical drive with sufficient power to obviate the need for the servo valve/1st stage valve. The main power spool is directly driven by torque motors requiring a higher signal current, hence the term 'direct drive'. Development work relating to the direct drive concept including comparison with Tornado requirements and operation with 8 000psi hydraulic systems has been investigated by Fairey Hydraulics see reference [9]. This paper also addresses the direct digital control of aircraft flight control actuators.

Fly-By-Wire Actuator

The advent of Fly-By-Wire (FBW) flight control systems in civil aircraft commencing with the Airbus A320 introduced the need for a more sophisticated interface between the FCS and actuation. Most first generation FBW aircraft may operate in three distinct modes that may be summarised in general terms as follows:

- *Full FBW Mode*. This mode encompasses the full FBW algorithms and protection and is the normal mode of operation
- *Direct Electrical Link Mode*. This mode will usually provide rudimentary algorithms or possibly only a direct electrical signalling capability in the event that the primary FBW mode is not available
- *Mechanical Reversion Mode*. This provides a crude means of flying the aircraft – probably using a limited number of flight control surface following the failure of FBW and direct electrical link modes

In later implementations such as the Airbus A380 and Boeing 787 no mechanical reversion is provided. The interface with the actuator is frequently achieved by means of an Actuator Control Electronics (ACE) unit that closes the control loop electrically around the actuator rather than mechanical loop closure as hitherto described (see Figure 1.22). The digital FBW or direct link demands from the flight control system are processed by the ACE which supplies an analogue command to the actuator SV. This allows aircraft systems hydraulic power to be supplied to the appropriate side of the ram piston moving the ram to the desired position. In this implementation the ram position is detected by means of a Linear Variable Differential Transducer (LVDT) which feeds the signal back to the ACE where the loop around the actuator is closed. Therefore ACE performs two functions: conversion of digital flight control demands into analogue signals and analogue loop closure around the actuator.

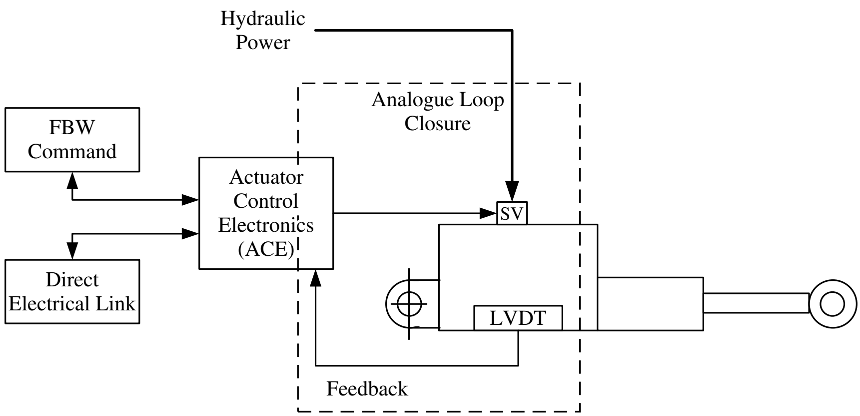

Figure 1.22 Fly-by-wire actuator

Electro-Hydrostatic Actuator (EHA)

The move towards more-electric aircraft has coincided with another form of electrical actuation – the Electro-Hydrostatic Actuator (EHA) which uses state-of-the-art power electronics and control techniques to provide more efficient flight control actuation. The conventional actuation techniques described so far continually pressurise the actuator whether or not there is any demand. In reality for much of the flight, actuator demands are minimal and this represents a wasteful approach as lost energy ultimately results in higher energy offtake from the engine and hence higher fuel consumption.

The EHA seeks to provide a more efficient form of actuation where the actuator only draws significant power when a control demand is sought; for the remainder of the flight the actuator is quiescent (see Figure 1.23). The EHA accomplishes this by using the three-phase AC power to feed power drive electronics which in turn drive a variable speed pump together with a constant displacement hydraulic pump. This constitutes a local hydraulic system for the

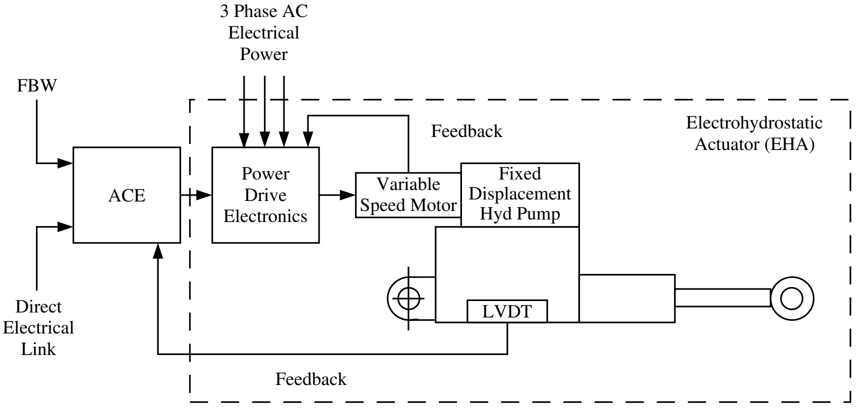

Figure 1.23 Electro-Hydrostatic Actuator (EHA)

actuator in a similar fashion to the IAP; the difference being that when there is no demand the only power drawn is that to maintain the control electronics. When a demand is received from the ACE the power drive electronics is able to react sufficiently rapidly to drive the variable speed motor and hence pressurise the actuator such that the associated control surface may be moved to satisfy the demand. Once the demand has been satisfied then the power electronics resumes its normal dormant state. Consequently power is only drawn from the aircraft buses bars while the actuator is moving, representing a great saving in energy. The ACE closes the control loop around the actuator electrically as previously described.

EHAs are being applied across a range of aircraft and Unmanned Air Vehicle (UAV) developments. The Airbus A380 and Lockheed Martin F-35 Lightning II both use EHAs in the flight control system. For aircraft such as the A380 with a conventional three-phase, 115 VAC electrical system, the actuator uses an in-built matrix converter to convert the aircraft three-phase AC power to 270 VDC to drive a brushless DC motor which in turn drives the fixed displacement pump. The Royal Aeronautical Society Conference, More-Electric Aircraft, 27–28 April 2004, London is an excellent reference for more-electric aircraft and more-electric engine developments where some of these solutions are described.

Aircraft such as the F-35 have an aircraft level 270 VDC electrical system and so the matrix converter may be omitted with further savings in efficiency. Furthermore, electric aircraft/more-electric engine development programmes with civil applications envisage the use of 540 VDC or ±270 VDC systems on the aircraft or engine platform and therefore making similar savings in energy. These developments, including a European Community (EC) funded programme called Power Optimised Aircraft (POA), were described and discussed at the Technologies for Energy Optimised Aircraft Equipment Systems (TEOS) forum in Paris, 28–30 June 2006.

A common feature of all three new actuator concepts outlined above is the use of microprocessors to improve control and performance. The introduction of digital control in the actuator also permits the consideration of direct digital interfacing to digital flight control computers by means of data buses (ARINC 429/ARINC 629/1553B). The direct drive developments described emphasise concentration upon the continued use of aircraft hydraulics as the power source, including the accommodation of system pressures up to 8000 psi. The EMA and EHA developments, on the other hand, lend themselves to a greater use of electrical power deriving from the all-electric aircraft concept, particularly if 270 VDC power is available.

Electro-Mechanical Actuator (EMA)

The electromechanical actuator or EMA replaces the electrical signalling and power actuation of the electro-hydraulic actuator with an electric motor and gearbox assembly applying the motive force to move the ram. EMAs have been used on aircraft for many years for such uses as trim and door actuation; however the power, motive force and response times have been less than that required for flight control actuation. The three main technology advancements that have improved the EMA to the point where it may be viable for flight control applications are: the use of rare earth magnetic materials in 270 VDC motors; high power solid-state switching devices; and microprocessors for lightweight control of the actuator motor [10].

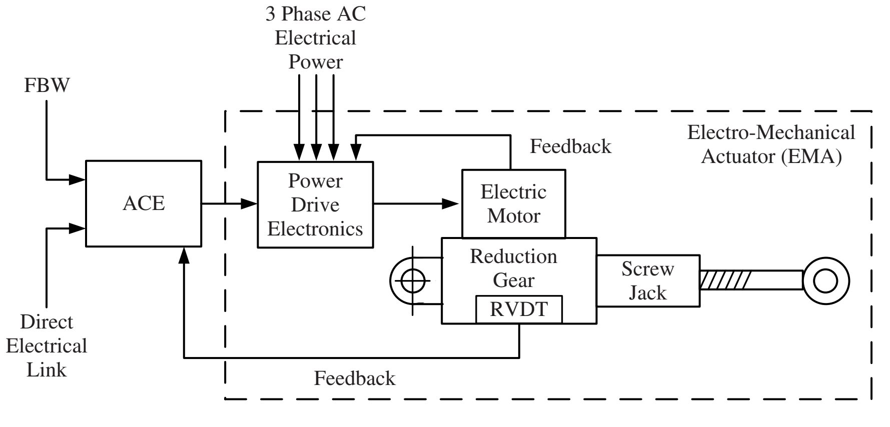

Figure 1.24 Electro-mechanical actuator

As the EHA is the more-electric replacement for linear actuators so the Electro-Mechanical Actuator (EMA) is the more-electric version of the screw-jack actuator as shown in Figure 1.24. The concept of the EMA is identical with the exception that the power drive electronics drives a brushless DC motor operating a reduction gear that applies rotary motion allowing the jack ram to extend or retract to satisfy input demands. EMAs are therefore used to power the THS on civil aircraft and flap and slat drives and also find a use in helicopter flight control systems. A major concern regarding the EMA is the consideration of the actuator jamming case and this has negated their use in primary flight controls on conventional aircraft.

Actuator Matrix

Most of these actuation types are used in civil aircraft today. Table 1.1 lists how the various actuator types may be used for different actuation tasks on a typical civil airliner.

Table 1.1 Typical applications of flight control actuators

Actuator type	*Power source*	*Primary flight control*	*Spoilers*	*Tailplane horizontal stabilator*	*Flaps and slats*
Conventional Linear Actuator	Aircraft Hydraulic Systems: B/Y/G or L/C/R [1]	X	X		
Conventional Screw-jack Actuator	Aircraft Hydraulic or Electrical Systems [2]			X	X
Integrated Actuator Package (IAP)	Aircraft Electrical System (115VAC)	X	X		
Electrically Signalled Hydraulic Actuator	Aircraft Hydraulic Systems	X	X		
Electro-Hydrostatic Actuator (EHA)	Aircraft Electrical System [3] [4]	X	X		
Electro-Mechanical Actuator (EMA)	Aircraft Electrical System [3]			X	X

Notes: (1) B/Y/G = Blue/Green/Yellow or L/C/R = Left/Centre/Right (Boeing)
(2) For THS and Flaps & Slats both hydraulic and electrical supplies are often used for redundancy
(3) 3-phase VAC to 270 VDC matrix converter used in civil
(4) 270 VDC aircraft electrical system used on F-35/JSF

1.11 Civil System Implementations

The flight control and guidance of civil transport aircraft has steadily been getting more sophisticated in recent years. Whereas Concorde was the first civil aircraft to have a fly-by-wire system, Airbus introduced a fly-by-wire system on to the A320 family [11] and a similar system has been carried forward to the A330/340. Boeing's first fly-by-wire system on the Boeing 777 was widely believed to a response to the Airbus technology development. The key differences between the Airbus and Boeing philosophies and implementations are described below.

1.11.1 Top-Level Comparison

The importance and integrity aspects of flight control lead to some form of monitoring function to ensure the safe operation of the control loop. Also for integrity and availability reasons, some form of redundancy is usually required. Figure 1.25 shows a top-level comparison between the Boeing and Airbus FBW implementations.

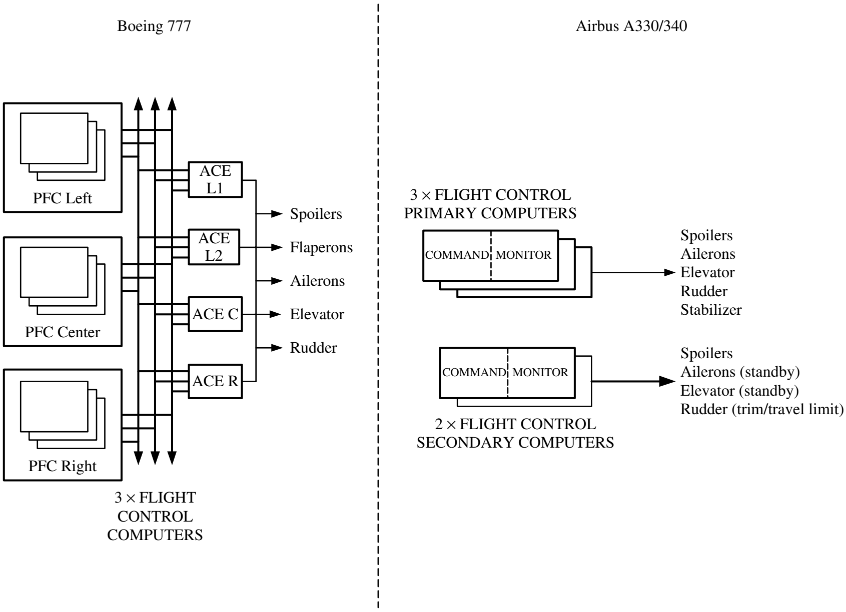

Figure 1.25 Top-level Boeing & Airbus FBW comparison

In the Boeing philosophy, shown in simplified form on the left of Figure 1.25, the system comprises three Primary Flight Computers (PFCs) each of which has three similar lanes with dissimilar hardware but the same software. Each lane has a separate role during an operating period and the roles are cycled after power up. Voting techniques are used to detect discrepancies or disagreements between lanes and the comparison techniques used vary for different types of data. Communication with the four Actuator Control Electronics (ACE) units is by multiple A629 flight control data buses. The ACE units directly drive the flight control actuators. A separate flight control DC system is provided to power the flight control system. The schemes used on the Boeing 777 will be described in more detail later in this Module.

The Airbus approach is shown on the right of Figure 1.25. Five main computers are used: three Flight Control Primary Computers (FCPCs) and two Flight Control Secondary Computers (FCSCs). Each computer comprises command and monitor elements with different software. The primary and secondary computers have different architectures and different hardware. Command outputs from the FCSCs to ailerons, elevators and the rudder are for standby use only. Power sources and signalling lanes are segregated.

1.11.2 Airbus Implementation

The Anglo-French Concorde apart, Airbus was the first aircraft manufacturer in recent years to introduce Fly-By-Wire (FBW) to civil transport aircraft. The original aircraft to utilise FBW was the A320 and the system has been used throughout the A319/320/321 family and more recently on the A330/340. The A320 philosophy will be described and A330/340 system briefly compared.

A320 FBW System

A schematic of the A320 flight control system is shown in Figure 1.26. The flight control surfaces are all hydraulically powered and are tabulated as follows:

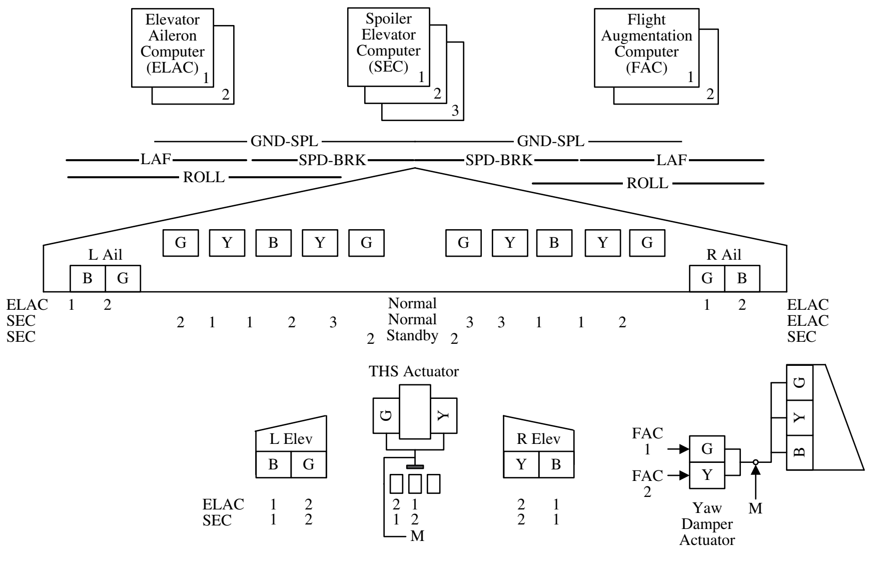

Figure 1.26 A320 flight control system

- *Electrical control:*

Elevators	2
Ailerons	2
Roll spoilers	8
Tailplane trim	1
Slats	10
Flaps	4
Speedbrakes	6
Lift dumpers	10
Trims	

- *Mechanical control*:
  Rudder
  Tailplane trim (reversionary mode)

The aircraft has three independent hydraulic power systems: blue (B), green (G) and yellow (Y). Figure 1.26 shows how these systems respectively power the hydraulic flight control actuators.

A total of seven computers undertake the flight control computation task as follows:

- Two Elevator/Aileron Computers (ELACs). The ELACs control the aileron and elevator actuators according to the notation in the figure
- Three Spoiler/Elevator Computers (SECs). The SECs control all of the spoilers and in addition provide secondary control to the elevator actuators. The various spoiler sections have different functions as shown namely:

  – ground spoiler mode: all spoilers
  – speed brake mode: inboard three spoiler sections
  – load alleviation mode: outboard two spoiler sections (plus ailerons); this function has recently been disabled and is no longer embodied in recent models
  – roll augmentation: outboard four spoiler sections

- Two Flight Augmentation Computers (FACs). These provide a conventional yaw damper function, interfacing only with the yaw damper actuators

The three aircraft hydraulic systems; blue, green and yellow provide hydraulic power to the flight control actuators according to the notation shown on the diagram.

In the very unlikely event of the failure of all computers it is still possible to fly and land the aircraft – this has been demonstrated during certification. In this case the Tailplane Horizontal Actuator (THS) and rudder sections are controlled directly by mechanical trim inputs – shown as M in the diagram – which allow pitch and lateral control of the aircraft to be maintained.

Another noteworthy feature of the Airbus FBW systems is that they do not use the conventional pitch and roll yoke. The pilot’s pitch and roll inputs to the system are by means of a side-stick controller and this has been widely accepted by the international airline community.

In common with contemporary civil aircraft, the A320 is not an unstable aircraft like the EAP system briefly described earlier in this chapter. Instead the aircraft operates with a longitudinal stability margin of around 5% of aerodynamic mean chord or around half what would normally be expected for an aircraft of this type. This is sometimes termed relaxed stability. The A320 family can claim to be the widest application of civil FBW with over 3000 examples delivered.

A330/340 FBW System

The A330/340 FBW system bears many similarities to the A320 heritage as might expected.

The pilot's input to the Flight Control Primary Computers (FCPCs) and Flight Control Secondary Computers (FCSCs) is by means of the side-stick controller. The Flight Management Guidance and Envelope Computers (FMGECs) provide autopilot pitch commands to the FCPC. The normal method of commanding the elevator actuators is via the FCPC although they can be controlled by the FCSC in a standby mode. Three autotrim motors may be engaged via a clutch to drive the mechanical input to the THS.

For the pitch channel, the FCPCs provide primary control and the FCSCs the backup. Pilots' inputs are via the rudder pedals directly or, in the case of rudder trim, via the FCSC to the rudder trim motors.

The yaw damper function resides within the FCPCs rather than the separate Flight Augmentation Computers (FACs) used on the A320 family. Autopilot yaw demands are fed from the FMGECs to the FCPCs.

There is a variable travel limitation unit to limit the travel of the rudder input at various stages of flight. As before, the three hydraulic systems feed the rudder actuators and two yaw damper actuators as annotated on the figure.

Therefore although the implementation and notation of the flight control computers differs between the A320 and A330/340 a common philosophy can be identified between the two families.

The overall flight control system elements for the A330/340 are:

- Three Flight Control Primary Computers (FCPCs); the function of the FCPCs has been described
- Two Flight Control Secondary Computers (FCSCs); similarly, the function of the secondary computers has been explained
- Two Flight Control Data Concentrators (FCDCs); the FCDCs provide data from the primary and secondary flight computers for indication, recording and maintenance purposes
- Two Slat/Flap Control Computers (SFCCs); the FSCCs are each able to control the full-span leading-edge slats and trailing-edge flaps via the hydraulically driven slat and flap motors

Spoiler usage on the A330/340 differs from that on the A320. There is no load alleviation function and there are six pairs of spoilers versus the five pairs on the A320. Also the functions of the various spoiler pairs differ slightly from the A320 implementation. However, overall, the philosophy is the same.

A380 Implementation

The A380 flight control system represents the most advanced system flying today and follows the philosophy used by Airbus over the past 20 years.

Airbus Fly-By-Wire Evolution

The first airbus FBW aircraft was the A320 that was first certified in 1988.Since then the A320 family has expanded to include the A318, A319 and A321; the A330 and A340 aircraft have entered service and the A380 did so in October 2007. In that time the number of flight control actuators has increased with the size of the aircraft as may be seen in Table 1.2.

Table 1.2 Airbus family – roll effectors

Airbus model	*Spoilers per wing*	*Ailerons/actuators per wing*
A320 family	5	1/2
A330/340 family	6	2/4
A380	8	3/6

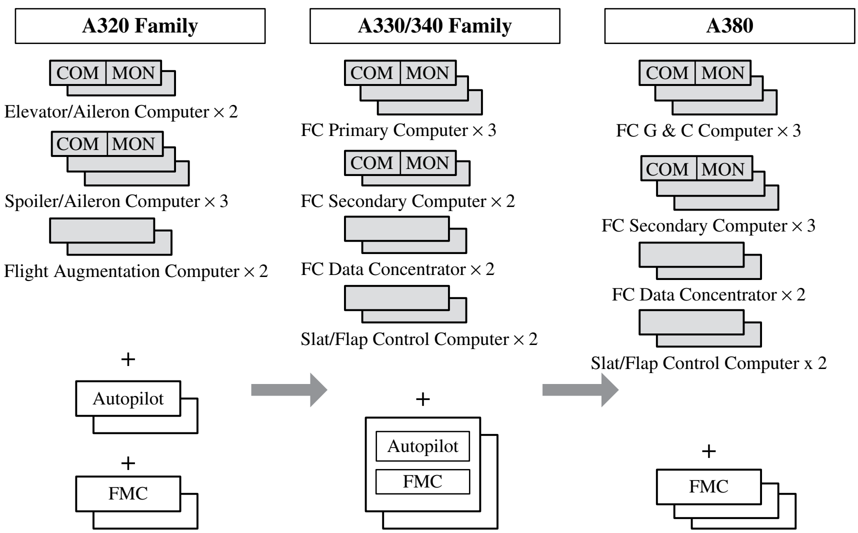

Figure 1.27 Evolution of Airbus fly-by-wire systems

The Airbus Family FBW has evolved historically from the A320 family through the A330/340 family to the latest A380 aircraft. Figure 1.27 clearly illustrates this progression. In this diagram the shaded portion represents the FBW or primary flight control system while the units shown below represent the associated autopilot and flight Management System (FMS) functions.

On the A320 family the autopilot and FMS functions are provided by standalone units. On the A330/340 flight guidance is provided by the Flight Management and Guidance Computers (FMGCs) that embody both autopilot and guidance functions. On the A380 integration has progressed with the autopilot function being subsumed into the FCS with the FMC as stand-alone.

Although the name of the computers has changed from application to application, a clear lineage may be seen with the A380 complement being:

- 3 x Flight Control Primary Computers (FCPCs)
- 3 x Flight Control Secondary Computers (FCSCs)
- 2 x Flight Control Data Concentrators (FCDCs)
- 2 x Flap/Slat Control Computers (FSCCs)

1.12 Fly-By-Wire Control Laws

While it is impossible to generalise, the approach to the application of control laws in a FBW system and the various reversionary modes does have a degree of similarity. The concept of having normal, direct and mechanical links has been outlined earlier in the chapter. The application of normal, alternate and direct control laws and, in the final analysis, mechanical reversion often follows the typical format outlined in Figure 1.28.

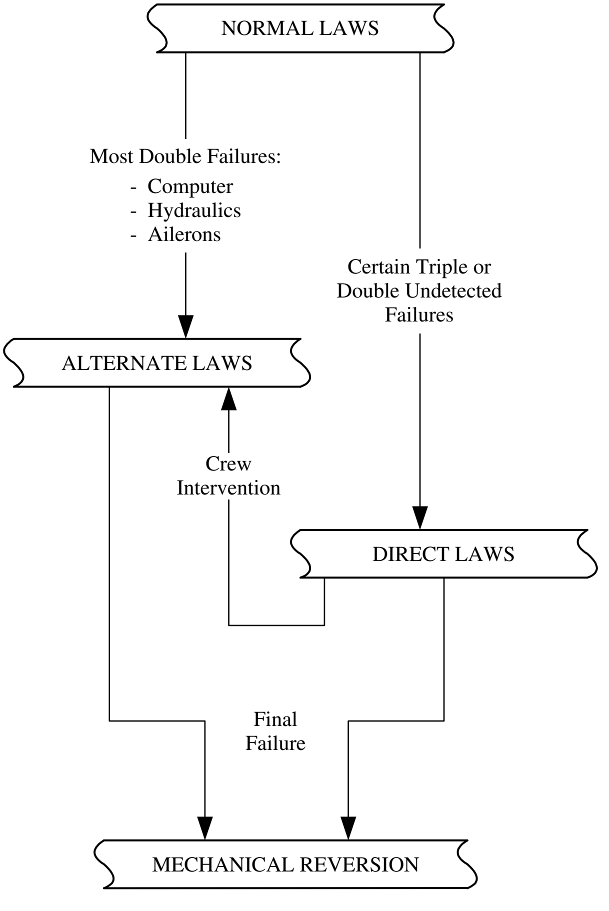

Figure 1.28 Typical interrelationship of FBW control laws

The authority of each of these levels may be summarised as follows:

- *Normal laws*: Provision of basic control laws with the addition of coordination algorithms to enhance the quality of handling and protection to avoid the exceedance of certain attitudes and attitude rates. Double failures in computing, sensors or actuation power channels will cause reversion to the Alternate mode
- *Alternate laws*: Provision of the basic control laws but without many of the additional handling enhancement features and protection offered by the Normal mode. Further failures cause reversion to the Mechanical mode
- *Direct laws*: Direct relationship from control stick to control surface, manual trimming, certain limitations depending upon aircraft CG and flight control system configuration. In certain specific cases crew intervention may enable re-engagement of the Alternate mode. Further failures result in reversion to Mechanical
- *Mechanical reversion*: Rudimentary manual control of the aircraft using pitch trim and rudder pedals to facilitate recovery of the aircraft electrical system or land the aircraft as soon as is practicable

1.13 A380 Flight Control Actuation

The electrical and hydraulic power derived for the A380 flight control actuators is summarised in Figure 1.29.

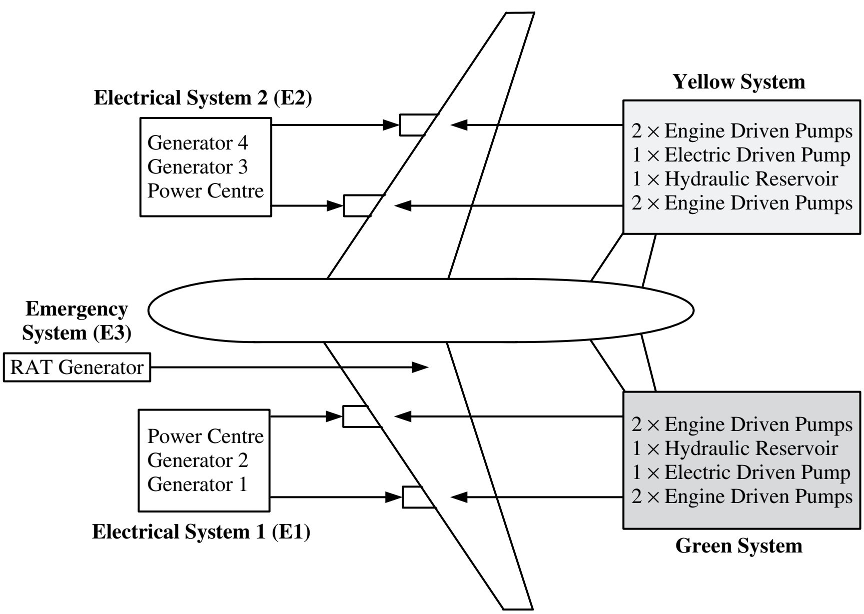

Figure 1.29 A380 hydraulic and electric power generation

Table 1.3 A380 Flight control system actuator matrix

LEFT WING					RIGHT WING			
AILERONS		SPOILERS			SPOILERS		AILERONS	
Inbd	G AC E2	1	Y		Y	1	G AC E2	Inbd
Mid	Y AC E1	2	G		G	2	Y AC E1	Mid
Outbd	Y G	3	Y		Y	3	Y G	Outbd
		4	G		G	4		
		5	Y + AC 2E		Y + AC 2E	5		
		6	G + AC 1E		G + AC 1E	6		
		7	Y		Y	7		
		8	G		G	8		
R ELEVATORS				THS			L ELEVATORS	
Inbd	AC E1 G			G Y AC E2			AC E1 Y	Inbd
Outbd	AC E2 G						AC E2 Y	Outbd
				RUDDER				
				Upper 1 Y + AC E1 2 G + AC E2 Lower 1 G + AC E1 2 Y + AC E3				
KEY:								
	G	Green Hydraulic System					AC E1	AC 1 Essential Side 1
	Y	Yellow Hydraulic System					AC E2	AC 2 Essential Side 2
							AC E3	AC Essential (RAT)

The A380 flight control actuator configuration is shown in Table 1.3. Many of the actuators are powered only by the aircraft green (LH side powered by engines 1 and 2) and yellow (RH side powered by engines 3 and 4) hydraulic systems. However, many are powered by a combination of conventional hydraulic and electro-hydrostatic actuators (see Figure 1.30).

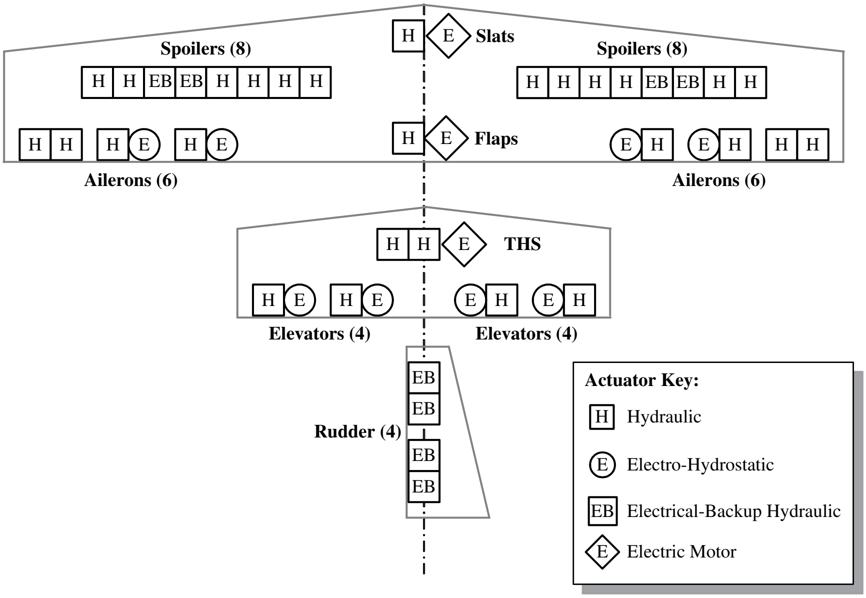

Figure 1.30 A 380 flight control actuation

The use of the various actuation types may be summarised as follows:

- The two outboard aileron surfaces and six spoiler surfaces on each wing are powered by conventional hydraulic actuators – yellow or green system
- The mid and inboard aileron surfaces and the inboard and outboard elevator surfaces are powered by both hydraulic and EHAs, each of which can drive the surface in the event of a failure of the other
- Two spoiler surfaces (five and six on each wing) and both rudder sections are powered by Electrical Backup Hydraulic Actuators (EBHAs) which combine the features of hydraulic actuators and EHAs
- The Tailplane Horizontal Stabilisor (THS) actuator is powered independently from green and yellow channels and from E2

For completeness, the diagram also shows the flap and slat drives. Slats may be powered by green or E1; flaps may be powered from green or yellow channels.

EBHAs receive a hydraulic input from the appropriate channel (green or yellow) and electrical channel (E1 or E2, or exceptionally E3 AC Essential (RAT)). In the case of the rudder, the upper surface is powered by green and yellow, E1 and E2 AC 2; the lower surface is powered by green and yellow, E1 and E3.

EBHAs are capable of two modes of operation:

- *Normal – hydraulic mode*: In the normal mode the actuator receives hydraulic power from the appropriate green or yellow hydraulic system and the SV moderates the supply to the actuator according to the FBW computer demand

- *Backup – EHA mode*: In the backup mode the actuator operates like an EHA. Electrical power is received from the aircraft AC electrical system and the FBW computer feeds demands to the EHA control package. The rotational direction and speed of the electrical motor determine the direction and rate of travel of the actuator ram

A top-level schematic of an EBHA is shown in Figure 1.31. The combination of multiple redundant FBW computing resources (three primary and three secondary flight control computers) and the actuator hydraulic and electrical power architectures described mean that the aircraft is not fitted with a mechanical reversion.

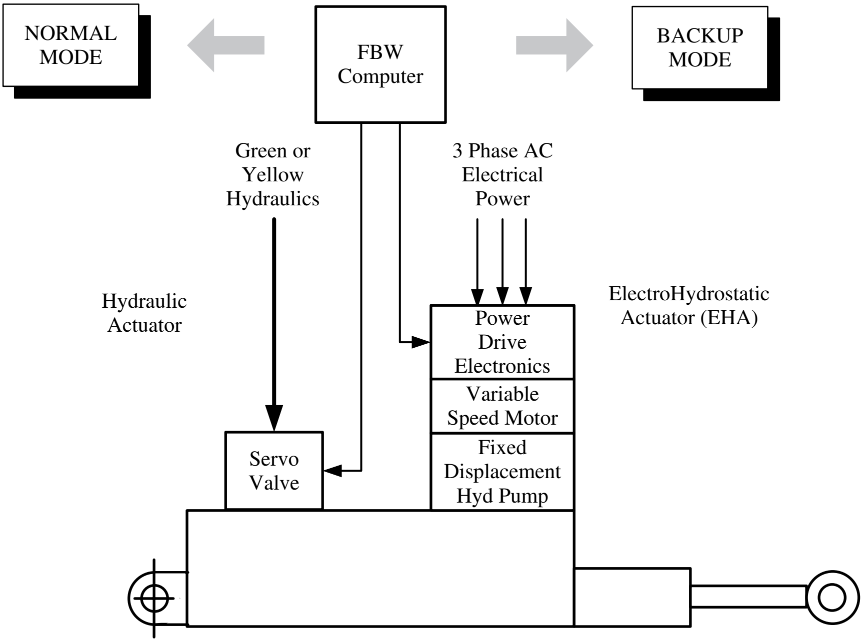

Figure 1.31 A380 EBHA modes of operation

1.14 Boeing 777 Implementation

Boeing ventured into the FBW field with the Boeing 777 partly, it has been said, to counter the technology lead established by Airbus with the A320. Whatever the reason, Boeing have approached the job with precision and professionalism and have developed a solution quite different to the Airbus philosophy. References [12] and [13] give a detailed description of the B777 FBW system.

The B777 PFCS is outlined at a system level in Figure 1.32. The drawing shows the three Primary Flight Control Computers (PFCS), four Actuator Control Electronics (ACEs) and three Autopilot Flight Director Computers (AFDCs) interfacing with the triple redundant A629 flight control buses. The AFDCs have terminals on both the flight control and A629 data buses. Attitude and information is provided by the ADIRU, and SAHRU and air data by the Air Data Modules (ADMs). The three Control and Display Units (CDUs) and the left and right Aircraft Information Management System (AIMS) cabinets provide the flight deck interface. In total there are 76 ARINC 629 couplers on the flight control buses.

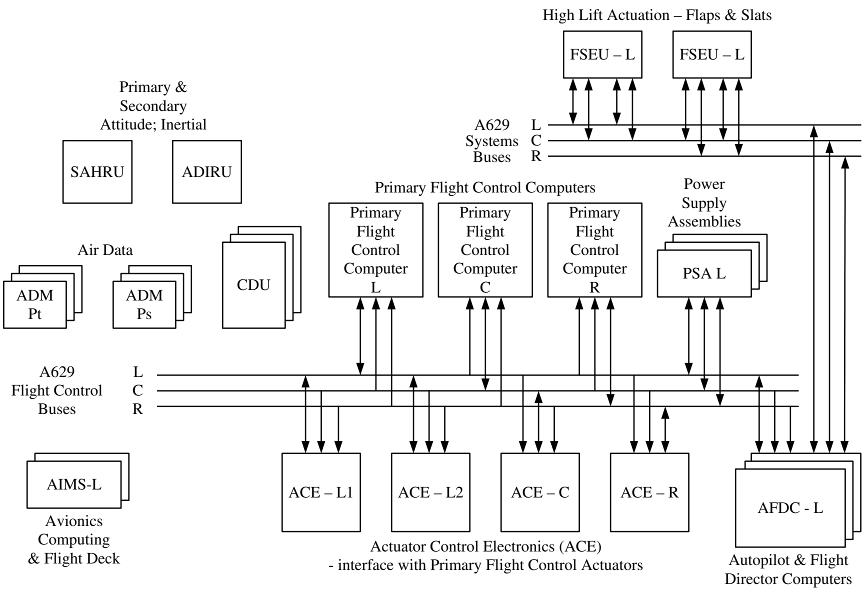

Figure 1.32 B777 Primary Flight Control System (PFCS)

The PFCS system comprises the following control surface actuators and feel actuators:

- Four elevators: left and right inboard and outboard
- Elevator feel: left and right
- Two rudders: upper and lower
- Four ailerons: left and right inboard and outboard
- Four flaperons: left and right inboard and outboard
- Fourteen spoilers: seven left and seven right

The flight control actuators are interfaced to the three A629 flight control data buses by means of four Actuator Control Electronics (ACE) units. These are:

- ACE Left 1
- ACE Left 2
- ACE Centre
- ACE Right

These units interface in turn with the flight control and feel actuators in accordance with the scheme shown in Table 1.4.

Table 1.4 ACE to PCU interface

ACE L1	*ACE L2*	*ACE C*	*ACE R*
ROB Aileron	LOB Aileron	LIB Aileron	RIB Aileron
LOB Aileron	RIB Aileron	ROB Aileron	LIB Aileron
		Upper Rudder	Lower Rudder
LIB Elevator	LOB Elevator	ROB Elevator	RIB Elevator
	L Elevator Feel	R Elevator Feel	
Spoiler 2	Spoiler 5	Spoiler 1	Spoiler 3
Spoiler 13	Spoiler 4	Spoiler 7	Spoiler 6
	Spoiler 11	Spoiler 8	Spoiler 9
	Spoiler 10	Spoiler 14	Spoiler 12

The Actuator Control Electronics (ACE) units contain the digital-to-analogue and analogue-to-digital elements of the system. A simplified schematic for an ACE is shown in Figure 1.33. Each ACE has a single interface with each of the A629 flight control data buses and the unit contains the signal conversion to interface the 'digital' and 'analogue' worlds.

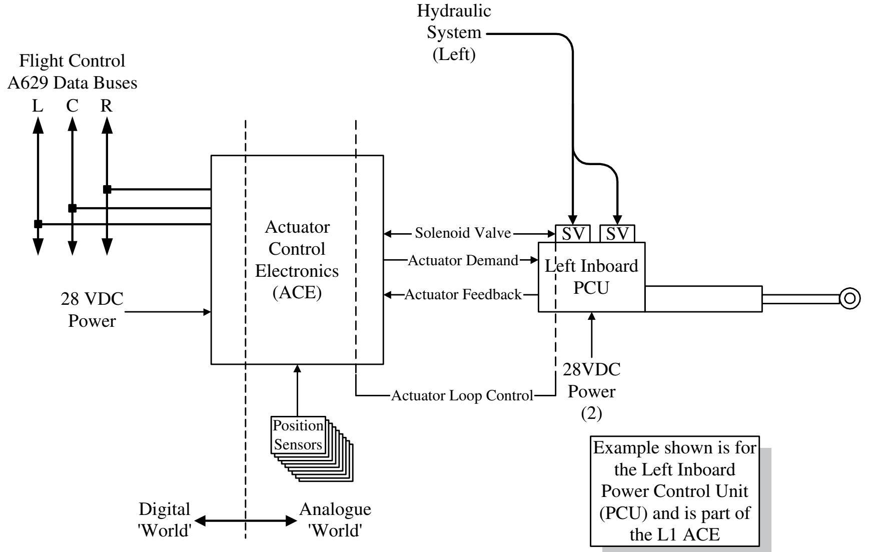

Figure 1.33 Actuator Control Electronics (ACE) Unit

The actuator control loop is shown in the centre-right of the diagram. The actuator demand is signalled to the Power Control Unit (PCU) which moves the actuator ram in accordance with the control demand and feeds back a ram position signal to the ACE, thereby closing the actuator control loop. The ACE also interfaces to the solenoid valve with a command to energise the solenoid valves to allow – in this example – the left hydraulic system to supply the actuator with motive power and at this point the control surface becomes 'live'.

The flight control computations are carried out in the Primary Flight Computers (PFCs) shown in Figure 1.34. The operation of the PFCs has been briefly described earlier in the chapter but will be recounted and amplified in this section.

Each PFC has three A629 interfaces with each of the A629 flight control buses, giving a total of nine data bus connections in all. These data bus interfaces and how they are connected and used form part of the overall Boeing 777 PFCS philosophy. The three active lanes within each PFC are embodied in dissimilar hardware. Each of the three lanes is allocated a different function as follows:

- *PFC command lane*: The command lane is effectively the channel in control. This lane will output the flight control commands on the appropriate A629 bus; e.g. PFC left will output commands on the left A629 bus
- *PFC standby lane*: The standby lane performs the same calculations as the command lane but does not output the commands on to the A629 bus. In effect the standby lane is a 'hot standby', ready to take command in the event that the command lane fails. The standby lane only transmits cross lane and cross-channel data on the A629 data bus
- *PFC monitor lane*: The monitor lane also performs the same calculations as the command lane. The monitor lane operates in this way for both the command lane and the standby lane. Like the standby lane, it only transmits cross lane and cross-channel data on the A629 data bus

Figure 1.34 shows that on the data bus, each PFC will only transmit aircraft control data on the appropriate left, centre or right A629 data bus. Within each PFC the command, standby and monitor lane operation will be in operation as previously described and only the command channel – shown as the upper channel in the figure – will actually transmit command data.

Within this PFC and A629 architecture:

- Cross lane comparisons are conducted via the like bus (in this case the left bus)
- Cross channel comparisons are conducted via the unlike buses (in this case the centre and right buses)

This use of standard A629 databuses to implement the flight control integration and to host the cross lane and cross-channel monitoring is believed to be unique in flight control. There are effectively nine lanes available to conduct the flight control function. In the event that a single lane fails, then only that lane will be

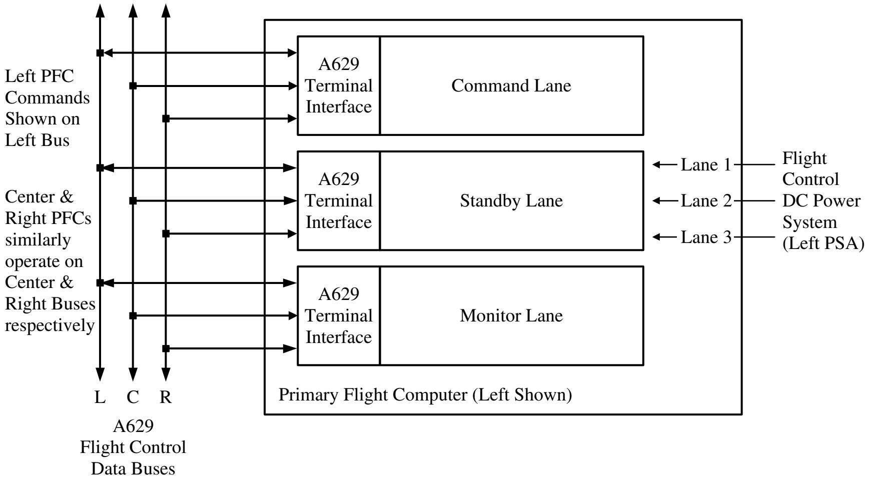

Figure 1.34 B777 Primary Flight Computer (PFC)

shut down. Subsequent loss of a second lane within that channel will cause that channel to shut down, as simplex control is not permitted. The aircraft may be operated indefinitely with one lane out of nine failed and the aircraft may be dispatched with two out of nine lanes failed for ten days. The aircraft may be operated for a day with one PFC channel inoperative.

The autopilot function of the B777 PFCS is undertaken by the three Autopilot Flight Director Computers (AFDCs): left, centre and right. The AFDCs have A629 interfaces on to the respective aircraft systems and flight control data buses. In other words, the left AFDC will interface on to the left A629 buses, the centre AFDC on to the centre buses and so on.

1.15 Interrelationship of Flight Control, Guidance and Flight Management

Figure 1.35 shows a generic example of the main control loops as they apply to aircraft flight control, flight guidance and flight management.

The inner loop provided by the FBW system and the pilot's controls effectively control the attitude of the aircraft.

The middle loop is that affected by the AFDS that controls the aircraft trajectory, that is, where the aircraft flies. Inputs to this loop are by means of the mode and datum selections on the FCU or equivalent control panel.

Finally, the FMS controls where the aircraft flies on the mission; for a civil transport aircraft this is the aircraft route. The MCDU controls the lateral demands of the aircraft by means of a series of waypoints within the route plan and executed by the FMS computer. Improved guidance required of 'free-flight' or DNS/ATM also requires accurate vertical or 3-Dimensional guidance, often with tight timing constraints upon arriving at a way-point or the entry to a terminal area.

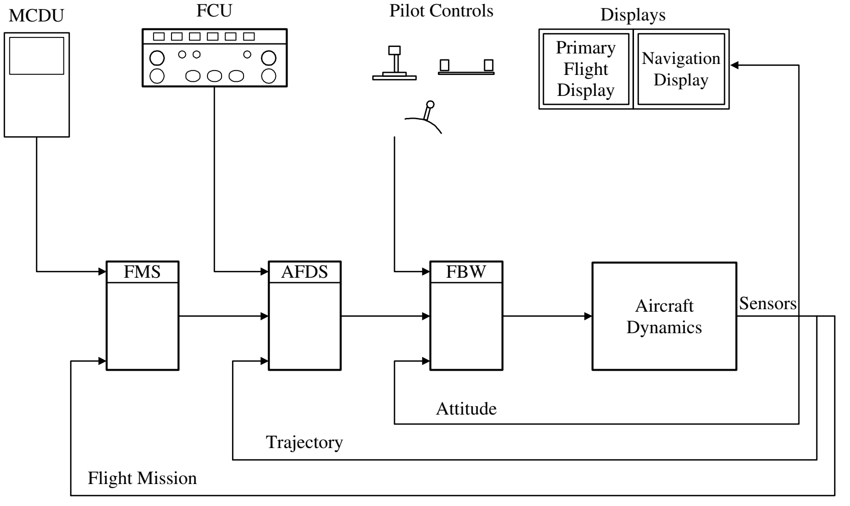

Figure 1.35 Definition of flight control, guidance and management

References

[1] British Aerospace (1990) *Hawk 200.* Marketing publication CO.095.0890.M5336.

[2] Farley, B. (1984) 'Electronic Control and Monitoring of Aircraft Secondary Flying Controls', *Aerospace*, March.

[3] Howard, R.W. (1973) 'Automatic Flight Controls in Fixed Wing Aircraft – The First 100 Years', *Aeronautical Journal*, November.

[4] Daley, E., Smith, R.B. (1982) 'Flight Clearance of the Jaguar Fly-By-Wire Aircraft', *Royal Aeronautical Society Symposium*, April.

[5] Yeo, C.J. (1984) 'Fly-by-Wire Jaguar', *Aerospace*, March.

[6] Kaul, H-J., Stella, F., Walker, M. (1984) 'The Flight Control System for the Experimental Aircraft Programme (EAP) Demonstrator Aircraft', *65th Flight Mechanics Panel Symposium*, *Toronto*, October.

[7] Young, B. (1987) 'Tornado/Jaguar/EAP Experience and Configuration of Design', *Royal Aeronautical Society Spring Convention*, May.

[8] Snelling, K.S., Corney, J.M. (1987) 'The Implementation of Active Control Systems', *Royal Aeronautical Society Spring Convention*, May.

[9] Anthony, M.J., Mattos, F. (1985) 'Advanced Flight Control Actuation Systems and their Interfaces with Digital Commands', *A-6 Symposium*, *San Diego*, October.

[10] White, J.A.P. (1978) 'The Development of Electromechanical Actuation for Aircraft Systems', *Aerospace*, November.

[11] Davies, C.R. (1987) 'Systems Aspects of Applying Active Control Technology to a Civil Transport Aircraft', *Royal Aeronautical Society Spring Convention*, May.

[12] B.G.S. Tucker (1993) 'Boeing 777 Primary Flight Control Computer System – Philosophy and Implementation', RAeS Conference – Advanced Avionics on the A330/A340 and the Boeing 777, November.

[13] James McWha (1995) '777 – Ready for Service',RAeS Conference – The Design & Maintenance of Complex Systems on Modern Aircraft, April.