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V-22 TILTROTOR FLY-BY-WIRE FLIGHT CONTROL SYSTEM

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ABSTRACT

This paper describes the flight control system for the V-22 Osprey tiltrotor aircraft (formerly JVX). V-22 is a multimission, multiservice program to develop a common aircraft for amphibious assault (U.S. Marine Corps), combat search and rescue/special warfare (U.S. Navy), special operation (U.S. Air Force) and long range logistics/cargo (U.S. Army).

Requirements, vehicle characteristics and design trades that influenced the flight control system configuration are summarized. Initial studies baselined a distributed, optically-connected digital fly-by-wire system. Detail design studies showed that the distributed approach resulted in cost, reliability and installation penalties due to the overhead associated with the remote electronics. A centralized configuration was selected. Redundancy management trades led to the selection of a triplex in-line monitored Primary Flight Control System (PFCS), using a pair of processors per channel. Triplex configuration was favored for its ability to isolate processor failures without interchannel cooperation.

This paper reviews the selected configuration in terms of cockpit control configuration, computer architecture, power supplies, actuation, and redundancy management.

BACKGROUND

TILTROTOR DEVELOPMENT

The tiltrotor is a breakthrough development which combines helicopter hover characteristics with the high speed, long range, and high cruise efficiency of turboprop aircraft. The concept has been in development since the 1950s when Bell flew the XV-3 under U.S. Army sponsorship. The concept reached maturity in the NASA/Army/Navy XV-15 program. The Bell XV-15 tiltrotor demonstrator (Figure 1) has confirmed the capability and mission suitability of the tiltrotor. The XV-15 provides the technical basis for the Joint Services Advanced Vertical Lift Aircraft (JVX) which is going into full-scale development (FSD) as the V-22 Osprey (Figure 2).

V-22 PROGRAM

V-22 is a multimission, multiservice program to develop a common aircraft for medium assault (U.S. Marine Corps), combat search and rescue/special warfare (U.S. Navy), special operations (U.S. Air Force) and long range logistics/cargo (U.S. Army). NAVAIR awarded a contract to the Bell-Boeing Tiltrotor Team in April 1983 for a preliminary design phase. The Stage I contract, which included wind tunnel models, simulation, trade studies, critical structural testing and mockups, met the objective to provide a lower risk program in full-scale development (FSD). A Preliminary Design Stage II contract was awarded in May 1984 to complement the Stage I effort and start detail design of major components. An FSD proposal was submitted in August 1984 and updated in February 1985. The FSD contract will be awarded in the summer of 1985. Key characteristics of the proposed design are: nominal gross weight-18,144 kg (40,000 lb), payload-5,443 kg (12,000 lb) for mission radius of 1,298 km (700 nmi), maximum speed-705 km/hr (380 knots). The ferry range of the V-22 will be in excess of 3,894 km (2,100 nmi) which will allow worldwide self-deployment. The vehicle has a wingspan (to the rotor center line) of 14.20 meters (46 ft 6 in.), rotor diameter of 11.58 meters (38 ft) and overall length of 17.65 meters (57 ft 11 in). The vehicle airframe is primarily made of composite material. Six test aircraft will be built. First flight is scheduled for February 1988. Production delivery is planned beginning in 1991 for the 913 aircraft in the current requirement, including 50 for the U.S. Navy, 552 for the U.S. Marine Corps, 80 for the U.S. Air Force, and 231 for the U.S. Army. V-22 will fill a vital need in providing strategic mobility and tactical flexibility for conventional and special forces.

Development work is equally divided between Bell in Fort Worth, Texas and Boeing Vertol in Philadelphia, Pennsylvania. Bell is responsible for the wing/nacelle, propulsion system/rotor and dynamic performance while Boeing Vertol is responsible for the fuselage/empennage, landing gear, subsystems (including flight controls), performance and handling qualities. Boeing Military Airplane Company is avionics integrator. Boeing Commercial Airplane Company performed the preliminary design, first article fabrication and test of the composite wing spar for Bell. Both companies have design teams currently working on the program. The manpower will increase to a peak of 1,600 people at each company during the full-scale development program.

KEY TECHNOLOGIES

Design of a viable production tiltrotor has been enhanced by two key developments in recent years: 1) composite airframe technology which results in a substantial reduction in airframe weight while providing a tough, durable, survivable structure and 2) redundant digital flight control technology which allows the complex control task imposed by the tiltrotor to be achieved at reasonable cost, weight, and safety. Boeing Vertol and other primes engaged in the helicopter development have benefited by technology programs sponsored by the U.S. Army in the past 10 to 15 years. Included here are the Heavy Lift Helicopter (HLH) technology development program of the early 1970s and more recently the ACAP composite airframe development and ADOCS optical flight control demonstration. The V-22 program has also benefited by the experience of the Navy F-18 program which brought the first redundant digital flight control system into production. The Bell-Boeing tiltrotor team has been extensively involved in the Army programs and has carefully reviewed the experience gained on the F-18.

PRELIMINARY DESIGN STUDIES

Bell-Boeing conducted a comprehensive configuration study during the preliminary design phases of the program. The studies were focused initially on a review of actuation concepts and then on a definition of the system architecture.

Actuation

Early swashplate actuation studies considered use of five to six single actuators to support a conventional swashplate. This approach would compensate for an open connection at the swashplate actuator. This approach was abandoned because of control/monitoring complexity (the need to react actuator force fight through the swashplate) and the space and weight needed to install the multiple actuators. A conventional three-point support scheme was selected. Configuration studies for the swashplate actuator considered two basic approaches: 1) use of conventional tandem control valves to control the output ram, 2) direct output flow/force summing at the output ram using high flow electrohydraulic valves. The conventional tandem control valve was selected based on performance and developmental risk.

System Architecture

Primary drivers in the selection of the system architecture were the method of channel failure monitoring and the disposition of actuator control electronics. Two methods have been used to provide the two-failure operational capability specified for the vehicle primary flight control functions: cross-channel monitoring where four channels vote inputs and outputs to detect failures and in-line monitoring where three channels use self-monitoring to detect failures within a local channel without dependance on others. In the configuration considered for the Osprey, in-line monitoring is accomplished by the provision of two processors within each channel either of which can accomplish fault reaction. This approach was preferred for its capability to isolate a failed channel processor without intervention by other channels. The resulting configuration is a hybrid with respect to input monitoring in that it uses cross-comparison of some sensor inputs to detect failures not covered by in-line monitoring. The selected approach also allows interchannel communication for the synchronization and equalization needed for integrator operation. The truly "brick-wall" in-line monitored system is possible only with a relatively simple unscheduled linkage. The emphasis here is to cover the processor failure and use in-line monitoring as the primary but not exclusive approach.

The actuator control electronics studies considered a number of options for location of actuator control electronics. These ranged from separate control of each actuator group (i.e., control of a set of swashplate actuators) to control of a full wing side of actuation, and control of all system functions in a central electronic unit. Distributed architectures were coupled using redundant optical data busses. Data for the triplex and quadruplex channel architecture approaches were developed for each case. A centralized configuration was selected based on cost, reliability, maintainability and installation considerations. Figure 3 shows a normalized comparison of the system configurations. The centralized configurations incorporate one electronic unit per channel located in the avionics bay. The distributed configurations include one electronic unit per channel in the avionics bay and one unit per channel at the end of each wing to service wing and nacelle equipment. Factors shown are for the system electronics and wiring without including the cockpit controls and actuation. Weight shown includes cabling using 22 gauge wire with 50 percent double shielding.

Independent Backup

The study also determined that an independent backup should be provided for initial FSD aircraft. The backup is needed until the system software has been fully evaluated over a period of time in flight and shown to be free of generic failures. The backup is mechanized as a dual-path analog channel which commands the Primary Flight Control System (PFCS) analog servoloops. Monitoring is for maintenance only and does not preclude pilot selection which is manual. Performance on backup will approximate that achieved with the digital PFCS.

SYSTEM DESIGN

DESIGN DRIVING REQUIREMENTS

Specification requirements which have strongly influenced the V-22 flight control system design include: safety, flying qualities, reliability and survivability. The configuration of the system will satisfy these important requirements without introducing excessive complexity which would drive cost and weight, factors which must also be acknowledged as important drivers in any successful design.

Safety

Operation after any two failures is desired. The design meets this requirement in signaling paths and power supplies. Exception is taken for the dualized power sections of the swashplate actuator, its mounting connection, and for second mechanical failures of the actuator transducers.

Flying Qualities

With the full system operating, the requirement is for Level 1 flying qualities in accordance with MIL-F-8785C (forward flight) and with MIL-F-83300 (hover/low speed flight). With only the flight critical primary control functions operating, the requirement is for Level 2 flying qualities in accordance with these specifications. The system is single fail-operate/failsafe for augmentation functions. Primary control functions allow completion of all mission tasks, but at increased pilot workload.

Reliability

Requirements cover flight safety, mission and maintenance reliability. The specified mean time between **flight critical failures** of $1x10^7$ hours is a primary driver of redundant channel architecture and actuator/power supply configuration. This requirement dictates two fail-operate for signal paths and the power supplies. It is dominated by single and dual point failures within the flight critical rotor swashplate actuators.

Mean time between **mission aborts** is defined by the need to terminate a mission when a subsequent failure would mean loss of the vehicle or when a mission critical function is lost. The requirement is 2,155 hours. This factor is usually affected equally by electronics, actuation, and power supplies.

Mean time between **unscheduled maintenance actions** is allocated from the overall vehicle level. The current allocation is 125 hours. This failure rate is controlled by careful selection of system designs and components which have inherent simplicity and by design of failure monitoring techniques to minimize nuisance failure indications which result in no-defect removals. The trend toward large scale integration of electronic functions promises to dramatically reduce the failure rate of flight control electronics. Given this opportunity to reduce parts count, designers must resist the temptation to push the complexity of the system because it is now possible to do almost anything with the emerging microprocessor technology.

Survivability

The system must be designed to survive in the future electronic battlefield. Given the composite structure of the vehicle, natural lightning dictates careful design of system interfaces and cabling. Approaches used to enhance survivability include:

A) Separation of channel electronics, cabling and hydraulic lines

B) Use of interface filters and transient suppression

C) Routing of cabling in shielded raceways

D) Use of rip-stop design and jam clearing features in actuators.

OPERATIONAL ELEMENTS OF THE SYSTEM

Figure 4 shows the operational elements of the V-22 flight control system which are integrated by the flight control electronics. Pilot command inputs are via redundant Linear Variable Differential Transformers (LVDT) attached to conventional displacement controls. The Swashplate actuators control rotor pitch in all flight regimes; they are the only control elements needed to operate the vehicle in hover/low speed flight. Maneuver inputs to these actuators are phased out during conversion so that only collective pitch input for power control and cyclic pitch inputs for gust alleviation are allowed in cruise.

Conversion from the helicopter to airplane configuration is accomplished by rotating the nacelles via the conversion actuator (linear ballscrew type). The aerodynamic surfaces (flaperon, rudder, elevator) are used as the primary maneuvering and trim controls in transition/cruise flight.

The flight control system interfaces with the avionics data bus to obtain sensor inputs and to provide status information. The outer control loops of the engine are controlled through an interface with the Full Authority Digital Engine Control (FADEC) units associated with each engine.

EQUIPMENT INSTALLATION

Figure 5, a cutaway of the Osprey, locates the flight control equipment shown in Figure 4. Flight control electronics and attitude/heading sensors are in a closed avionics bay aft of the pilots' station. System cabling is routed via shielded raceways and conduit to the actuators located in the wing, nacelle and tail. Cabling must pass through two rotating joints (in the wing fold and nacelle pivot) to reach the nacelle. Single ram electro-hydraulic actuators are used to control the aerodynamic surfaces.

Each rotor transmission drives a hydraulic pump dedicated to flight control. The rotor transmissions are connected via a cross shaft which also drives a center accessory gearbox. This gearbox drives a third hydraulic pump which is shared between flight control and utility functions. Permanent magnet generators dedicated to flight control are integrated with deice generators located on the left rotor transmission and center accessory gearbox. Main generators are located on the right rotor transmission and center accessory gearbox. The APU also drives one of the main generators and the flight control/utility hydraulic pump.

SYSTEM FUNCTIONS/INTERFACE

The basic functions and interface between the Primary Flight Control System (PFCS), Automatic Flight Control System (AFCS) and aircraft are shown in Figure 6. The pilot makes direct input to the rotor, aerodynamic surfaces and engines via the PFCS feed forward shopping, gain scheduling and mixing. Flight critical aircraft parameters (e.g., airspeed, nacelle angle, rotor rpm) are feedback directly to the PFCS. Other stabilization feedbacks are made through the AFCS. Sensor inputs are shaped to augment static and dynamic stability and provide gust rejection. Pilot inputs are also passed to the AFCS where a command model modifies stabilization feedbacks to provide desired rate, attitude and linear velocity response. This approach allows tailoring of both control response and stability. The net AFCS output is sent to the PFCS via a rate/authority limited differential interface and a low rate parallel path for steady state trim inputs. These interfaces limit the effects of multichannel AFCS failure to acceptable levels.

CHANNEL ARCHITECTURE

Figure 7 shows triplex channel architecture used in the Osprey primary flight control function. Details on monitoring of sensors and actuators are given later in the Redundancy Management Section. Single self-monitored sensors are provided for each channel. These inputs are passed to the redundant PFCS processors (A and B) via a nonredundant sensor interface and I/O processor. Self-checks of the sensors by the A and B processors also check the nonredundant interface and I/O processor. Each processor calculates output commands and expected response. Each processor controls the engagement of the analog servo loops through dedicated discrete outputs. Output commands from processor A are provided to the actuators by the I/O processor; performance of the output processing and actuator is checked by each processor (A or B) using wrap-arounds. Each processor incorporates a hardware watch-dog monitor which must be serviced during each major frame; otherwise all actuators associated with the channel are shutdown via hardware logic.

MAJOR ELEMENT DESCRIPTION

This section gives details on the cockpit controls, flight control electronics, power supplies and flight control actuators planned for V-22.

COCKPIT CONTROLS

Conventional trimmable mechanically synchronized displacement type controls have been selected for V-22. Small displacement unique trim sidearm controls were evaluated in configuration studies, but rejected because of lack of experience with such controllers on VTOL aircraft. Figure 8 shows a block diagram of the selected arrangement. Four motion transducers provide primary control inputs to the flight control computers and the backup computer. Force feel gradients are programmed with nacelle angle for the longitudinal, lateral and directional control axes. Force feel in the thrust/power axis is provided by an adjustable friction brake. Each axis incorporates a control drive actuator for parallel trim and autopilot inputs. Thrust/Power friction is relieved when automatic parallel drive is operating. Each axis incorporates a backup magnetic brake to hold trim when the control drive function is not operating.

Figure 9 shows the mechanical arrangement of the thrust/power axis which is typical of other axes. The arrangement allows operation of at least one control station after any open. All transducers and the force feel devices are designed to shear open if jammed. A separate damper provides feel if the force feel unit is disconnected. Series trim is available in all axes to circumvent a jam. Cross links are designed to minimize jam after open and foreign object entrapment. The fourth motion transducer is shared with the primary control channels as a standby so that dual fail-operate capability is provided for both electrical and mechanical failures to the pilots' input transducers.

FLIGHT CONTROL COMPUTER

All flight control processing and control functions are packaged in a central flight control computer. Figure 10 shows the internal architecture of the unit. The input/output processor handles all of the digital data flow in and out of the unit with exception of the dedicated discrete outputs used by the PFCS processors (A and B) to initiate fault reaction control of the flight control actuators. In addition, the I/O processor acts as controller for the dedicated flight control data bus associated with each channel. This MIL-STD-1553B bus ties in a Standard Attitude Heading Reference (SHARS) unit, the Full Authority Digital Engine Control (FADEC) units and cockpit discrete inputs associated with the channel. The I/O processor also handles cross channel communication and outputs to the avionics MIL-STD-1553B data bus.

Figure 11 shows the physical arrangement of the flight control computer. The unit employs a large number of analog and digital custom large-scale integrated devices to minimize discrete parts count. Processing will be accomplished using the Fairchild 9450 implementation of the MIL-STD-1750A instruction set operating at a clock rate of 15 megahertz. General Electric Aerospace Control Systems Department is supplying the Flight Control Computer and other electronic units of the V-22 flight control system. GE is also assisting Bell-Boeing in detail system design and system integration.

Flight critical software will be developed using an extensive library of assembly code macros and standard algorithms. Boeing has developed a discrete graphical representation for each function. Control laws are developed through simulation programmed using these same building blocks. Control laws are specified in block diagram format using the graphic representation. The same definition is used by all groups involved with software. Final validation will be achieved by tieing the flight hardware back into the simulation which generated the original requirements.

ELECTRICAL POWER SUPPLY

The flight control computers are powered from the aircraft 28 VDC busses, dedicated and aircraft battery power, and dedicated permanent magnet generators as shown in Figure 12. The dedicated battery is changed and tested by the FCC. Battery selection is also controlled by the FCC. The power supply is configured to handle three operating conditions:

A) Ground operation with rotors stopped: Power is supplied from the No. 1 and No. 2 DC and DC essential busses. Extended battery demand is inhibited to prevent discharge. APU derived DC is the primary input.

B) Rotor Start/Shutdown: In channel 2, dedicated battery power is provided to backup the APU supplied 28 VDC during rotor start and shutdown.

C) Rotors Turning: Primary power is via the permanent magnet generator and DC essential bus. DC busses back up the permanent magnet generators.

Batteries are also used to suppress transient power dropout in all three operating regimes.

FLIGHT CONTROL HYDRAULIC POWER

As shown in Figure 13 each set of flight control actuators is supplied by the two flight control hydraulic systems. All functions except the rudder are also supplied by the flight control/utility system. The flight control/utility system also powers flight control No. 1 and No. 2 circuits via ground checkout switching valves.

The swashplate actuators are most critical since they are required in all flight regimes. To provide a high integrity supply for these actuators, transmission driven supplies are protected from leaks in the wing and fuselage by isolation valves in the nacelle. These valves shut when loss of fluid from the system is detected. The flight control/utility system backs up the flight control supply to the swashplate actuators via switching valves. The FCS has logic to detect failures of lines down stream of these valves so that both systems are not lost in the event of leakage in this area.

SWASHPLATE ACTUATOR

The Swashplate actuator (Figure 14) is an integrated design with a three-valve control stage driving a conventional dual-tandem main control valve. The MOD piston/MCV configuration is quite similar to that used in the F-16 integrated servo actuator. A pair of electrohydraulic valves (EHV) are flow summed on to one MOD piston while a single EHV controls a ½ area MOD piston associated with the second hydraulic supply. (In the F-16 this is a standby channel.) The two MOD pistons force fight in normal operation, but because of the area unbalance the deadband normally produced by a one-on-one force fight is not seen. The large piston dominates performance. Excellent threshold and dynamic performance has been achieved on a brassboard of this actuator. It does not have time-critical first failure modes. Second failure transients are limited by using a low velocity gain in the control stage (i.e., low-flow valve with large piston area).

The unit is designed to have very high chip-shear force in the main control valve to preclude the need for anti-jamming devices. The output ram is a dual tandem configuration with a stroke of 18 inches. The unit will be designed to withstand and/or clear the damage resulting from ballistic impact. One EHV of each actuator is interfaced to each FCC. Feedback of EHV, MCV and ram position is supplied to each FCC. In addition a standby input is supplied to the single EHV hydraulic side. The standby input is required to meet reliability specifications for the case of a hydraulic loss to the flow summed side followed by a electronic control loss to the single EHV side. Allied Bendix Electrodynamics Division is supplying the swashplate actuator for V-22.

AERODYNAMIC SURFACE ACTUATORS

A generic configuration of the V-22 aerodynamic surface actuator is shown in Figure 15. The actuator consists of a single EHV with monitor LVDT, differential pressure transducer for load equalization, a bypass/damper valve to block EHV output and bypass the actuator into a damping mode, and ram position LVDT. Each actuator is controlled by a single flight control computer. The features included in each application are as follows:

A) Flaperon and Rudder: These surfaces may be shutdown and the actuator provides flutter damping.

B) Elevator: The need for good resolution and the close mechanical coupling lead to the provision of differential pressure feedback for equalization of the three single actuator outputs. The sensor is also used to improve the damping of the actuator in the presence of surface inertia in through dynamic pressure feedback. The actuator incorporates jam clearing features because jamming in the airplane mode can not be tolerated.

Moog, Incorporated, Aircraft Controls Division is supplying the Flaperon Actuator. The elevator and rudder actuators are still in competition.

CONVERSION ACTUATOR

The conversion actuator (Figure 16) consists of a single, telescoping ballscrew normally powered by redundant hydraulic drive units (integral with the wing attach end of the actuator) and alternately powered by an electrical drive (integral with the nacelle end of the actuator). Simultaneous operation of both hydraulic drive units is the normal mode, with the brakes on the backup electrical drive unit providing the ground. Differential pressure transducers are used to equalize hydraulic motor output torque and to control preload at the nacelle downstop in the cruise condition. Dual shutoff valves are provided. EHV interface is similar to the other actuators. With hydraulic drives off, the electric motor drives with the brakes on the hydraulic drive units providing the ground. The unit makes extensive use of redundant load paths in mounts, geared paths and ball screws. Redundant ball circuits are provided. Ice scrapers are sized to maintain structural integrity after loss of both ball circuits. Each hydraulic drive unit is interfaced with a separate flight control computer. The electrical drive unit is interfaced with the third flight control computer and with the analog backup computer. Western Gear Applied Technology Division is supplying the conversion actuator for V-22.

REDUNDANCY MANAGEMENT

This section gives some details on failure monitoring of PFCS and AFCS functions. The overall PFCS redundant channel architecture has been described at the system level.

PFCS SENSOR INTERFACE/MONITORING.

Linear Variable Differential Transformers are the primary motion transducers used in the PFCS. Figure 17 shows a typical LVDT/Flight Control Computer interface. The differential output of the LVDT secondaries is proportional to probe position while the sum of the secondary outputs is essentially constant with probe position. The difference is used in control laws while the sum is checked to determine the electrical integrity of the device and its interfaces. The difference is demodulated and passed directly to software using a dedicated analog-to-digital convertor channel. Both the sum and difference are also multiplexed into software. In software the sum is checked to define the electrical integrity of all elements of the path. The continuous and sampled difference are compared to isolate faults to the computer or external circuitry. This method cannot detect mechanical failures between the LVDT probe and the using device. These failures (which are of low probability) are detected by interchannel comparison.

Resolvers used to measure nacelle angle are checked in software using the relationship $\sin^2 x + \cos^2 x = 1$ to verify electrical integrity. Other inputs such as airspeed and rotor rpm are self-monitored using reasonableness tests.

TYPICAL ACTUATOR INTERFACE/MONITORING

Each of the three PFCS channels controls response of a portion of the actuation channels associated with a given control function (i.e., one channel of a triplex swashplate actuator). Channel outputs are summed in the control stage of the actuator or at the control surface itself. Actuator failure detection is accomplished by monitoring current drive and the second stage spool position of the electrohydraulic valves, and the integrity of feedback LVDT.

Figure 18 shows a typical actuator/FCC interface including the analog servo loop closure. Wraparounds to the digital processors include: servo command, EHV drive current, EHV spool position, and ram position. Shutdown of hydraulic power is controlled by dual hardware latches driven by the channel processors (A and B).

Methods used to monitor actuator position LVDT, the EHV drive current and EHV input/output performance in software are shown in Figure 19. The LVDT self-monitor has been discussed. EHV drive current is synthesized in software by forming a digital equivalent of the analog servo error and processing it through a servo amplifier model (gain and limit). The model output is compared with actual current. In a similar manner the EHV model (gain, lag and limit) predicts EHV spool position for comparison with measured spool position. Also shown is the actuator shutoff indication generated by EHV spool offset when pressure is removed from the channel.

As has been noted under System Design, the I/O processor and PFCS processors (A and B) are checked as part of the sensor and actuator monitoring described above.

CHANNEL SYNCHRONIZATION

The channels of the PFCS and AFCS are frame synchronized to assure proper control of channel integrators and the availability of congruent data sets for interchannel comparison. The synchronization mechanization must be shown to be free of single failure points.

FAULT REACTION

Each processor outputs to a redundant hardware latch which controls hydraulic power to the actuators. Shutdown of the third PFCS channel is inhibited. A dedicated fault display is provided for first and second failure annunciation. Detail failure information is provided via the avionics control/display unit. A system reset is provided to clear nuisance faults.

AFCS INTERFACES/MONITORING

The AFCS is configured as a triplex cross-channel monitored system providing operation after first failure in most triplex sensor and signalling paths. On second failure the function is shutdown in a failsafe manner. Certain selectable modes of the AFCS are failsafe after failure of a single sensor input.

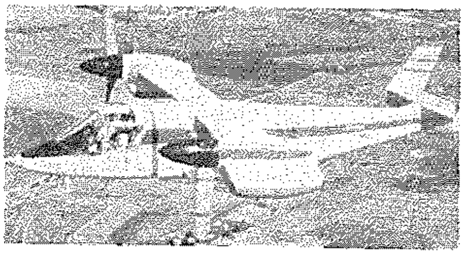
A) Sensors - Triplex sensors are received hardwired or via the flight control data bus. A single sensor input is made to each FCC. Inputs are exchanged through the inter computer data links and voted. Single sensor inputs are received on the avionics data bus. Sensor validity discretes are used to disengage using selectable modes in case of sensor or bus failure. The AFCS control laws incorporate limiting to suppress failure of nonredundant sensors.

B) Outputs - Computed outputs are passed to the PFCS Processors by the I/O processor for voting.

CONCLUSION

This paper has provided background on the V-22 Osprey and trades made to configure the flight control system. The system was described in terms of structure and physical installation. Major elements of the system including cockpit controls, control electronics, power supplies, and actuation were also described; finally the approach to system failure monitoring at the sensor, processing and actuation level was reviewed.

At this time the system is in detail design. Initial full-system hardware integration testing is scheduled for February 1987. All major risk areas have been addressed and system development is expected to proceed on schedule. Bell-Boeing views the V-22 as a major opportunity to field an aircraft of unique capability and is working to achieve that goal.



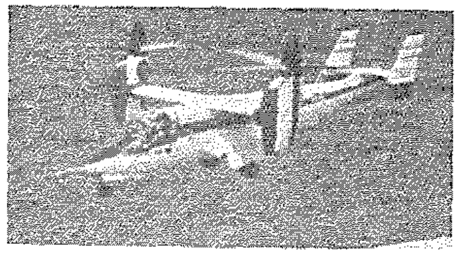


Figure 1. XV-15 Tiltrotor Demonstrator.

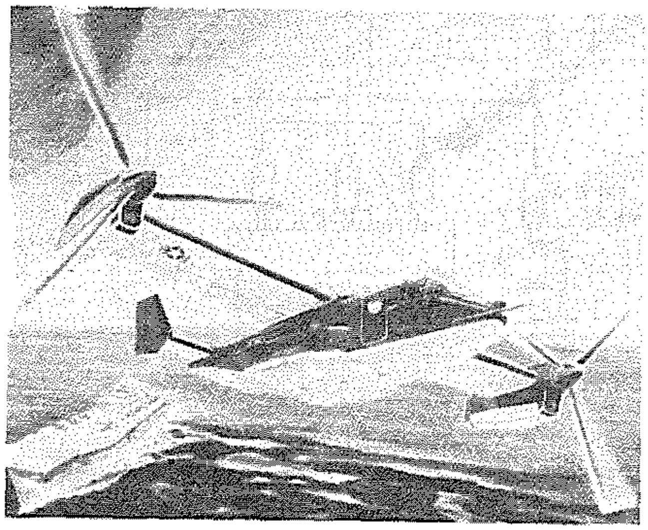


Figure 2. V-22 Osprey.

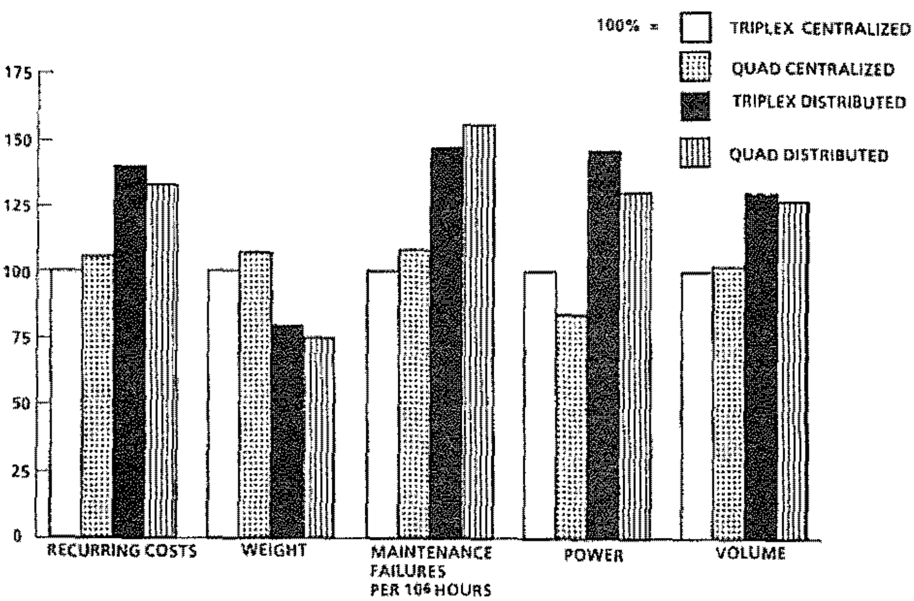


Figure 3. Normalized Comparison — PFCS Electronics/Wiring.

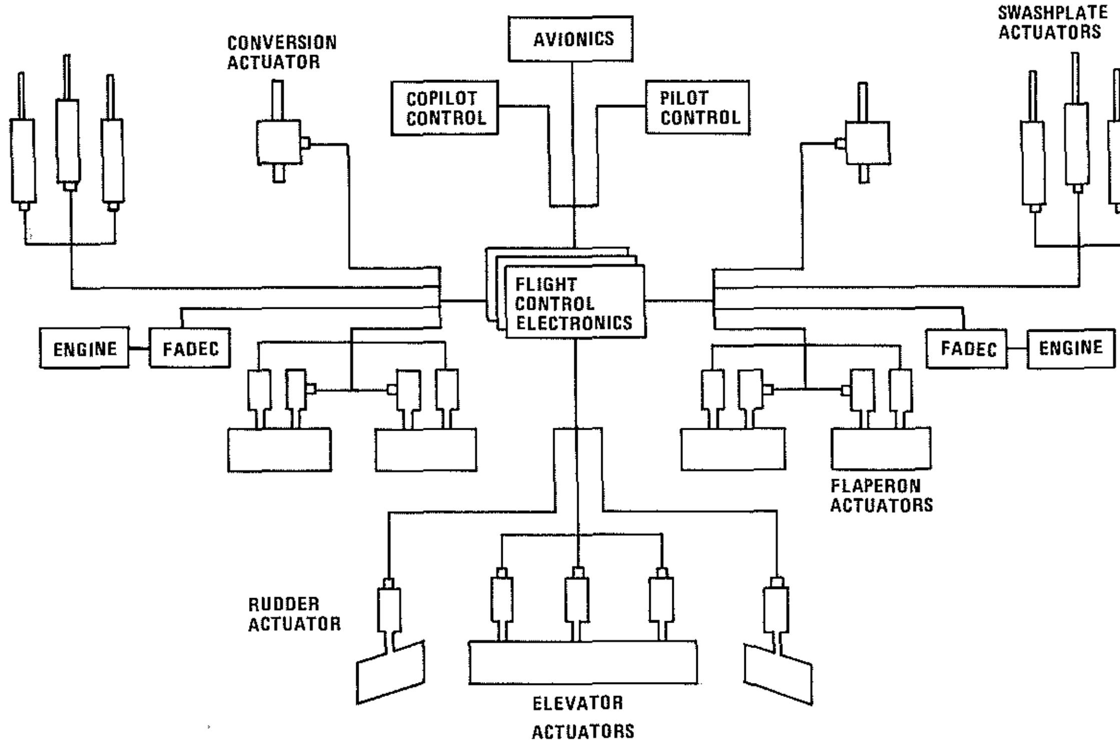


Figure 4. Flight Control System Elements.

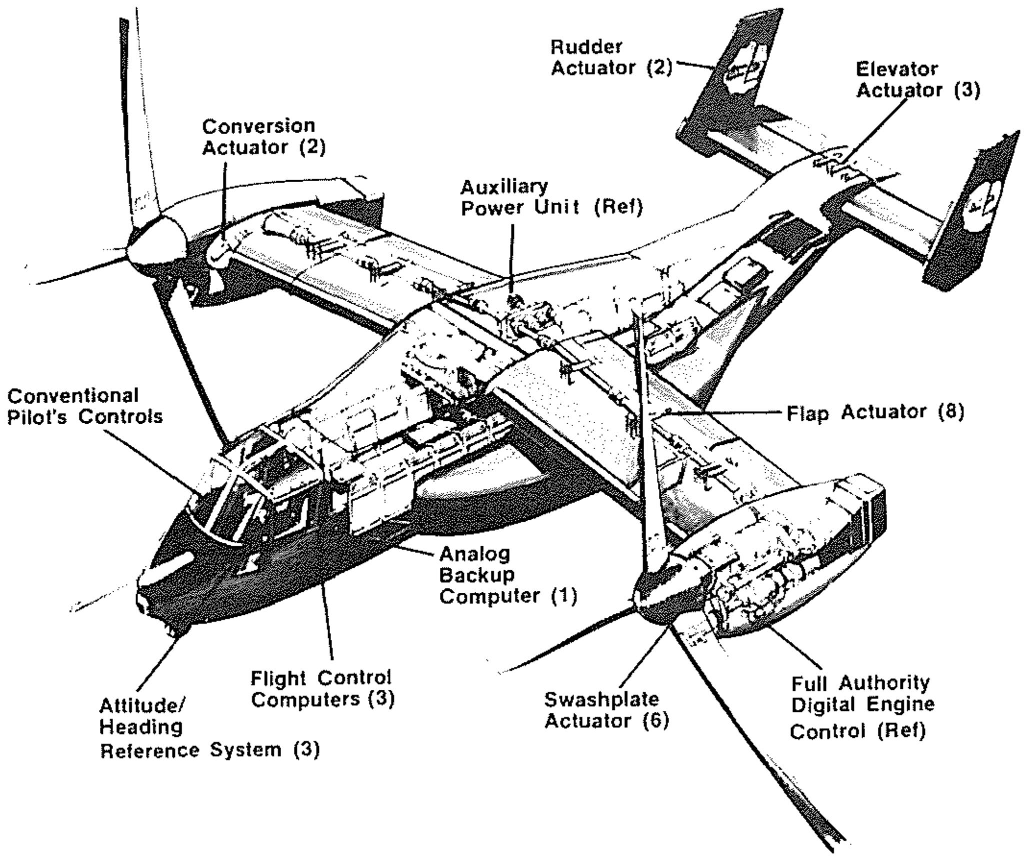


Figure 5. V-22 Flight Control System Equipment Installation.

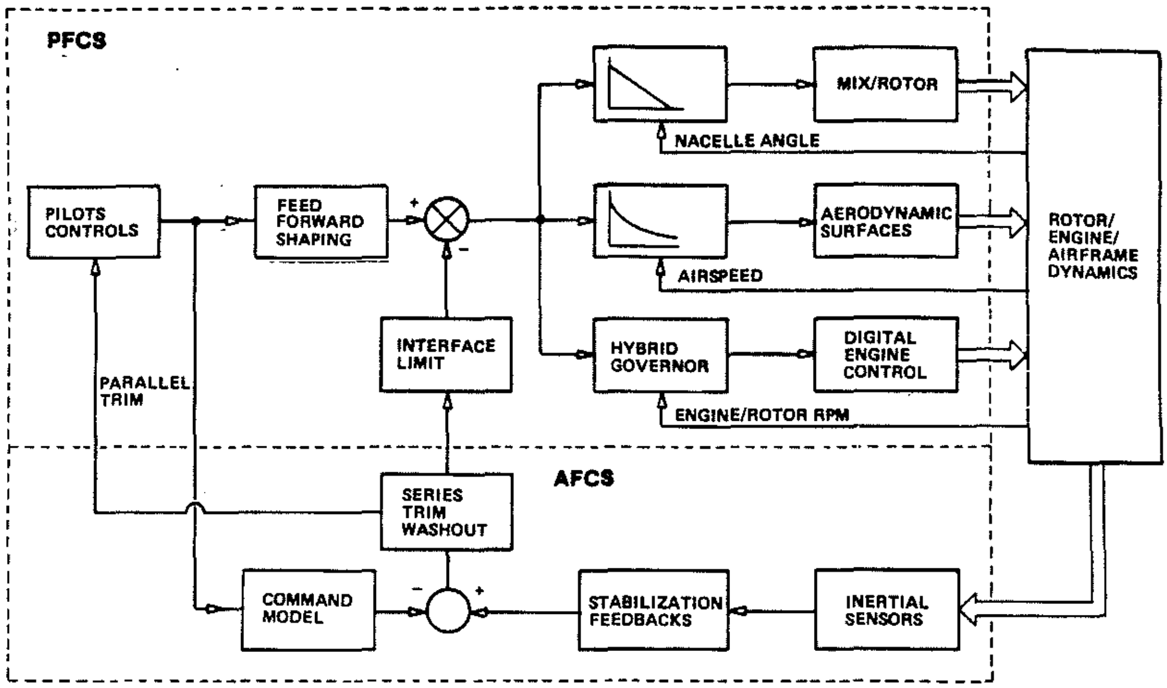


Figure 6. V-22 Flight Control Basic Functions/Interface.

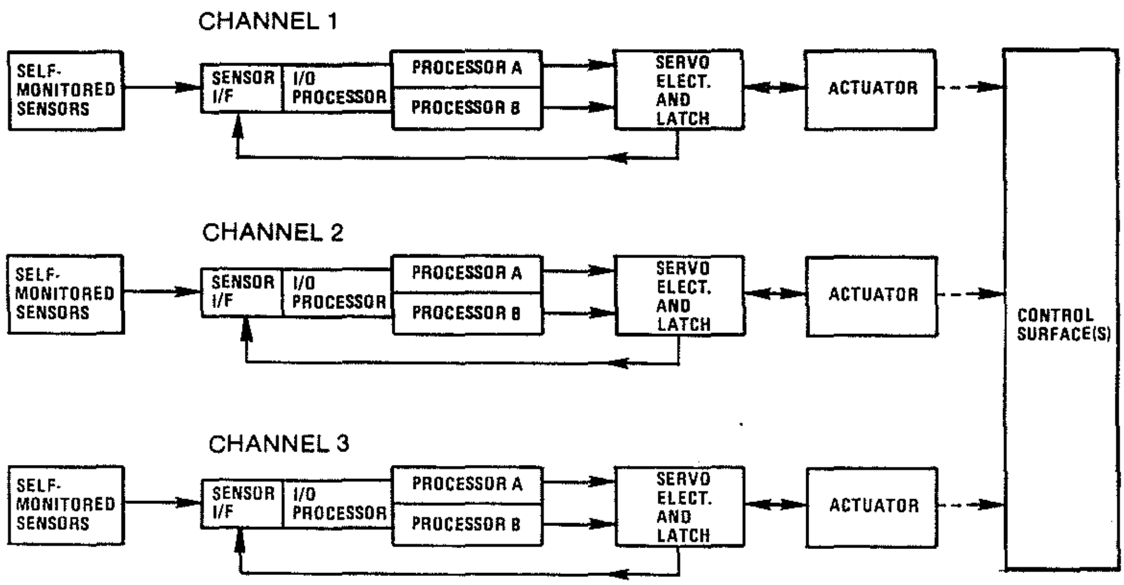


Figure 7. PFCS Channel Architecture.

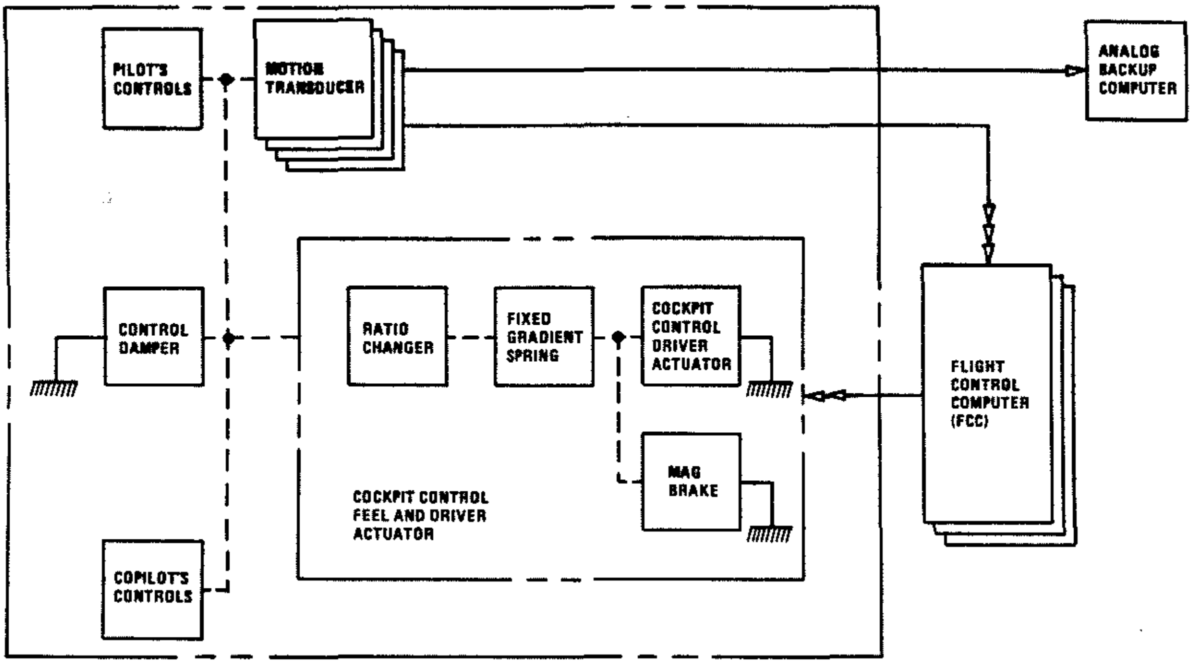


Figure 8. Cockpit Control Configuration.

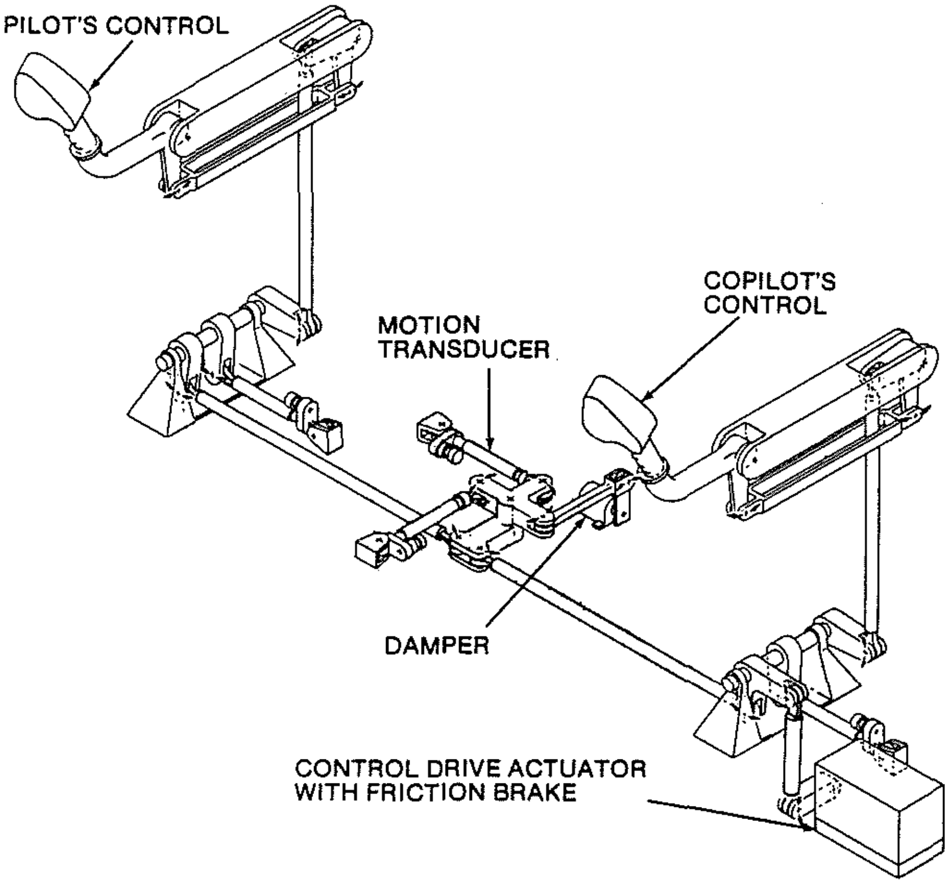


Figure 9. Thrust/Power Control.

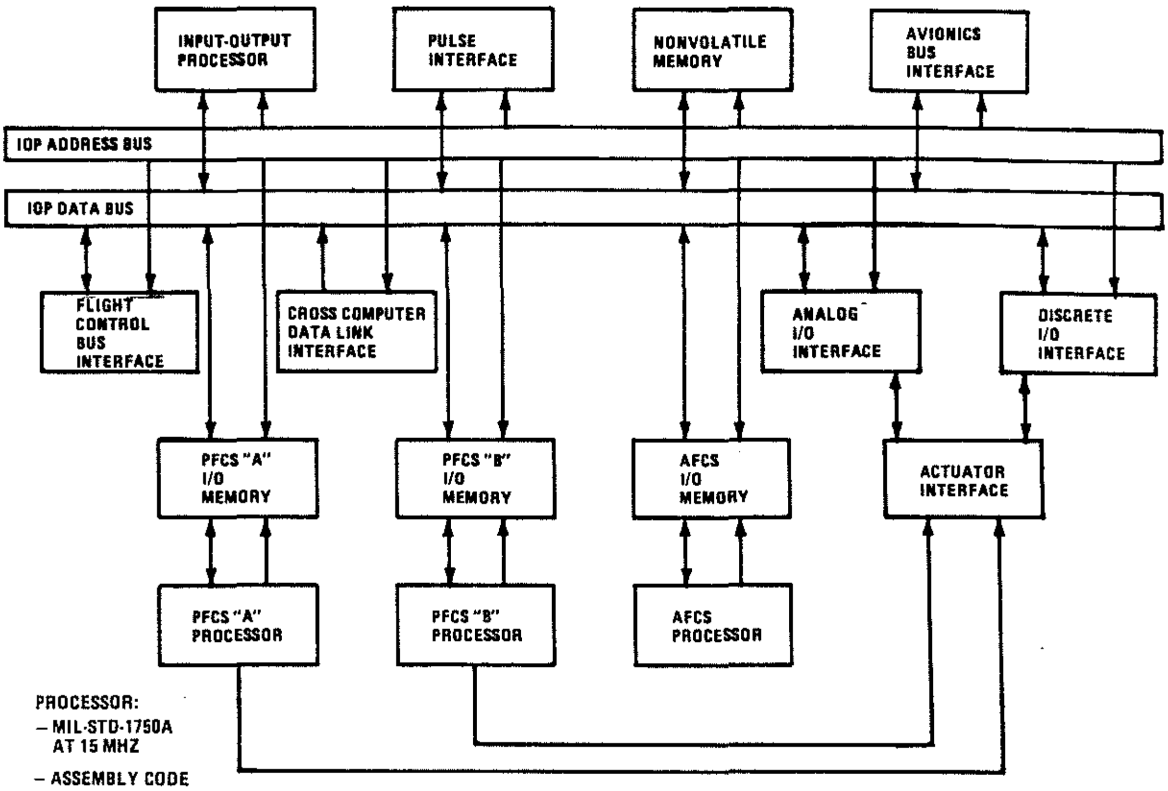


Figure 10. Flight Control Computer Architecture.

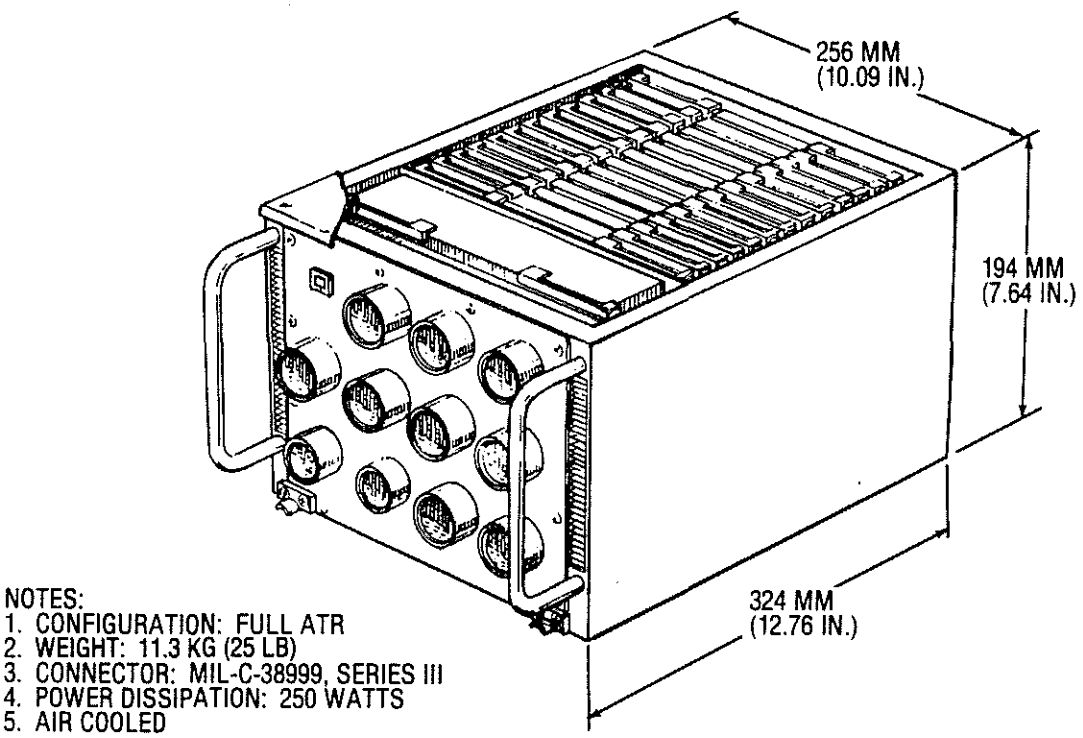


Figure 11. Flight Control Computer.

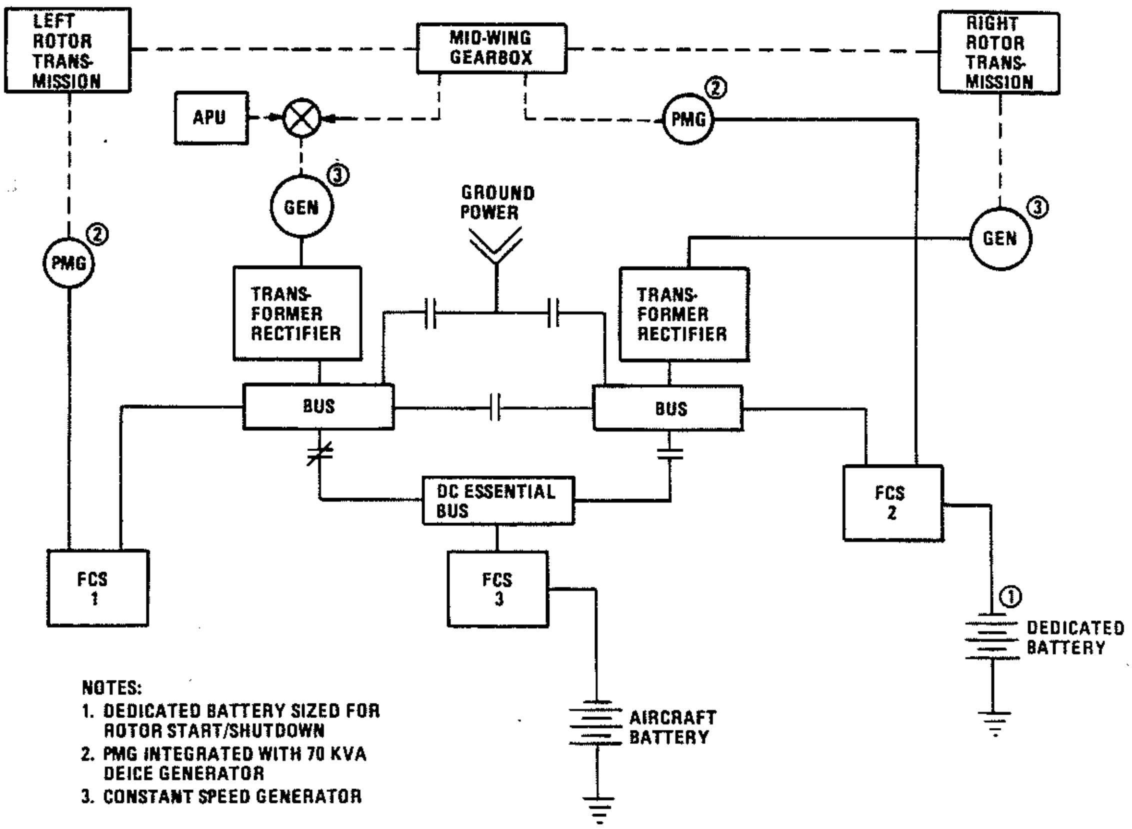


Figure 12. Flight Control System Electrical Power Sources.

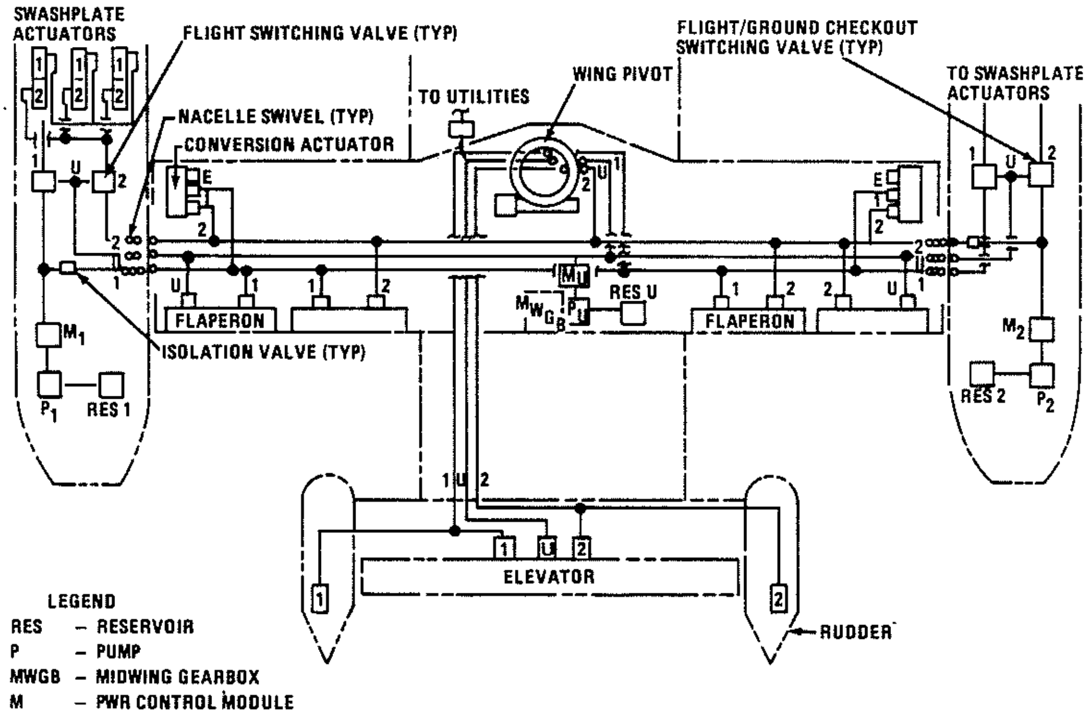


Figure 13. V-22 Flight Control Hydraulic System.

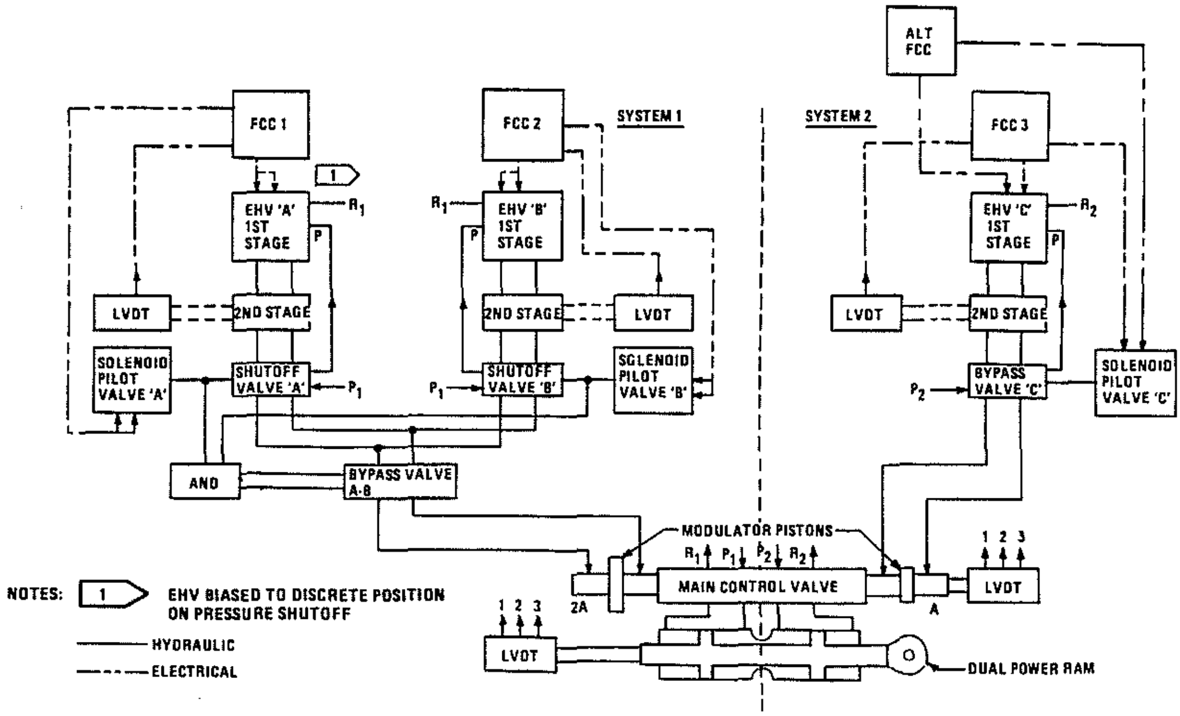


Figure 14. V-22 Swashplate Actuator.

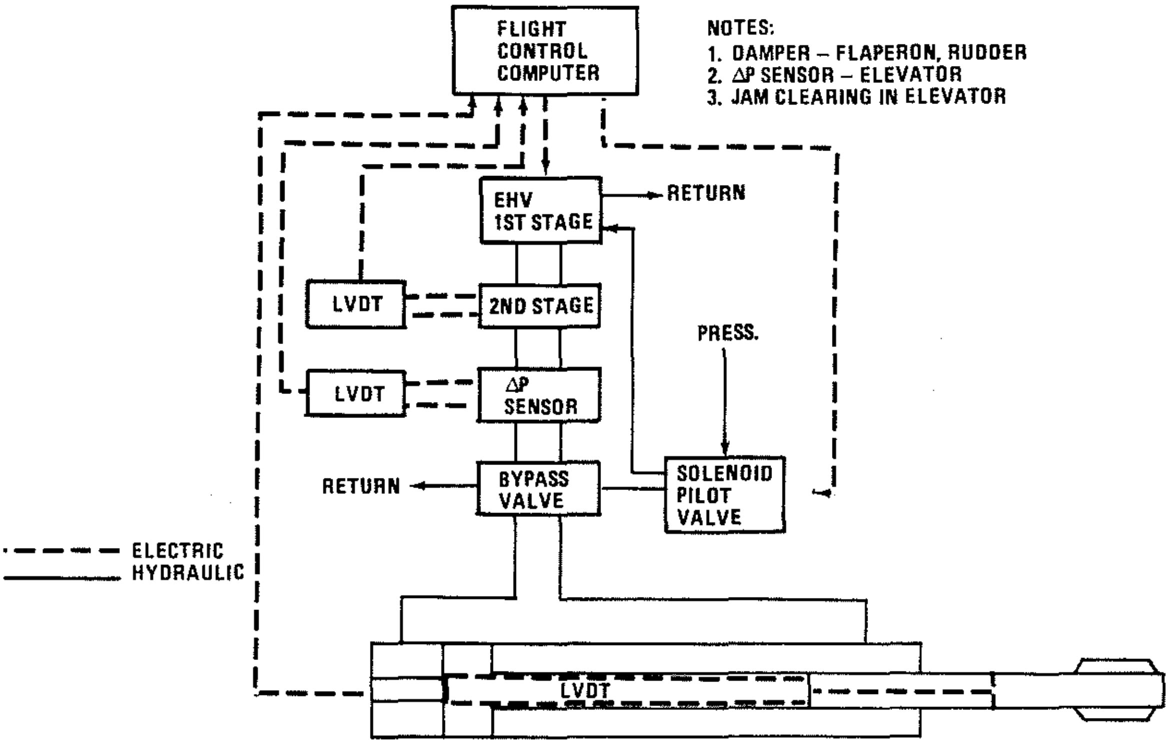


Figure 15. Simplex Linear Flight Control Actuator for V-22 Flaperon, Elevator, and Rudder.

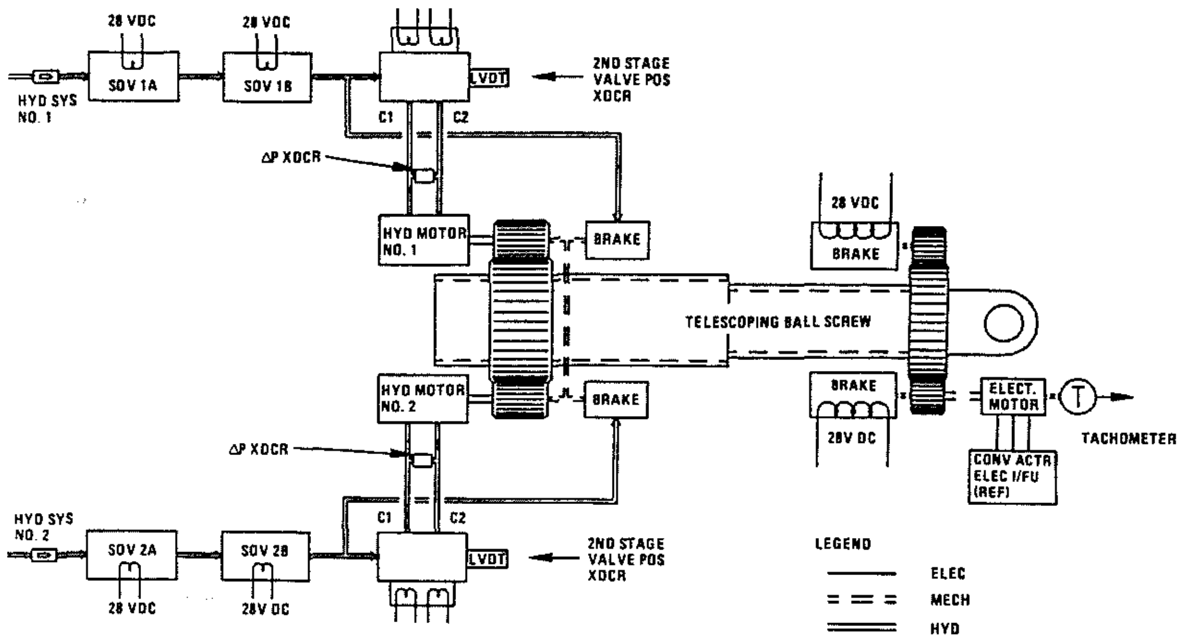


Figure 16. V-22 Conversion Actuator.

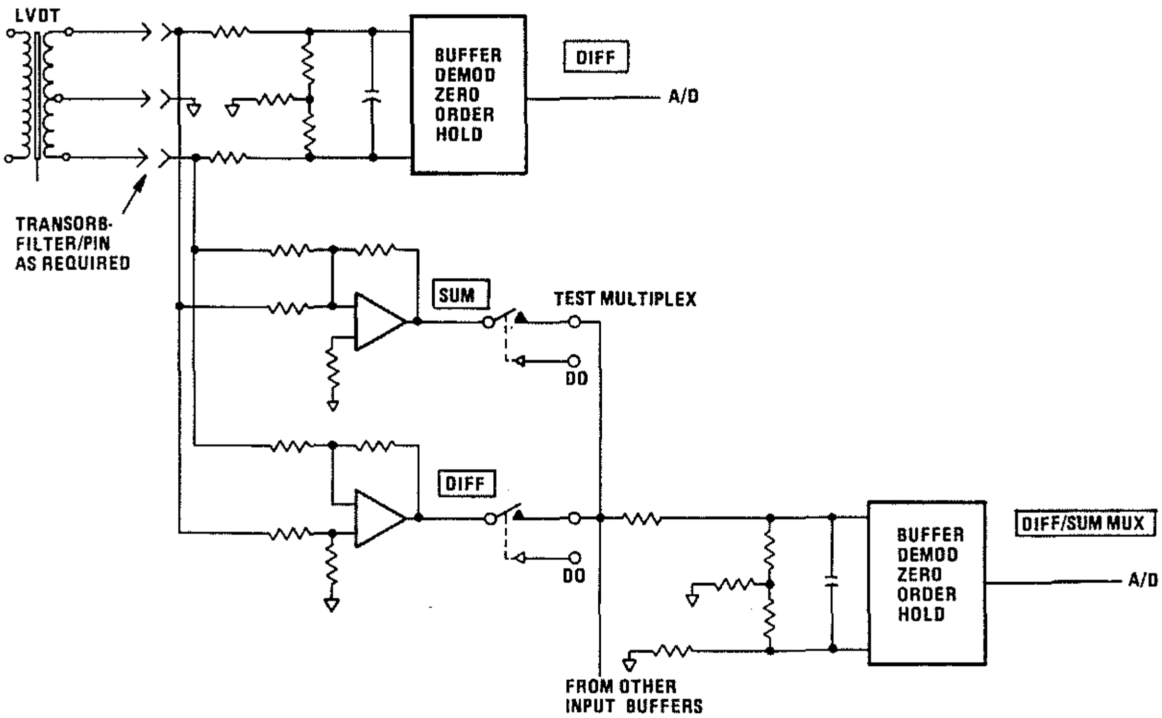


Figure 17. Typical LVDT Interface.

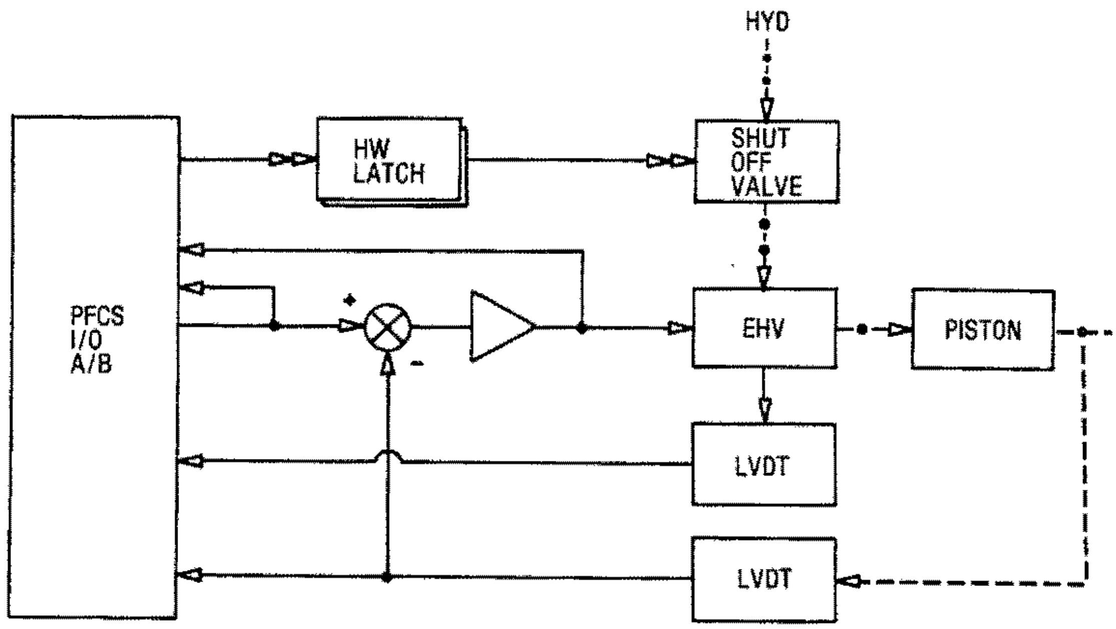


Figure 18. Typical Actuator Interface.

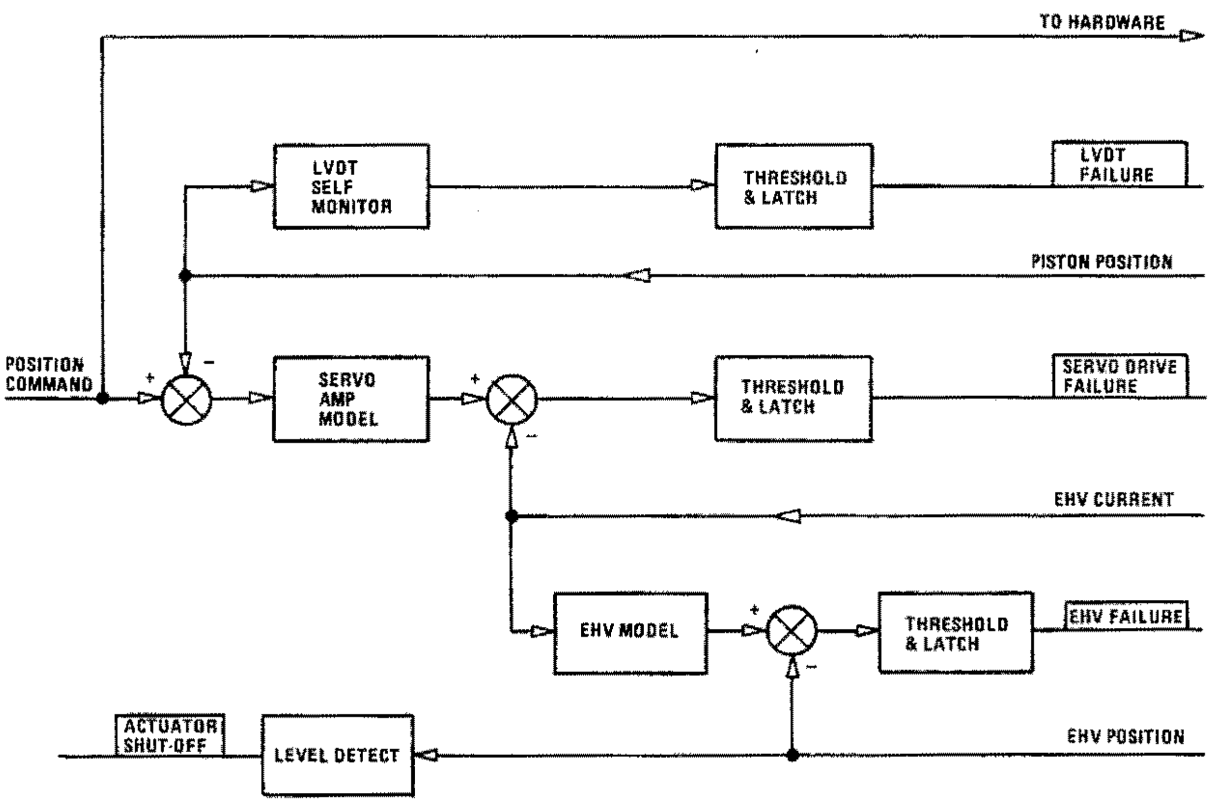


Figure 19. Typical Actuator Monitoring in Software.