NASA Contractor Report 144865

DESCRIPTION AND THEORY OF OPERATION OF THE COMPUTER BY-PASS SYSTEM FOR THE NASA F-8 DIGITAL FLY-BY-WIRE CONTROL SYSTEM

Sperry Flight Systems
Phoenix, Arizona

(NASA-CR-144865) DESCRIPTION AND THEORY OF OPERATION OF THE COMPUTER BY-PASS SYSTEM FOR THE NASA F-8 DIGITAL FLY-BY-WIRE CONTROL SYSTEM Final Report (Sperry Flight Systems, Phoenix, Ariz.) 99 p HC A05/MF A01 CSCL 01C G3/05 N78-22076 Unclas 16676

May 1978

NAS4-2248

National Aeronautics and
Space Administration

Hugh L. Dryden Flight Research Center
P.O. Box 273
Edwards, California 93523
AC 805 258-3311

TABLE OF CONTENTS

TABLE OF CONTENTS (CONT)

ILLUSTRATIONS

TABLES

TABLES (cont)

DESCRIPTION AND THEORY OF OPERATION OF THE COMPUTER BYPASS SYSTEM FOR THE NASA F-8 DIGITAL FLY-BY-WIRE CONTROL SYSTEM

Prepared by

Sperry Flight Systems

INTRODUCTION

The main objective of the phase II digital fly-by-wire (DFBW) program was to evaluate a triplex redundant DFBW flight control system. The single channel digital flight control computer used in phase I of the program was replaced in phase II by a triplex redundant digital flight control system configuration.

A simplified block diagram of the signal flow for a single channel is shown in figure 1. Although this block diagram does not show the complete triplex system, it illustrates the manner in which a single channel operates. The system may be engaged in the manual or auto mode via the status/engage panel. When engaged, the engage signal flows to the bypass and servo electronics (BASE) units for monitoring and routing to the secondary actuator solenoids.

The pitch, roll, yaw, and trim commands from the cockpit control sensor flow to the computer bypass system (CBS) and to the digital computer system (DCS) electronics located in the BASE units. A discrete command from the mode and gain panel is fed to the BASE unit, which sets the CBS/DCS switch to accept surface commands from either the CBS electronics or the DCS system. (At this point, although not shown, the triplex surface commands are cross channel monitored and the middle value surface command is selected for each channel.)

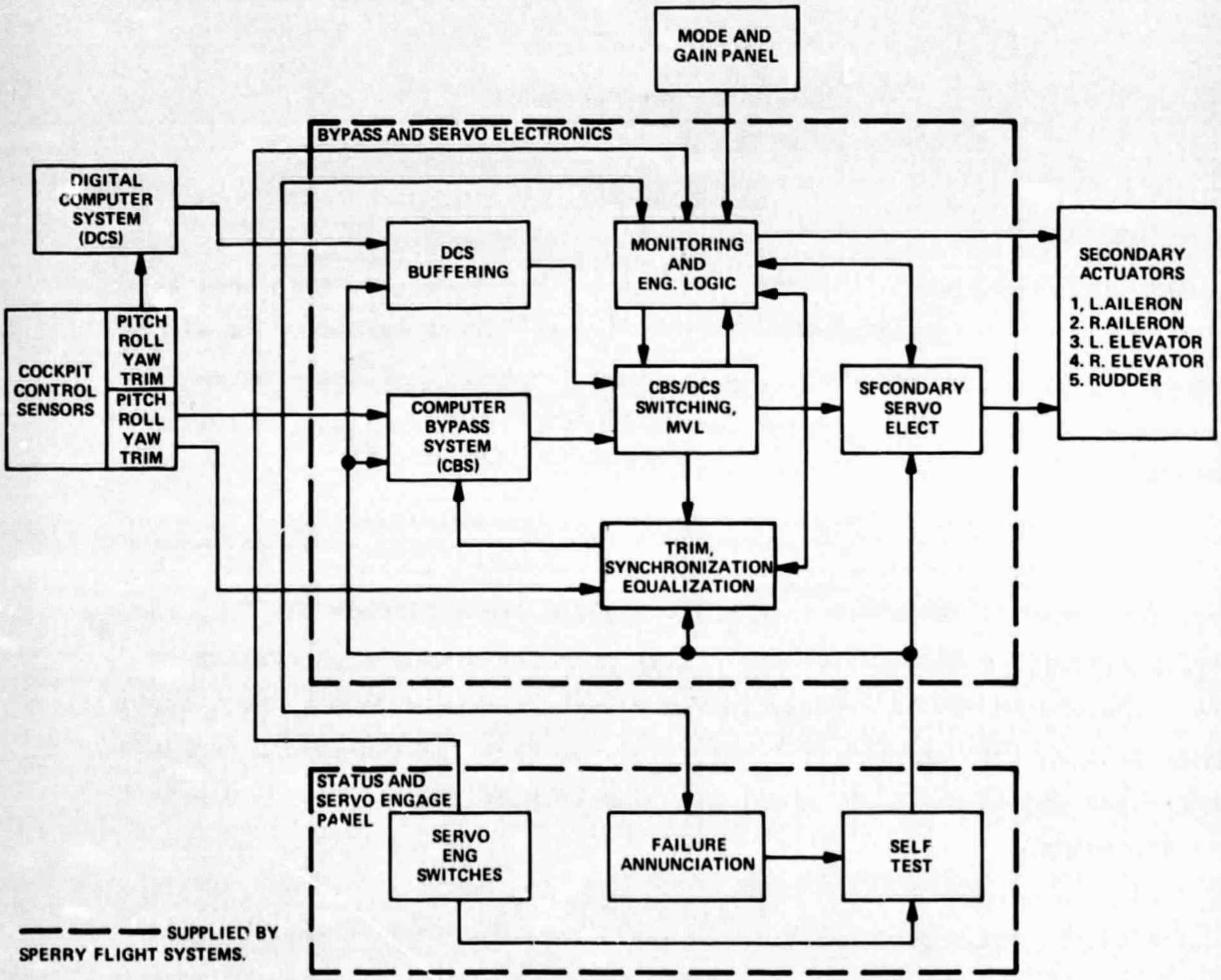


Figure 1. Simplified F-8 DFBW computer bypass and servo electronics system block diagram.

The surface commands from either the CBS or DCS system are then routed to the secondary servo electronics for surface positioning. Synchronization is provided to force the CBS surface commands to equal the DCS surface commands while the flight control system is in a DCS mode to reduce surface transients when downmoding from DCS to CBS. Triplex failure monitoring is provided in the BASE units, and detected failures are reported on the status/engage panel.

Self-test is an automated closed loop ground test which injects stimuli into strategic points in the BASE units and reads the desired results through the normal failure annunciation path.

In summary, the BASE units of the Sperry Flight Systems (SFS) -supplied equipment provide DCS surface command buffering, computer bypass system electronics, failure monitoring and engage logic, CBS/DCS switching and middle value logic selection, trim, synchronization, and equalization, and secondary servo electronics for both the DCS and CBS flight control modes. The status/engage panel of the SFS-supplied equipment provides servo engage switches, failure annunciation, and self-test.

Triplex operation is described in detail later in this report. For triplex operation, there are three DCS systems, three BASE units, and one status/engage panel. The single status/engage panel services all three BASE units. Separate aircraft sensors are provided for each triplex channel. Cross channel (between BASE units) referencing for triplex operation is performed for surface command middle value selection, failure monitoring, and secondary actuator pressure signal middle value selection.

Under this contract, Sperry Flight Systems delivered the following units to NASA: one status/engage panel breadboard, two status/engage panel flyable units, three bypass and servo electronic breadboards, three bypass and servo electronic flyable units, and one complete set of spare cards for the bypass and servo electronics flyable units.

SYMBOLS/ABBREVIATIONS LIST

$\overline{A}$ - Not A

ADI - Attitude Direction Indicator

AMP - Amplifier

ASSY - Assembly

AUTO - Automatic

BASE - Bypass and Servo Electronics

C - Comparator Status

CAS - Command Augmentation System

CBS - Computer Bypass System

CE - Control Electronics

CH - Channel

CLE - Comparator Status of Left Electronics

CM - Centimeter

CMD - Command

COMP - Computer

CPEM - Comparator Pre-Engage Monitor

CRE - Comparator Status of Right Electronics

db - Decibel

DCS - Digital Computer System

DEG - Degree

DFBW - Digital Fly-by-Wire

DIR - Direct Mode

DN - Down

E - Electronics

ENG - Engage

EQUAL - Equalization

F - Failure

FDBK - Feedback

FLT - Flight

G - Good

Hz - Frequency

IN/SEC - Inch per Second

INTEG - Integrate

L - Left

LRU - Line Replaceable Unit

LSB - Least Significant Bit

LVDT - Linear Voltage Differential Transmitter

ma - Milliamps

MS - Milliseconds

MSEC - Milliseconds

MVL - mid-value logic

N/A - Not Applicable

NASA - National Aeronautics and Space Administration

P - Pitch

PEM - Pre-Engage Monitor

SYMBOLS/ABBREVIATIONS LIST (CONT)

POT - Potentiometer

PSI - Pressure per Square Inch

R - Right

RAM - Random Access Memory

RRS - Roll Right Servo

S - Laplace Operator

S - Servo

SAS - Stability Augmentation System

SEC - Second

SFS - Sperry Flight Systems

TIP - Test Input Point

TMP - Test Monitor Point

V ac - Volts ac

V dc - Volts dc

VRMS - Volts Root Mean Square

Y - Yaw

P - Differential Pressure

δ_E, δ_A, δ_R - Elevator, Ailerons, Rudder

δ_{E_A}, δ_{E_B}, δ_{E_C} - Control Surface Commands to Surface Actuators

Detail descriptions of the system as a whole and system operation appear in later sections of this report. A description of the preflight self-test procedures is included as appendix A, and a comparator status check table during self-test is included in appendix B.

GENERAL SYSTEM DESCRIPTION

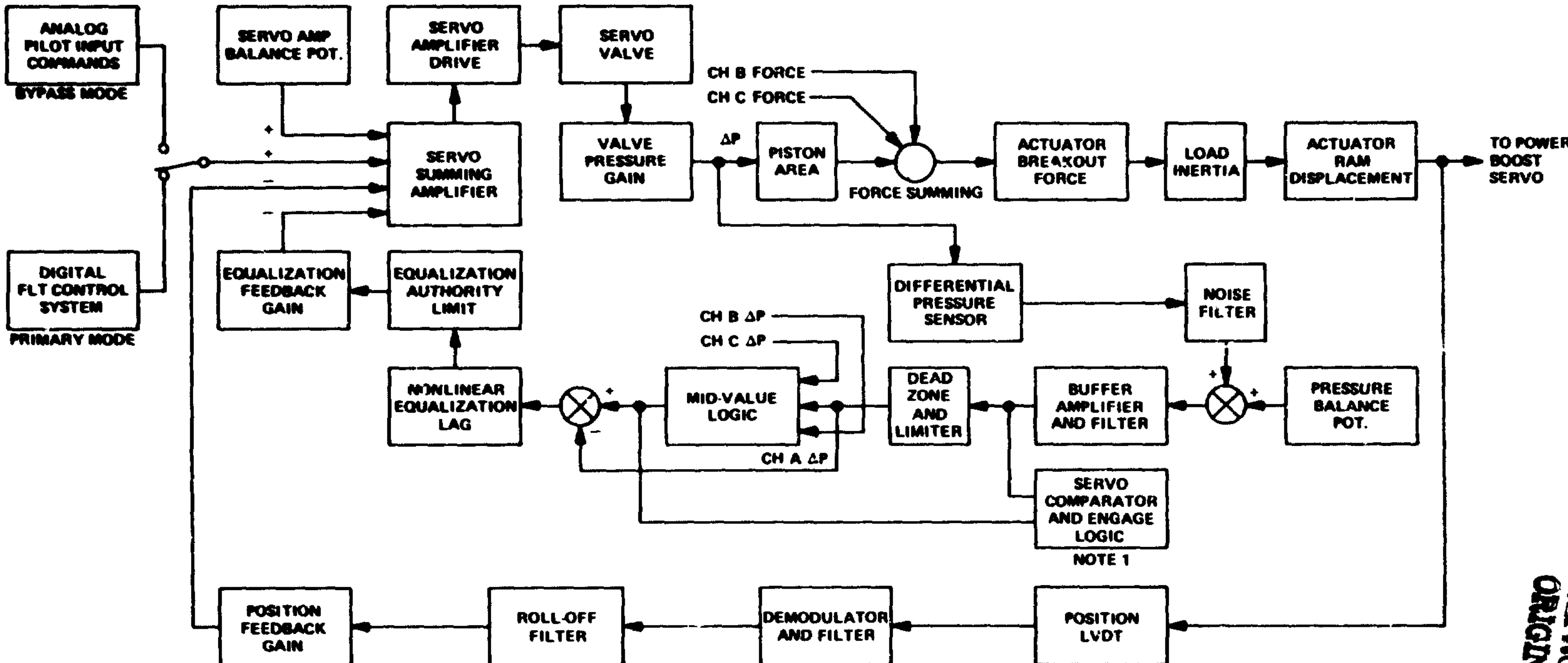
SERVO CONTROL SYSTEM

The F-8 secondary actuator is a triple force-summing tandem analog servo with both large output force capability and large system bandwidth. As such, it is capable of responding rapidly to input commands in the presence of any practical actuator operating load. The system redundancy features include triplex servo valves, electronics, position LVDT, and differential pressure sensors. The aircraft hydraulics system is triplex. Three dc buses and their associated inverters provide the necessary power for each servo channel. In the normal operating mode, these buses are fed by the flight control system dc generator via isolating diodes. Separate 28-volt batteries power each bus if generator power is lost.

The force output of each channel of a secondary actuator is developed across its respective piston. These pistons are cast in a single mechanical ram so that the forces produced by each piston are algebraically summed to drive the load. The inertial portion of the load consists of the sliding valve spool of the power boost servo and the associated mechanical linkages that connect the ram to the spool. Other opposing forces include actuator stiction, sliding friction effects, and Bernoulli forces on the main valve spool during load motion. If one of the servo channels fails, the monitoring system, upon sensing the failure, deactivates the appropriate engage solenoid. This results in the channels being hydraulically bypassed. The other two channels are then able to move the load without any opposition from the failed channel. Under two-channel operation, system performance remains essentially unchanged.

Five identical secondary actuators are associated with the F-8 fly-by-wire airplane. These include the left and right elevator actuators, the left and right flaperon actuators, and the rudder secondary actuator. Figure 2 is a functional block diagram of the typical closed loop system, but shows only a single channel of the triplex system. A diagram of the complete system would include a triplication of all blocks of figure 2 except those between the force summing point and the actuator output. A general description of the secondary actuator servo system design and operation is given in the following paragraphs.

Actuator servo input commands can originate from two sources. In the primary mode of operation the digital system provides these commands via its digital-to-analog conversion electronics. In the back-up mode, these commands originate from the sensors of the pilot cockpit controls. In either case, the command is routed to the servo amplifier via the mid-value logic (MVL), which selects the mid-value of the three command signals. One MVL command circuit is associated with each channel of the triplex actuator.



NOTE 1: SERVO CHANNEL DISENGAGED AND HYDRAULICALLY BYPASSED WHEN DIFFERENCE BETWEEN TWO COMPARATOR INPUTS EXCEED 1200 PSI (8.274×10^6 N/M^2) . . . FOR A MINIMUM DURATION OF 400 MILLISECONDS.

Figure 2. F-8 DFBW triplex secondary servo functional block diagram (only single channel shown).

The MVL input command at the servo amplifier is summed with the pressure and position feedback signals. The position loop is the dominant control loop that determines the basic dynamic response of the system. Differences between command and feedback signals are amplified in the servo summing amplifier and driver, and the resulting signal constitutes the driving signal to the servo valve. The latter is a two-stage electrohydraulic flapper valve. Initially, the servo amplifier error signal causes the valve to develop a differential pressure, ΔP, across the piston. This pressure, times the piston area, results in a force across the piston of each channel. These forces are summed in the ram, and after overcoming the effects of actuator stiction, the resultant force moves the inertial load. As the load moves, the inherent valve pressure to flow feedback characteristic reduces the pressure across the ram, while the linear voltage differential transmitter position signal cancels the MVL position command. In the steady state, when the ram comes to rest, the position and command signals cancel each other. For all intents and purposes this cancellation is exact, since the position feedback and command signals track each other to better than .1 percent under normal operating conditions.

As suggested in Figure 2, the position feedback signal from the LVDT is ac in nature. This signal is demodulated, and to reduce ripple to a tolerable level a two stage roll-off filter is provided. Additional high frequency filtering is included in the servo amplifier electronics to stabilize the design of this amplifier. The resulting closed loop dynamics of the servo approximates that of a second order system having a relative damping ratio of approximately .7 and a natural frequency of 20 Hertz.

In order to realize the full potential of a triplex force summing servo, pressure equalization feedback is introduced. The introduction of this feedback path serves to reduce force fights among the three channels to acceptable levels, thereby providing the capability to monitor the servo system. Without equalization, relatively small system nulls and tolerances would result in the generation of opposing forces across the individual pistons. While not affecting performance significantly, this could result in the effective loss of the means to detect faults in the system design, thereby compromising the "fail-operational/fail-passive" capability.

Figure 2 illustrates the pressure feedback system used in the F-8 secondary actuator design. The pressure across each piston is sensed by a dc transducer. The sensor's low level signal, 60 millivolts for 20.68×10^6 N/m^2 (3000 psi), is amplified and the high frequency noise filtered. The signal then passes through a deadzone and limiter circuit which is inserted primarily to offset the adverse effects of servo valve nonlinearities. The shaped signal is then routed to the MVL, where the mid-value among the three different channel pressures is selected as the reference for each channel. This mid-value pressure is compared with the individual channel pressure and the difference between the two signals is used as the pressure equalization feedback. The feedback in that channel with the mid-value pressure automatically has a zero feedback command at that moment. This is because the output voltage signal from the MVL circuit is identical to the input signal, for the case that the input is the mid value. Therefore, the difference, which constitutes the feedback, is zero. The signal representing the pressure difference is then operated upon by a nonlinear first-order lag circuit which shapes the feedback signal, and controls the pressure loop stability and bandwidth. The rate of signal build up at the output of this lag is limited whenever the signal magnitude exceeds a prescribed level. This assists in detecting passive or open system failures. The output signal of the nonlinear lag is limited to approximately 5 percent of servo position authority before being summed with the other servo amplifier commands. In order to provide system failure monitoring capability, individual channel pressures and the mid-value pressure are compared. When the difference between these two signals exceeds a specified level for some minimum time, the servo channel is disengaged and hydraulically bypassed. System operation then proceeds on a two channel basis.

Figure 2 shows servo amplifier and pressure feedback balance potentiometers, which are adjusted to null each channel when the system is initially turned on. In addition, the servo balance potentiometer provides sufficient range to null the system in the presence of any intentional unbalance effects. The latter can be introduced, as is the case for this system, by biasing the LVDT position feedback signal in order to more readily detect passive failures in the position feedback loop.

GENERAL SWITCHING CONFIGURATION

Servo Actuator Engagement

The F-8 DFBW is a triplex redundant system with servo actuators for the left aileron, right aileron, left elevator, right elevator, and rudder. These actuators have three channels each (A, B, and C which engage and control them) and are engaged through the status/engage panel located on the pilot's left hand console (fig. 3). There are 15 engage switches, with status lights located above each switch to inform the pilot which channel is engaged. Three positions are provided for each switch; auto, manual, and off. Engaging one of the three channels in manual inhibits and locks out the other two channels from engaging. An example of the switching sequence is provided in table 1. It should be noted that the engagement of the left elevator is independent from the engagement of the right elevator; this is also true for the left and right ailerons.

Manual engagement is provided in order to preflight each channel for independent, proper operation before every flight. Manual engagement overrides all surface command monitoring and servo actuator monitoring disable logic. Manual engagement also disables surface command MVL select, so that whatever surface command is injected into that channel becomes the position command for that actuator. This is different from auto engage, which compares the surface command from all three channels and selects the middle value as the position command for all three channels. See figure 4 for an example of this logic. Auto engage also provides full monitoring and disable logic.

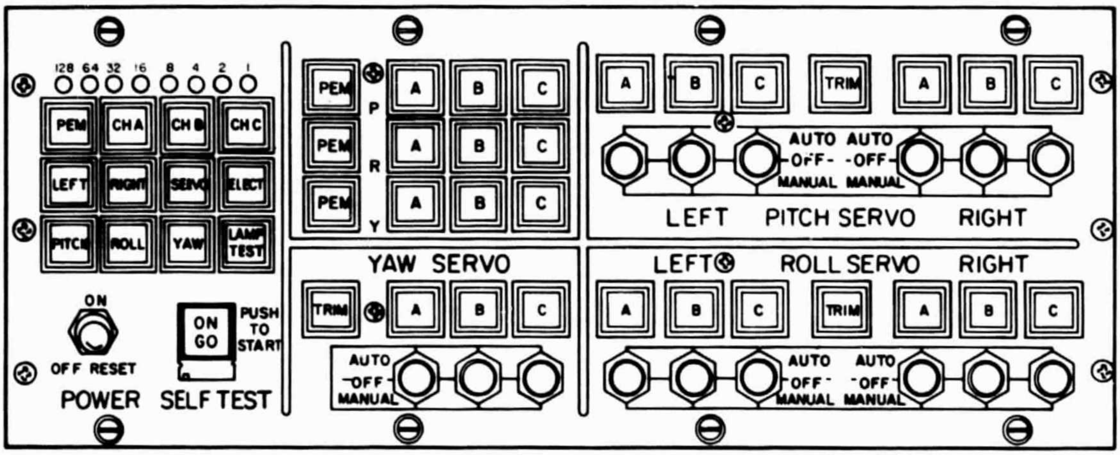


Figure 3. Status/engage panel.

TABLE 1. SERVO ACTUATOR SOLENOID ENGAGEMENT SWITCHING SEQUENCE EXAMPLE*

Channel A switch position	Channel B switch position	Channel C switch position	Channel A actuator	Channel B actuator	Channel C actuator
Off	Off	Off	Off	Off	Off
Auto	Off	Off	On	Off	Off
Auto	Auto	Off	On	On	Off
Auto	Auto	Auto	On	On	On
Manual	Auto	Auto	On	Off	Off
Manual	Manual	Auto	On	Off	Off
Manual	Manual	Manual	On	Off	Off

*Channel switch position sequence A-B-C.

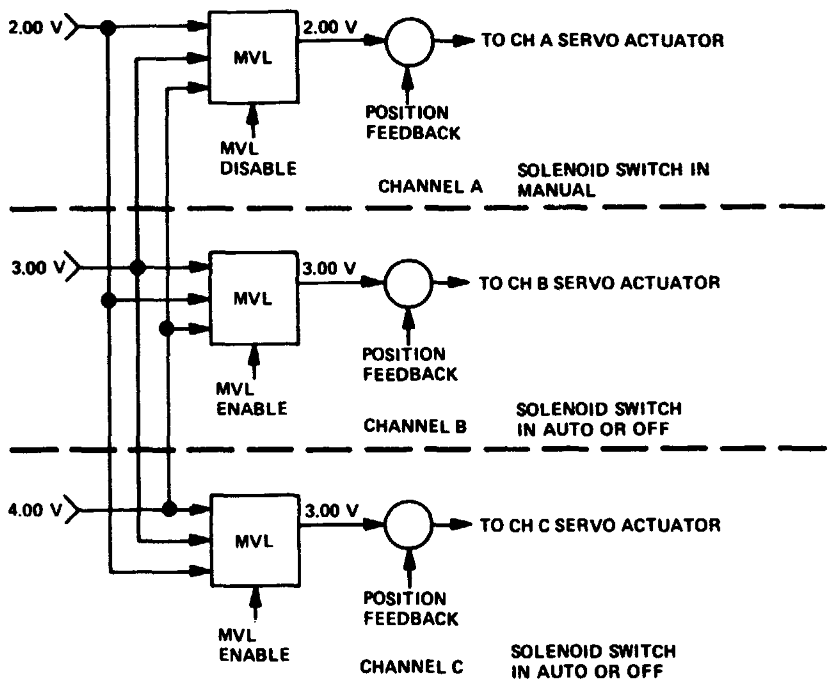


Figure 4. Manual engage MVL logic.

CBS or DCS Engage

The pilot has the option of selecting any of the digital control system modes (direct, SAS, or CAS) or the computer bypass system. The mode is selected on the mode and gain panel, which is located on the pilot's instrument panel directly above the Attitude Direction Indicator (ADI). Selecting either of the CBS or one of the DCS modes in one or all of the three aircraft axes (pitch, roll, and yaw) commands all channels (A, B, and C) to respond to that mode. For example, if the pilot selects pitch direct, roll CBS, and yaw CBS, all three bypass and servo electronics boxes (channels A, B, and C) respond to pitch commands from the three computers only. Roll and yaw commands from the computer are rejected, and only commands from the CBS are accepted. An interface block diagram of the elevator command from the stick to the surface actuator (aileron and rudder would be similar) is shown in figure 5.

COMPUTER BYPASS FUNCTION

The DCS is considered to be the primary system; the CBS provides the pilot with a backup system. Redundancy management of the three channels is handled in the bypass and servo electronics box for both the CBS and the DCS. While in the DCS mode, failure monitor logic can downmode an axis from DCS to CBS. However, failure monitor logic does not upmode an axis from CBS to DCS; it only shuts off a channel or actuator. Further discussion on failure monitor logic is discussed in SYSTEM OPERATION and SYSTEM DESIGN FEATURES.

SELF-TEST FUNCTION

The purpose of the preflight self-test is to verify the integrity of the triplex redundant BASE, the interface between the triplex redundant DCS and the triplex redundant CBS, and the triplex redundant secondary hydraulic servo actuator for the digital fly-by-wire flight control system.

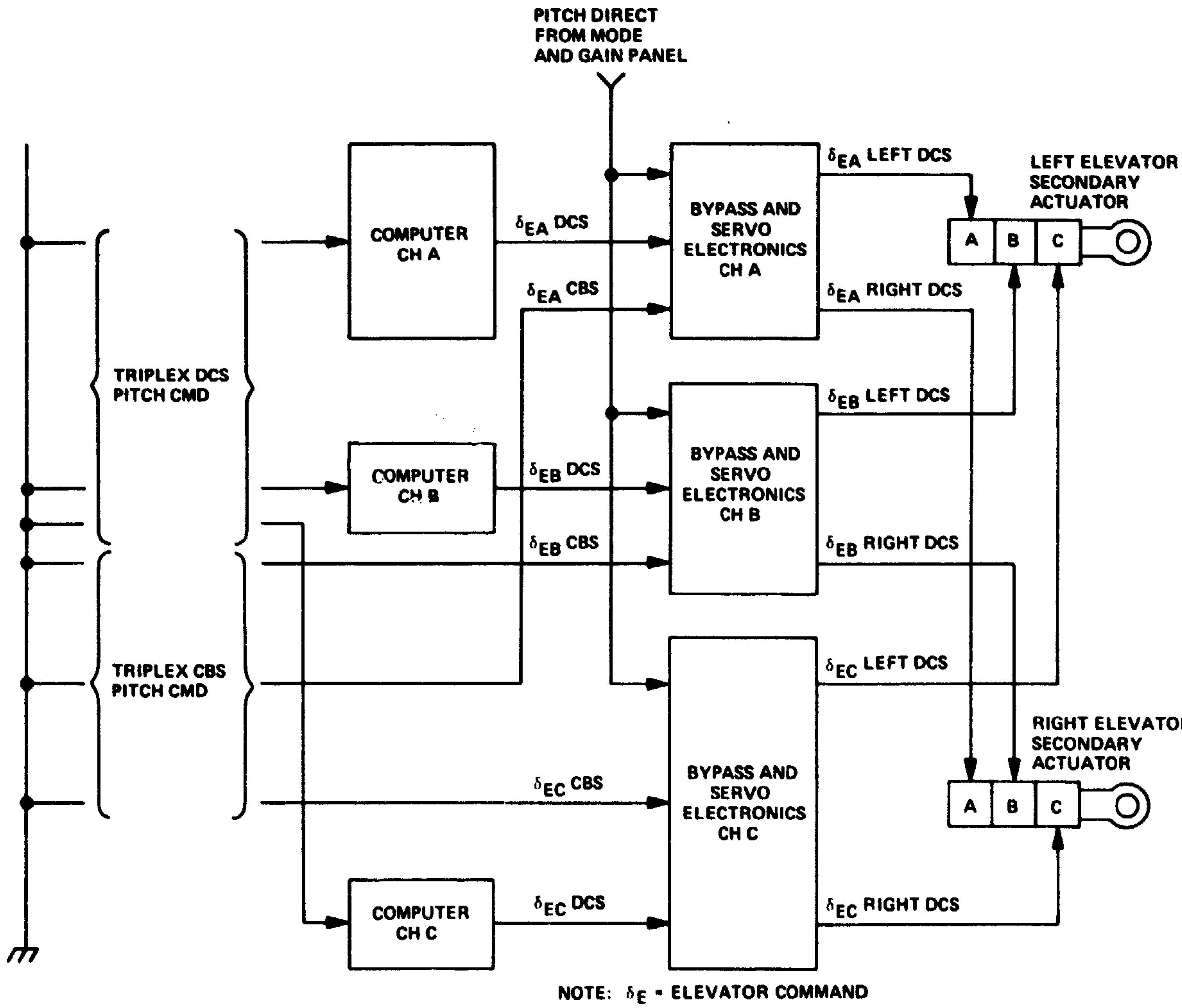


Figure 5. Triplex elevator command interface block diagram (DCS engaged).

SYSTEM OPERATION

DIGITAL SYSTEM INTERFACE

In addition to a computer valid discrete to the backup and servo control system, all three computers provide pitch, yaw, roll symmetric, and roll asymmetric surface commands. The backup and servo control system processes these signals, selects the middle value surface command for each axis, and drives all three channels of the secondary actuator in parallel. The backup and servo control system also provides the redundancy management monitoring to assure that each channel tracks within a specified window of the middle surface command value. Synchronization is provided to ensure that the CBS always has a command equal in value to the DCS middle value surface command, so that if an axis is downmoded from DCS to CBS, there is a minimum of surface transient.

Digital System Engagement

Upon initial power on, the backup and servo control system is reset to the CBS mode. Engagement in a DCS mode requires several criteria to be met. The sequence of operation would be as follows: The pilot pushes the DIR button on the mode and gain panel for the chosen axis. This results in a digital engage request to the backup and servo control system. The system then starts its search for two computer valids and a digital engage latch. (The electronic comparators need not be valid at this time.) If there is a digital engage latch signal and two computers are valid, two parallel operations occur. The switch which controls whether a CBS or a DCS surface command is to be accepted is set to the DCS position, provided that the electronic comparator for the channel is valid and that the computer for that channel is valid. The other operation is that one shot is triggered which enables the MVL (triplex voter) and attempts to reset the electronic comparators for that axis. If the reset is successful, and either two left or two right electronic comparators for that axis report a valid state, the system latches up into the DCS mode and the DIR button can be released. The entire sequence takes 15 milliseconds or less, depending on whether the electronic comparators are valid prior to upmoding, since there is a 15-millisecond delay prior to resetting the electronic comparators.

As previously mentioned, the latch signal is disabled either by pushing the CBS button on the mode and gain panel for that axis, or by depressing the paddle switch on the pilot's control stick, which downmodes all axes to the CBS mode.

The backup and servo control system searches for two computer valids and two electronic comparator valids; this is the underlying principle behind the triplex redundancy monitoring system (the two-out-of-three-valid principle). Since middle value logic is used for the surface command, it takes two valid commands to give a reliable middle value command. Therefore, if two computers report that they are invalid (their commands are unreliable), or if two electronic comparators detect two surface commands which are outside the specified window of the middle value command (the correct command is undeterminable), then the backup and servo control system does not allow an upmode since the commands are unreliable.

The backup and servo control system reports to both the mode and gain panel and to the computers of a DCS mode engagement.

Digital Computer System Surface Drive Commands

<u>Yaw digital computer system command</u>. - In reference to figure 6, the yaw DCS command is delivered from the three computers to their respective channels in the backup and servo control system. This signal is processed through a second-order, low-pass filter with a break frequency of 50 radians per second, and then scaled up by a gain of 2 and command limited to ±20.7 degrees of rudder surface deflection. The signal is then delivered to the CBS/DCS solid state switch which controls whether or not this signal can pass through to the middle value logic block. From this point the signal is fed to the middle value logic block in each of the three channels for

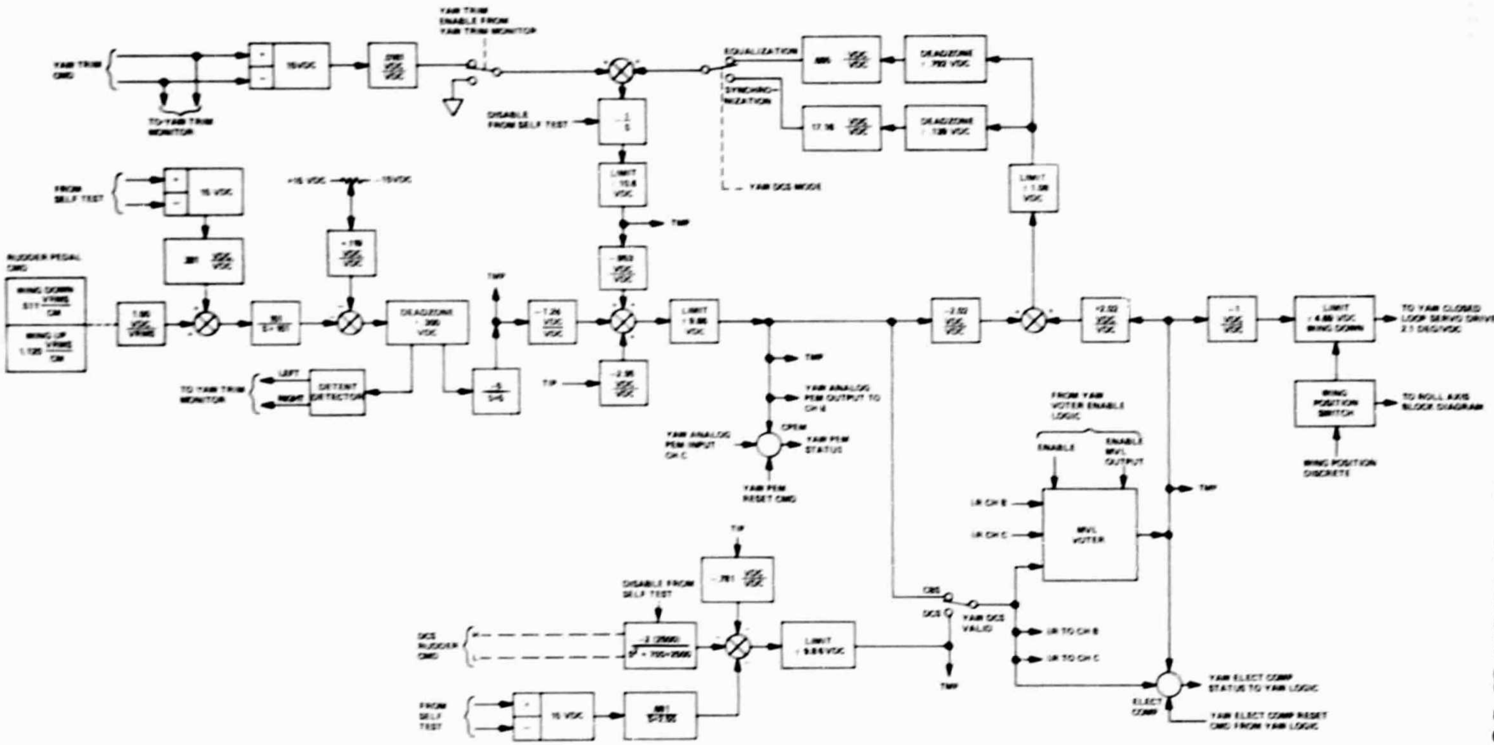


Figure 6. Yaw axis CBS/DCS analog block diagram - channel A.

comparison and middle value selection. The middle command value is then limited to ±10.0 degrees of rudder deflection if the wing is down. A discrete signal is provided by one of the three independent wing position switches located under the wing. If the wing is up, the signal is not limited and the entire signal is passed through to the closed loop drive which is scaled at 2.10 degrees per volt. This mathematical equation is:

$$\frac{\delta}{V} = \left(\frac{2500}{S^2 + 70S + 2500}\right)\left(-2.00\ \frac{V\ dc}{V\ dc}\right)\left(-1.0\ \frac{V\ dc}{V\ dc}\right)\left(2.1\ \frac{\text{Degrees}}{V\ dc}\right)$$

<u>Pitch digital computer surface command</u>. - In reference to figure 7, the pitch DCS command is received in each channel and processed through a second-order, low-pass filter with a break frequency of 50 radians per second, and then scaled up by a gain of 2 and command limited to 24.8 degrees of elevator leading edge down, and 5.57 degrees of leading edge up travel. The signal then takes two parallel paths; one path is for the left elevator actuator and one is for the right actuator. Each parallel path is first delivered to its respective CBS/DCS switch. This switch, which is controlled by logic, determines whether or not the DCS signal passes through to the middle value logic block. From this point the signal is fed to the middle value logic block in each channel for comparison and middle value selection. The middle value signal is then used as the position command for the closed loop servo drive, which has an approximate gain of 2.40 degrees per volt. This mathematical equation is:

$$\frac{\delta}{V} = \left(\frac{2500}{S^2 + 70S + 2500}\right)\left(-2.00\ \frac{V\ dc}{V\ dc}\right) \times \left(2.40\ \frac{\text{Degrees}}{V\ dc}\right)$$

Further gearing and signal shaping between the pilot's control stick and the backup and servo control system is provided by the computer in software. An elevator position signal for each channel is fed back to its respective computer.

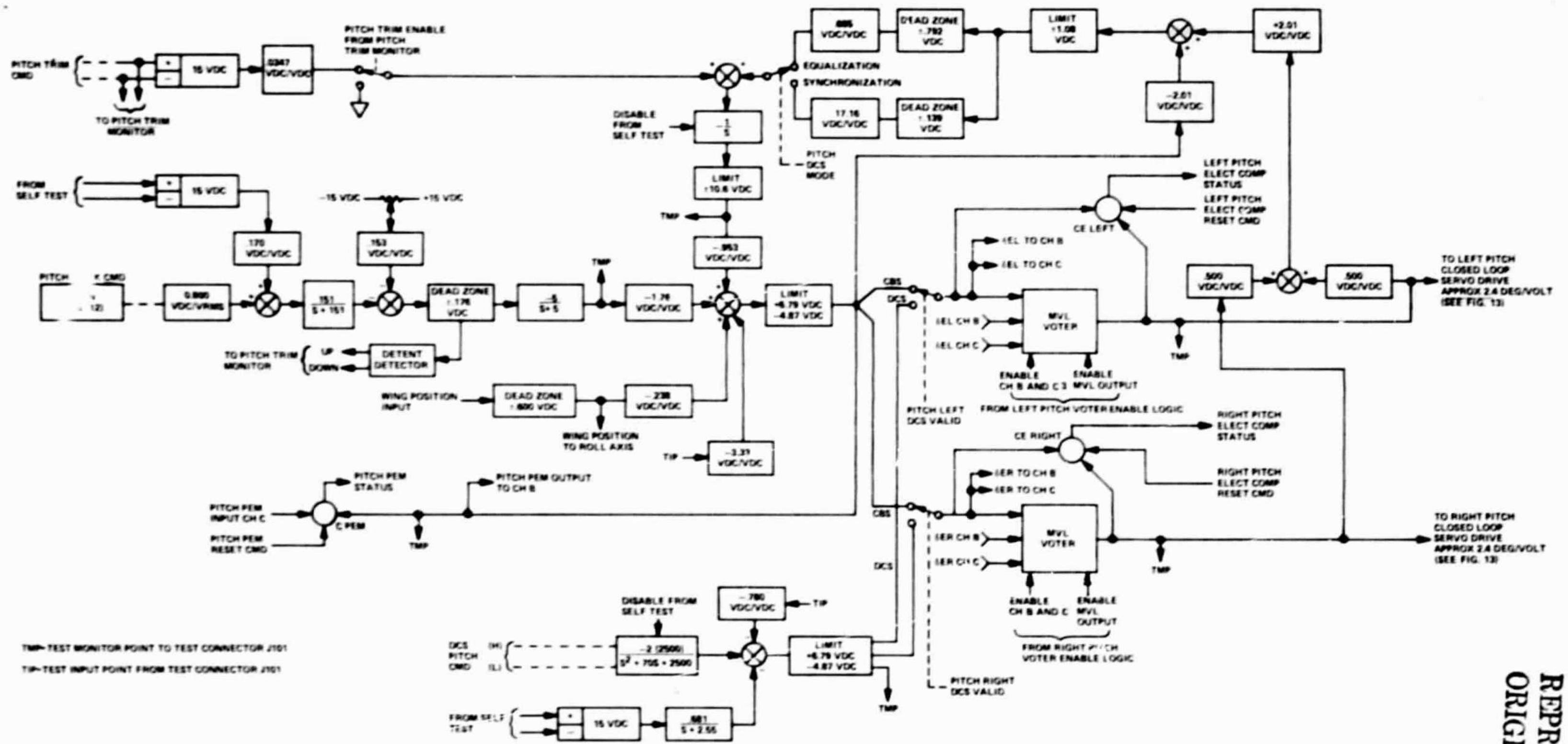


Figure 7. Pitch axis CBS/DCS analog block diagram - channel A.

Roll digital computer system command. - The three computers each supply a DCS symmetric aileron command and a DCS asymmetric aileron command to the backup and servo control system (figure 8). The DCS symmetric aileron command is normally used as a wing flap command to drive both aileron surface to 20 degrees trailing edge down for a wing up condition. However, this is under software control, and therefore is not controlled by the backup and servo control system. Both aileron commands are processed through a second-order, low- pass filter with a break frequency of 50 radians per second and scaled up by a gain of 2. The DCS symmetric aileron command is then added to the DCS asymmetric aileron command for the right hand actuator signal, and subtracted from the DCS asymmetric aileron command for the left hand actuator signal. These composite signals are then limited to 13.9 degrees of aileron trailing edge up travel and 42.9 degrees of trailing edge down travel. These signals are then delivered to their respective CBS/DCS switches, which control whether or not they pass through to the MVL block. From the switch, the signal is fed to the MVL block in all three channels for monitoring and middle value selection. The middle value command then drives the surfaces with a scale factor of 4.5 degrees per volt. The mathematical equation from input to surface is:

$$\frac{\delta}{V} = \left(\frac{2500}{S^2 + 70S + 2500}\right)\left(-2.00\ \frac{V\ dc}{V\ dc}\right)\left(-1.00\ \frac{V\ dc}{V\ dc}\right)\left(4.5\ \frac{\text{Degrees}}{V\ dc}\right)$$

The DCS symmetric aileron command is subtracted from the synchronization circuit and added directly to the CBS circuit in the proper manner for left and right aileron signals. When a downmode from DCS to CBS occurs, the DCS symmetric aileron command is gradually faded out and the wing position command used in the CBS mode is smoothly faded in with the same time constant of 1.0 second. This fadeout occurs for either a downmode, or if both CBS/DCS switches for that channel are switched to the CBS mode.

Further gearing and signal shaping between the pilot's control stick and the backup and servo control system is accomplished by the computer under software control.

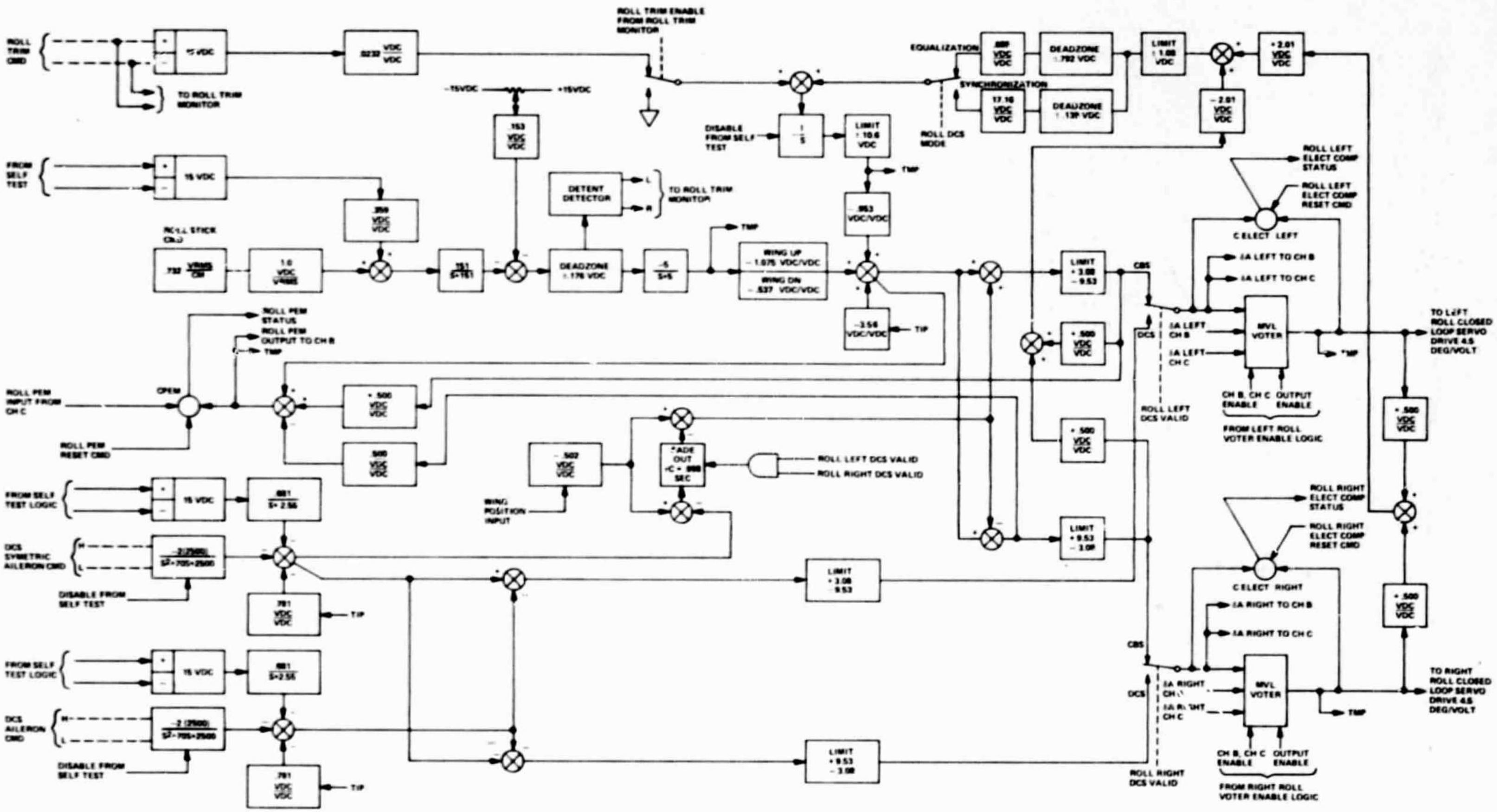


Figure 8. Roll axis CBS/DCS analog block diagram - channel A.

Digital Computer System Redundancy Management Monitoring

Redundancy management monitoring for a DCS mode consists of monitoring computer valid discretes, DCS surface commands, and CBS pre-engage monitoring. Figures 9, 10, and 11 are the logic functional diagrams for the pitch, roll, and yaw axes, respectively.

Digital computer system surface command monitoring. - Electronic comparators are used to compare a DCS channel command with the DCS middle value surface command. This comparator is set to trip to an invalid state if the surface command falls outside the specified window of ±5.96 degrees (2.84 V dc) of surface command for the yaw axis, ±6.82 degrees (2.84 V dc) of surface command for the pitch axis, and ±10.94 degrees (2.43 V dc) of surface command for the roll axis. Different logical sequences of events occur, depending upon the number of comparators that are tripped. Examples of possible sequences follow.

One left and/or one right electronic comparator trip: If a failure is reported by one left and/or one right electronic comparator in the pitch or roll axis, or one electronic comparator in the yaw axis, the CBS/DCS switch for that side in that channel switches to the CBS position, and synchronization forces that input to equal the DCS middle surface command value. Synchronization tracks the middle surface command value to ±.069 V dc at a maximum rate of 15.4 V dc per second. The appropriate electronic warning light on the status/engage panel lights up to alert the pilot of a tripped electronic comparator. The pilot may then attempt to reset this comparator by pressing the illuminated electronic warning light, which is also a momentary contact switch. If the reset is successful, the light goes out and the CBS/ DCS switch returns to the DCS position.

Two left or two right electronic comparator trips: If a failure is detected in two left or two right (but not both left and right) electronic comparators in the pitch or roll axis, the MVL output for that side in that axis is disabled for all three channels. The surfaces then automatically

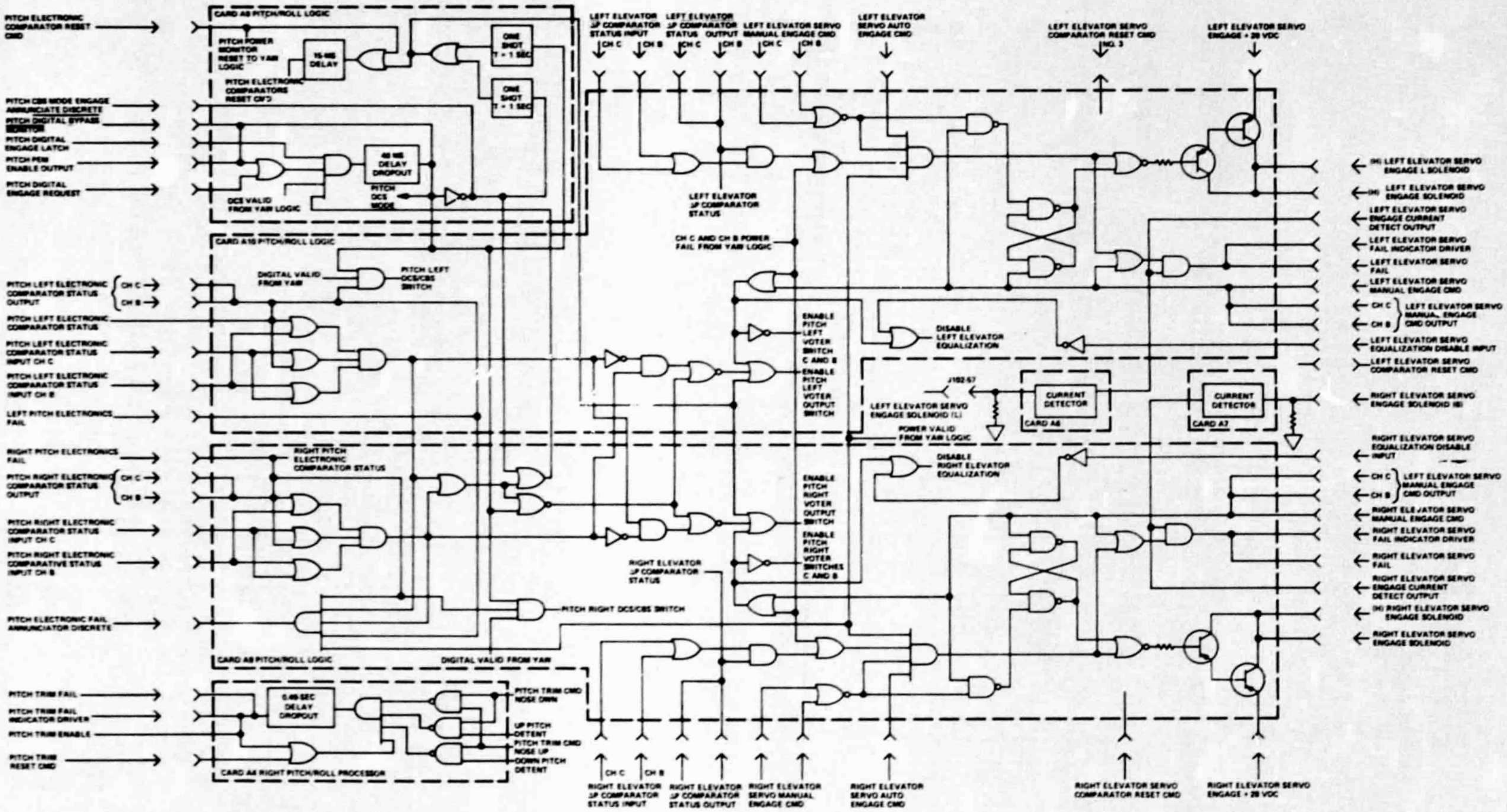


Figure 9. Pitch axis functional logic diagram.

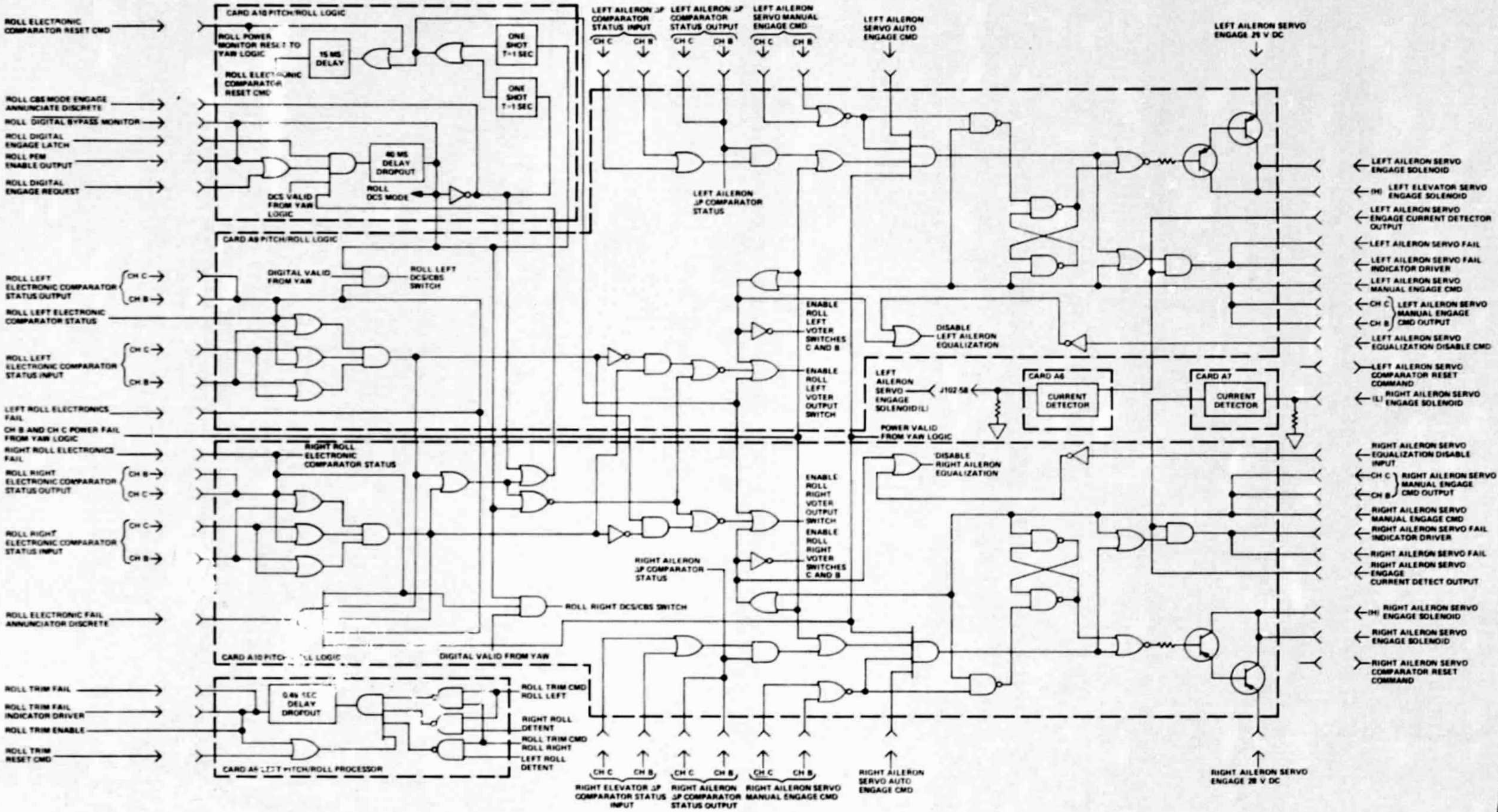


Figure 10. Roll axis functional logic diagram.

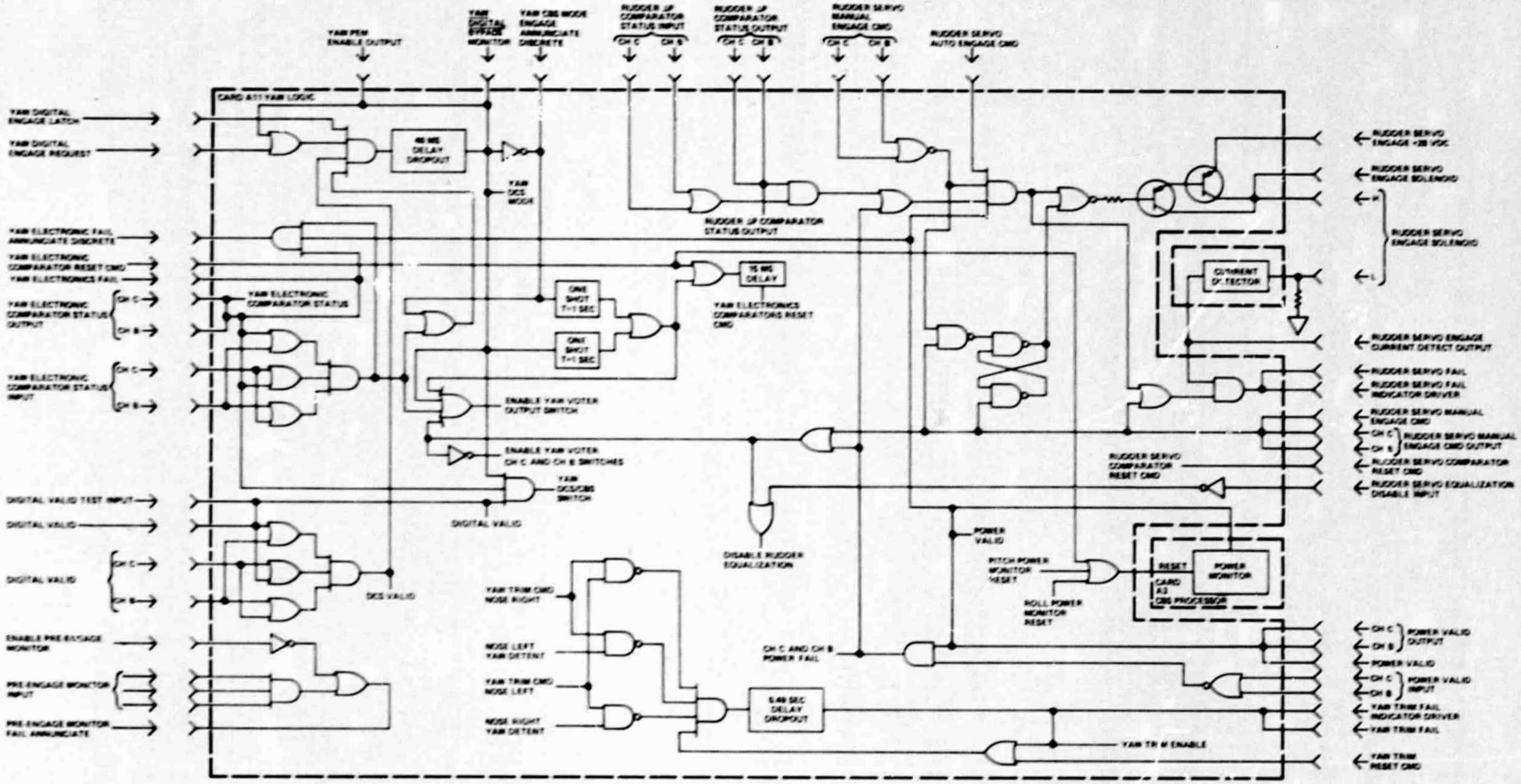


Figure 11. Yaw axis functional logic diagram.

return to a 0 command surface deflection for the aileron surface, or 5 degrees leading edge down for the elevator axis if the secondary actuator is still engaged (this refers to zero surface command). All three electronic warning lights for that axis on the status/engage panel light up. The pilot may then attempt to reset the electronic comparators. If the reset is successful and two or three of the electronic comparators become valid, the MVL again becomes operative and the switches are set in the appropriate positions depending upon whether or not their electronic comparators are valid. If the failure still exists, no reset is possible.

Two left and two right electronic comparators trip: If two left and two right electronic comparators in the pitch or roll axis, or two electronic comparators in the yaw axis trip, that axis is automatically downmoded to the CBS mode and all the electronic comparators in that axis automatically receive a reset command. The pilot is informed of this action on the mode and gain panel when the CBS mode light comes on and the DCS mode (DIR, SAS, CAS) light goes off.

Computer valid discrete monitoring. - As mentioned previously, if one computer is or becomes invalid, all of the CBS/DCS switches for that channel switch to the CBS mode. If two computers fail, all axes automatically downmode to the CBS mode. However, other failures can occur while remaining in the DCS mode, such as one computer failure and any number of electronic comparator failures up to the number required to downmode the system to the CBS mode. The examples following illustrate this situation.

One left or one right electronic comparator trips and one computer fails: If one left or one right (but not both) electronic comparator in the pitch or roll axis trips, and one computer from another channel fails, the appropriate CBS/DCS switches switch to the CBS position and synchronization forces that side to equal the MVL output from the other side (within the boundaries of the electrical limits). The appropriate electronic light on the status/engage panel will light up and the pilot may attempt to reset that electronic comparator.

One left and one right electronic comparator trips and one computer fails: If one left and one right electronic comparator in the pitch or roll axis, or one electronic comparator in the yaw axis trips, and one computer from another channel fails, the appropriate CBS/DCS switches will switch to the CBS position and the appropriate electronic lights on the status/engage panel will light up. This situation is unusual because two left and two right CBS/DCS switches are in the CBS position and therefore the system is controlled by the CBS mode even though a downmode has not occurred. The DCS mode light on the mode and gain panel remains illuminated. This, however, causes no problem, because as soon as the valid computer channels disagree with the backup system, the electronic comparators for that channel trip and properly downmode the system to the CBS mode.

Backup system monitoring while in digital computer system mode. - Monitoring is provided in the DCS mode to alert the pilot of a failure in the backup system. This is cross channel monitoring between the CBS commands in which channel A compares with channel B, channel B compares with channel C, and channel C compares with channel A. This channel signal is a composite of CBS rudder pedal command and synchronization in the yaw axis; CBS pitch stick command, synchronization, and wing position in the pitch axis; and CBS roll stick command, synchronization, DCS symmetric aileron command, and wing position in the roll axis. If the channels disagree by more than ±5.96 degrees (2.84 V dc) of surface command in the yaw axis, ±6.82 degrees (2.84 V dc) of surface command for the pitch axis, or ±12.78 degrees (2.84 V dc) of surface command for the roll axis, the PEM light on the status/engage panel will light up, alerting the pilot of a failure in the backup system for that axis. The pilot may attempt to reset this comparator by pressing on the illuminated light. This monitoring is only meant to be an alert to the pilot, and no disable logic is associated with it. The PEM alert warning circuitry is disabled while in the CBS mode.

COMPUTER BYPASS OPERATION

The CBS is provided to give the pilot a reliable backup system in case of DCS malfunction or pilot desire. The CBS receives inputs from the pilot's control stick, rudder pedals, manual trim commands, and wing position analog and discrete signals. These signals are combined and shaped in the appropriate manner for each axis. The middle value is then selected for each axis from the three channels and used to drive the three channels of the secondary actuator in parallel. Redundancy monitoring is provided to assure that each channel is tracking within a specified window of the middle value command. Equalization is provided to null any offset errors between channels.

Yaw Computer Bypass System Surface Command

Three LVDT signals controlled by the rudder pedals are delivered to the appropriate yaw channels in the backup and servo control system (figure 6). This signal has two scale factors: .511 volt rms per centimeter if the wing is down, and 1.120 volts rms per centimeter if the wing is up. The backup and servo control system demodulates this signal with a scale factor of 1.00 volt dc per volt rms with a first-order lag effect which has a break frequency of 151 radians per second. This signal is then nulled out to a 0 degree rudder command for zero rudder pedal deflection and then subjected to a deadzone of ±.300 volt dc. From here the signal is shaped with a first-order lag circuit with a break frequency of 5.0 radians per second. This signal is then scaled up by a gain of 1.260 volts dc per volt dc and summed with the trim signal and equalization circuit. This composite signal is then command limited to ±20.7 degrees of rudder surface deflection. From here this signal is compared with the other two channels for monitoring purposes and middle value selection. The middle command value is then limited to ±10.0 degrees of rudder deflection if the wing is down. A discrete signal is provided by one of the three independent wing position switches located under the wing. If the wing is up, the signal is not limited and the entire signal is passed through to the closed loop servo drive, which is scaled at 2.1 degrees of rudder deflection per volt. The linearized mathematical equation from the rudder pedal to the rudder when the wing is down is as follows:

$$\frac{\delta_R}{\text{Pedal}} = \left(.511 \frac{\text{Vrms}}{\text{cm}}\right)\left(1.00 \frac{\text{V dc}}{\text{Vrms}}\right)\left(\frac{151}{S + 151}\right)\left(\frac{5}{S + 5}\right)\left(1.26 \frac{\text{V dc}}{\text{V dc}}\right)\left(2.1 \frac{\text{Deg}}{\text{V dc}}\right)$$

When the wing is up the following equation is used:

$$\frac{\delta_R}{\text{Pedal}} = \left(1.120 \frac{\text{Vrms}}{\text{cm}}\right)\left(1.00 \frac{\text{V dc}}{\text{Vrms}}\right)\left(\frac{151}{S + 151}\right)\left(\frac{5}{S + 5}\right)\left(1.26 \frac{\text{V dc}}{\text{V dc}}\right)\left(2.1 \frac{\text{Deg}}{\text{V dc}}\right)$$

The pilot can manually trim the rudder with full surface authority at a rate of .484 degree (.231 V dc) per second. Equalization is provided to null out offsets between channels. This signal has a maximum rate of .396 degree (.189 V dc) per second and nulls out any steady state error between that channel and middle value command greater than .827 degree (.393 V dc). These voltages are measured at the point where the yaw closed loop servo receives the wing down limited signal (figure 6).

Pitch Computer Bypass System Surface Command

Three LVDT signals controlled by the pilot's control stick are delivered to the appropriate pitch channels in the backup and servo control system (figure 7). This signal has a nonlinear parabolic shape (figure 12). The backup and servo control system demodulates this signal with a scale factor of .800 volt dc per volt rms with a first-order lag effect which has a break frequency of 151 radians per second. This signal is then nulled to a 0 degree elevator command for a zero pitch stick deflection and then subjected to a deadzone of ±.176 volt dc. From here the signal is shaped with a first-order lag filter with a break frequency of 5.0 radians per second. This signal is then scaled up by a gain factor of 1.76 volts dc per volt dc, and summed with the trim command, equalization circuitry, and wing position input. The wing position input is scaled so that the elevators are offset

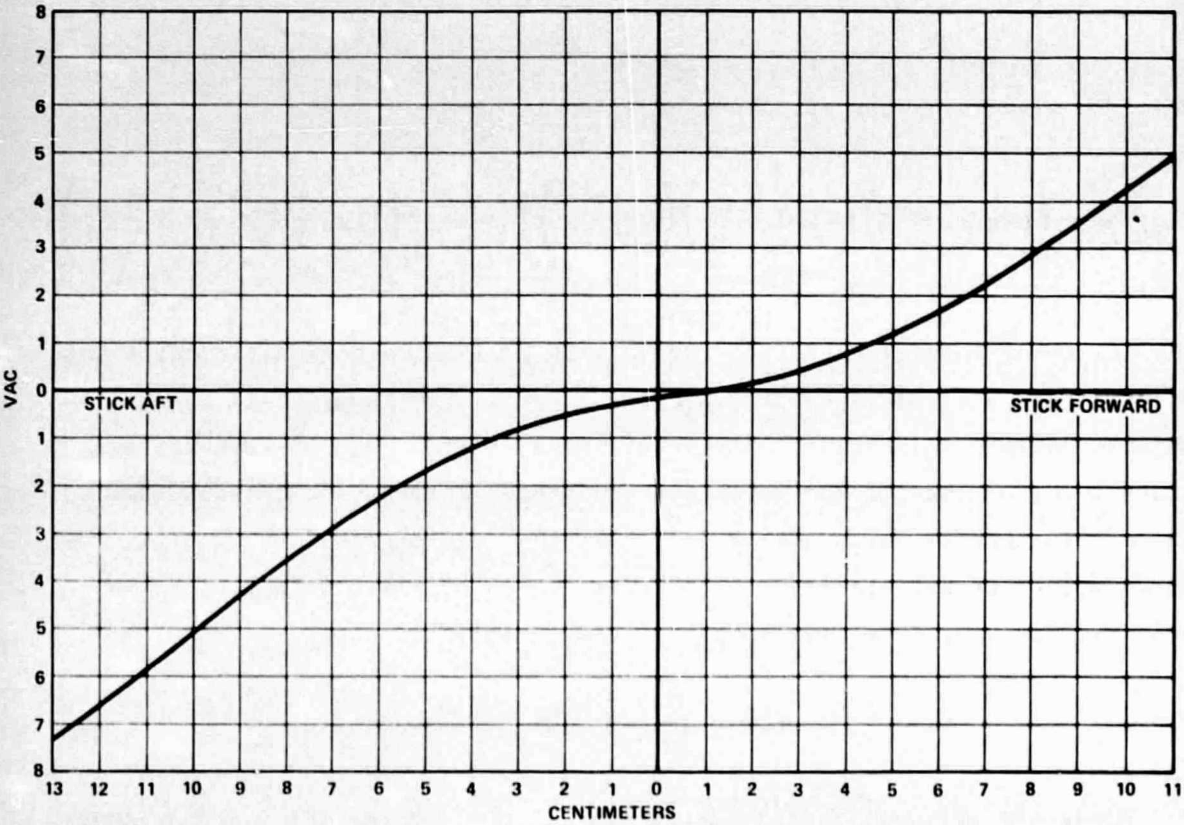


Figure 12. Pitch stick command in V ac versus pitch stick displacement in channel A.

from 5 degrees leading edge down for a wing down condition, and to 0 degrees for a wing up condition. This composite signal is then command limited to 24.8 degrees of elevator leading edge down, or to 5.57 degrees of leading edge up travel. From here the signal is divided into two parallel paths for left and right elevator commands. These signals are compared with the signals from the other two channels for monitoring purposes and middle value selection. The middle value then drives the surfaces at approximately 2.40 degrees per volt (figure 13). The linearized CBS scaling from control stick to elevator is as follows:

$$\frac{\delta_E}{\text{Pitch Stick}} = \left(\frac{\text{Vrms}}{\text{cm}}\right)\left(\frac{.800\ \text{V dc}}{\text{Vrms}}\right)\left(\frac{151}{S + 151}\right)\left(\frac{5}{S + 5}\right)\left(1.76\ \frac{\text{V dc}}{\text{V dc}}\right)\left(2.40\ \frac{\text{Deg}}{\text{V dc}}\right)$$

*Figure 12 gain gradient

The pilot can manually trim the elevator with full surface authority at a rate of 1.19 degrees (.497 V dc) per second. Equalization is provided to null offset errors between channels. This signal has a maximum rate of .454 degree (.189 V dc) per second and will null out any offset errors greater than .943 degree (.393 V dc). The voltages are measured at the input to the pitch closed loop servo drive (figure 7).

Roll Computer Bypass System Surface Command

Three LVDT signals controlled by the pilot's control stick are delivered to the appropriate roll channels in the backup and servo control system (figure 8). This signal has a scale factor of .732 volt rms per centimeter. The backup and servo control system demodulates this signal with a scale factor of 1.00 volt dc per volt rms and also has a first-order lag effect with a break frequency of 151 radians per second. This signal is then nulled out to a 0 degree surface command for zero roll stick deflection and then subjected to a deadzone of ±.176 volt dc. The signal is then shaped with a first- order lag circuit with a break frequency of 5.0 radians per second. This signal is then scaled by one of two scale factors, depending upon whether the wing is up or down. This scale factor is .537 volt dc per

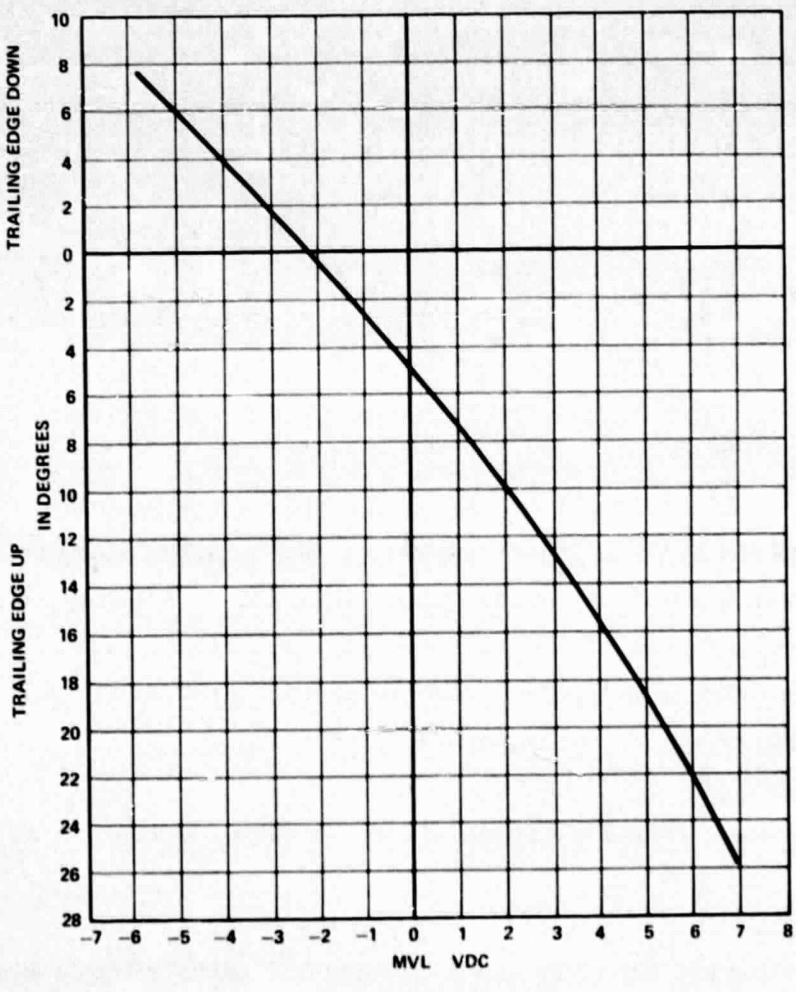


Figure 13. Elevator surface command in V dc versus surface position data taken channel A δ_{ER}.

volt dc for a wing down condition and 1.075 volts dc per volt dc for a wing up condition. This signal is then summed with the trim command and equalization circuit. The signal is then split into two parallel paths, one for the left aileron actuator and one for the right. Wing position is added to the left hand signal and subtracted from the right. The wing position signal is scaled so that when the wing is up, both aileron surfaces droop 20 degrees from an initial 0 degrees for a wing down condition. This composite signal is then command limited to 13.9 degrees of trailing edge up, or to 42.9 degrees of trailing edge down travel. This signal is then used with the signal from the other two channels for monitoring purposes and for middle value selection. The middle value then drives the secondary actuator with a scale factor of 4.5 degrees per volt dc. With the wing up, the linearized mathematical equation from the pilot's control stick to aileron surface is as follows:

$$\frac{\delta_A}{\text{Roll Stick}} = \left(.732\ \frac{\text{Vrms}}{\text{cm}}\right)\left(1.00\ \frac{\text{V dc}}{\text{Vrms}}\right)\left(\frac{151}{S + 151}\right)\left(\frac{5}{S + 5}\right)\left(.537\ \frac{\text{V dc}}{\text{V dc}}\right)\left(4.5\ \frac{\text{Deg}}{\text{V dc}}\right)$$

With the wing up, the equation is as follows:

$$\frac{\delta_A}{\text{Roll Stick}} = \left(.732\ \frac{\text{Vrms}}{\text{cm}}\right)\left(1.00\ \frac{\text{V dc}}{\text{Vrms}}\right)\left(\frac{151}{S + 151}\right)\left(\frac{5}{S + 5}\right)\left(1.075\ \frac{\text{V dc}}{\text{V dc}}\right)\left(4.5\ \frac{\text{Deg}}{\text{V dc}}\right)$$

The pilot can manually trim the aileron surfaces with full surface authority at a rate of 1.49 degrees (.332 V dc) per second. Equalization is provided to null out offsets between channels. This signal has a maximum rate of .851 degree (.189 V dc) per second, and will null out any steady state error between that channel and middle value command greater than 1.77 degrees (.393 V dc). The voltages are measured at the input to the roll closed loop servo drive.

Computer Bypass System Redundancy Management

CBS surface command monitoring is similar to DCS surface command monitoring in that electronic comparators (with identical trip levels) are used to detect mismatches between channel command and middle value command. However, since the CBS system is the backup system, no downmoding is provided. Therefore, if two or three electronic comparators for a particular command path (left pitch, right pitch, left roll, right roll, or yaw) detect failures, that path is disabled. For example, if one left pitch electronic comparator trips, two good channels remain, and the only action is that the electronic warning light for that channel lights up. However, if two left pitch electronic comparators trip, the correct surface command is undeterminable. Then all three of the left pitch MVL outputs are disabled and all of the pitch electronic warning lights light up. Disabling the output of the MVL would cause the left elevator to return to 5 degrees leading edge down and remain there as long as the actuator is engaged. (Disabling a roll MVL or yaw MVL would cause the surface to return to 0 degrees.)

Trim runaway protection is also a feature of the CBS. If for some reason a false trim command between the trim switch and the backup and servo control system occurs, the pilot simply pushes the control stick or rudder pedal (whichever axis the false trim command is coming from) far enough to overcome the deadzone (±.176 volt dc for the pitch and roll axis, ±.300 volt dc for the yaw axis), and the trim logic circuitry automatically disables the trim circuit for that axis. The trim warning light on the status/engage panel for that axis lights up, indicating to the pilot that the trim circuit is disengaged. The pilot has the option of resetting this circuit by pressing the trim warning light.

SERVO REDUNDANCY MANAGEMENT AND ENGAGEMENT INTERLOCKS

As mentioned in the section entitled GENERAL SWITCHING CONFIGURATION, the five actuators are engaged through the switches located on the status/engage panel. The following discussion covers the criteria that must be met for a channel to be engaged and stay engaged. It should be noted that both manual and auto engagement are independent of the CBS or DCS mode.

Auto Engagement

Certain criteria must be met before a channel can be put into the auto mode. The criteria and sequence of events that occur after the servo engage switch is placed in auto are as follows: the backup and servo control system for that channel checks to see that at least one servo comparator from one of the other two channels is valid and that its own servo comparator is valid. Simultaneously, it checks the validity of its own power monitor and makes sure that a manual engagement from one of the other two channels has not been selected. If these criteria have been met, an engage signal is sent to the secondary actuator solenoid for that channel. The same criteria are used for all three channels. After the channel has been engaged, the solenoid is automatically disengaged if the above criteria have not been met. The return path from the solenoid is monitored for the proper amount of current. If the current is greater than .49 amperes and an engage signal is sent to the solenoids, the servo light for that channel goes out to indicate to the pilot that the servo is engaged. The pilot may attempt to reset the servo comparators by pressing the servo light, which is also a momentary contact switch.

The power monitor is set to trip whenever the voltages used in that channel exceed 15 V dc ± 3 V dc, -15 V dc ± 3 V dc, 10 V ac ± .654 V ac, or -10 V ac ± .654 V ac; or if the 28-V dc supply drops below 19.2 V dc. The power monitor valid signal is cross monitored between each channel. If two power monitors trip, the remaining good channel locks into a mode similar to single channel manual engage operation. Although auto is still engaged, the remaining good channel provides a priority interrupt which overrides the electronic MVL and servo comparator solenoid disable logic. This interrupt signal also disables the ΔP equalization feedback and forces the electronic MVL to single channel operation. All the electronic warning lights (pitch, roll, and yaw) are illuminated for each channel that has a power monitor failure. The pilot may attempt to reset the power monitor by pressing one of the illuminated electronic warning lights. If the reset is successful, and if all the other criteria for auto engagement are met, the system resumes normal auto operation.

Manual Engagement

Manual engagement for a particular channel is a means of selecting single channel operation. Engaging this mode prevents the other two channels from engaging either the auto or the manual mode. Manual engagement overrides the electronic MVL and servo comparator solenoid disable logic. Manual engagement also disables the ΔP equalization feedback and forces the electronic MVL into single channel operation.

Servo Disengagement

If failures occur which would disable all three channels of a servo actuator - for example, two servo comparators trip, or the pilot elects to manually disable all three channels for a servo actuator - centering springs drive the secondary actuator to a predetermined point. This point corresponds to 10 degrees trailing edge down for an aileron surface, 2.5 degrees leading edge down for an elevator surface, and 0 degrees for the rudder surface.

SELF-TEST OPERATION

Preflight self-test is designed to verify the flight control system before taking off. It is not functional while the aircraft is airborne; therefore, the engagement of the preflight self-test is interlocked with both the weight-on-wheel switches. The preflight self-test can be initiated only when the weight-on-wheel switches are closed, which occurs when the aircraft is on the ground.

Since the preflight self-test for pitch, roll and yaw are fully independent of each other, any axis can be tested separately. The servo switches of the axis to be tested are set to auto on the status/engage panel, with the servo switches of the other two axes off. The preflight self-test will not function properly if the servo switches of more than one axis are engaged. The normal sequence for a complete three axis checkout will be pitch axis, roll axis, and yaw axis.

To start the preflight self-test, turn the lever locked power switch in the self-test section on the status/engage panel to on and depress the DIR switch of that axis under test on the mode and gain panel. Also, reset the PEM indicator of that axis on the status/engage panel. Momentarily depress the push-to-start switch under the switch guard. If the pitch axis is selected, all the lights in the preflight self-test section on the status/engage panel will be turned on while the push-to-start switch is depressed. This serves as the lamp test function at the beginning of the self-test. If the roll axis is selected, only counter lights 32 and 8 and the on light will be turned on while the push-to-start switch is depressed. If only the yaw servos are engaged in the auto position, counter lights 64 and 16 and the on light will be turned on while the push-to-start switch is depressed.

The preflight self-test starts automatically as soon as the push-to-start switch is released. It sequences through the tests as the counter advances if no failure is detected. At the end of the test of that particular axis, the amber on light will go off, and the green pitch or roll or yaw light (depending on the axis being tested) and the go light will come on to indicate that the test has been successful. Test numbers 0 through 39 are allocated for the pitch axis; 40 through 79 are for the roll axis; and 80 through 111 are allocated for the yaw axis.

If a failure is detected during self-test, the on light goes off and the counter stops. The self-test circuitry latches and the red diagnostic lights come on to indicate that the failure is in either the left or the right servo or in the electronics. The lights also indicate which channel and which axis the failure is in. From the failed test number, the diagnostic lights and the preflight self-test table (appendix A), the ground maintenance crew can easily identify the failed area. Turning the power switch off is the only way to reset the self-test.

Figure 14 is a block diagram for the digital fly-by-wire flight control system preflight self-test.

SYSTEM DESIGN FEATURES

PILOT CONTROL INPUTS AND PROCESSING

The CBS design permits a limited amount of flexibility in adjusting shaping filters, gains, deadzones, and limiting circuitry. This was necessary in order to integrate the CBS with the characteristics of the aircraft input sensors and actuators after initial installation.

The primary purpose of the deadzone circuit incorporated in pitch, roll, and yaw servo control paths (figures 6, 7, 8) is to reduce the effects of control stick and rudder pedal centering errors. A secondary purpose, discussed in Servo Equalization and Monitor Design, is to reduce the transmission of unwanted high-gain low-level signals. The deadzone must be large enough to compensate for mechanical centering errors to prevent nuisance trim disable trips from occurring. The rudder pedal deadzone was increased to ±.300 V dc (±13.2% surface) because of the mechanical linkage tolerance between the pedals and the LVDT sensors located in the tail of the aircraft. A ±.100 V dc deadzone for the pitch and roll axes proved to be sound for mechanical centering errors. However, it was discovered after the first two flights that on takeoff and during certain maneuvers the pilot intentionally pulled the stick out of the deadzone and gave a trim command in the opposite direction, which automatically disabled the trim circuit. Therefore, the deadzone was increased to ±.176 V dc for the pitch and roll axes. This deadzone corresponds to ±4.5% of pitch surface level, and ±5.7% of roll

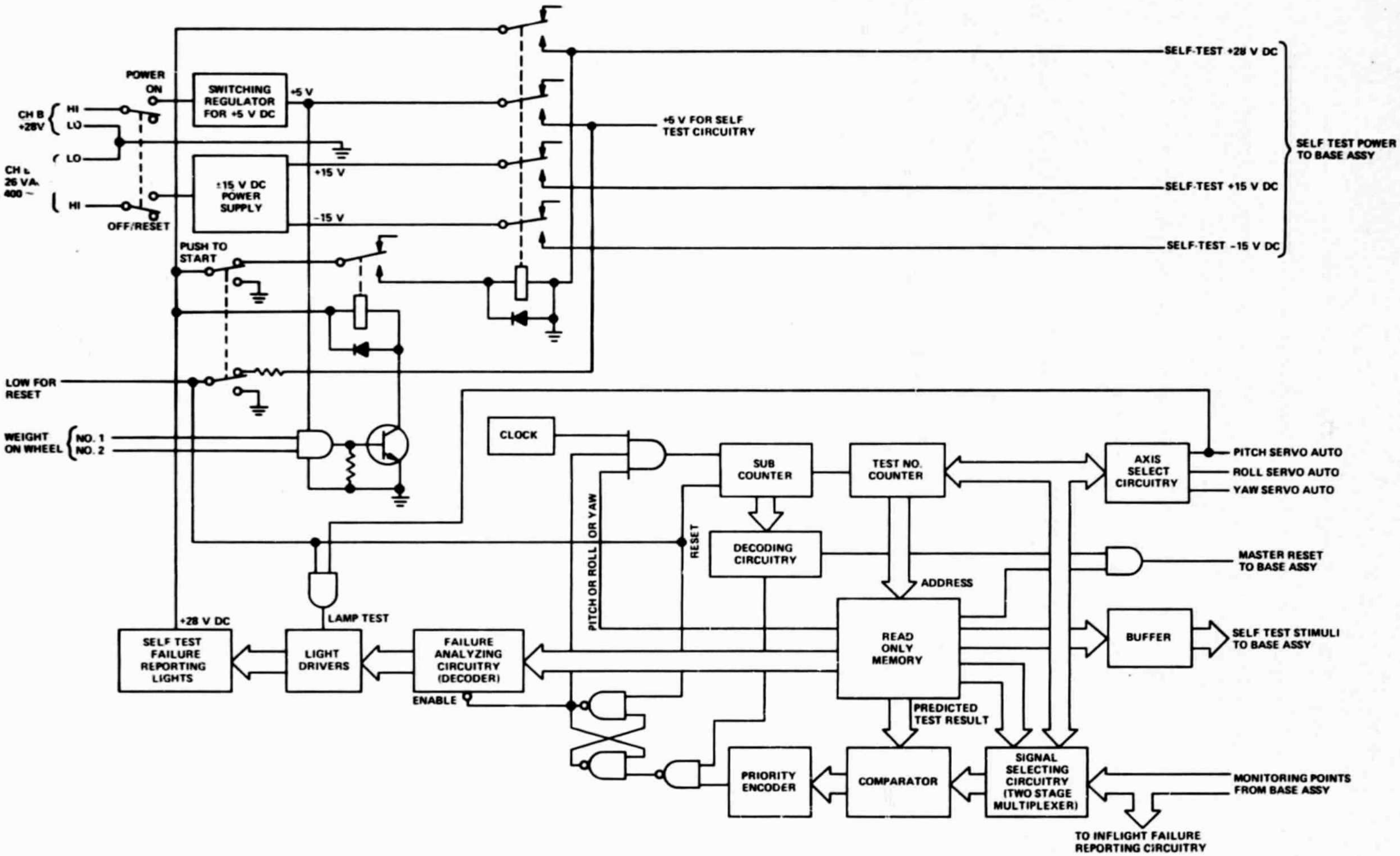


Figure 14. Pre-flight self-test block diagram.

surface travel (wing up). The trim logic dropout time constant was increased from .160 seconds to .490 seconds. Widening the time constant has two effects; it allows for beep trimming, and, if a trim runaway occurs it takes just that much longer before the trim circuit is disabled. The delay would correspond to an increase of .190 to .583 degree of elevator surface travel and .238 to .730 degree of aileron surface travel. If this continues to be a nuisance, a hardware change would be in order. The change should separate the deadzone and the trim detent detector in such a way as to make it possible to adjust the trim detent circuit without affecting the deadzone circuit.

The limiting circuit incorporates temperature compensated zener diodes which have a .01 percent error drift per degree centigrade temperature change. Although these limiters are temperature compensated, they still have a 5 percent value tolerance. Therefore, they were placed in front of the MVLs so that the three channels could be scanned and the middle value selected. The limiting values were selected to give 95 percent to 98 percent of the full mechanical limit of the secondary servo actuators. This limiting circuit is necessary to prevent bottoming-out of the secondary actuator ram and consequent trips of the comparator. The bottoming-out creates trips because of transients resulting from the difference in the pressure characteristics between channels.

DIGITAL SYSTEM INPUTS AND PROCESSING

Flexibility in gain changes are not provided for the DCS surface commands since it is assumed that gain changes can be accomplished under software control. The second-order, low-pass filter does have adjustment capabilities. This filter is provided to reduce the unwanted noise that was seen during the checkout of the breadboards. The limit circuit was duplicated for the same reason as stated for the CBS. The limit is also adjustable and should agree with the CBS limit value.

The CBS/DCS switch and the synchronization/equalization switch are both solid state switches. No relays are used in the backup and servo control system.

COMPUTER BYPASS TRIM SYSTEM

The computer bypass trim system is implemented by inputting a beep trim input from a stick-mounted pushbutton into a digital integrator. The output of the integrator is summed with the stick input to command elevator displacement. The trim implementation is illustrated in figures 6, 7, and 8 for yaw, pitch and roll axes.

The beep trim integrator has a digital rather than an analog implementation due to the requirement that aircraft trim must be held on this integrator over long periods of time. This digital implementation has no long term drift characteristics, whereas any analog integrator has a finite long term drift rate. The integrator is also used in channel equalization and synchronization in the DCS mode.

The digital integrator implementation utilizes an 8-bit up/down counter as the memory element. The digital-to-analog conversion of the counter outputs yields a resolution of one least significant bit or approximately 90 millivolts. This resolution is equivalent to .386 degree in roll, .206 degree in pitch, and .180 degree in yaw. This resolution results in a noticeable granularity in surface movement when trim is being used. Also, the trim resolution in roll is such that small vernier adjustments to aileron trim are difficult to accomplish. An obvious solution to the above problem was to increase the resolution of the digital integrator by increasing the number of bits in the up/down counter to 9 or 10. However, because of physical space constraints in the hardware, this solution was not feasible without a major system redesign. Another possible solution would seem to be a reduction of the trim authority (trim rate and authority are adjustable); thereby increasing the integrator digital to analog resolution. However, channel equalization and DCS synchronization are also accomplished using the digital integrator. Synchronization of the CBS to the DCS when flying in the DCS mode requires that the synchronization integrator be able to synchronize over the full range of surface authority. Any reduction of trim authority would also reduce the synchronization authority. Reducing synchronization authority could result in unacceptable surface transients when a system downmode from DCS to CBS occurs. It is apparent that increasing the resolution of the trim integrator by reducing the trim authority is an unacceptable solution.

The CBS detent detector is set to detect any stick or rudder pedal deflection greater than the CBS deadzone. The trim logic compares this detent with the trim command. If the trim command and detent detector disagree, then the trim circuit is automatically disabled and the trim warning light lights up. The trim logic latches up in the disabled position and the stick or rudder pedal can be released. This logic is resettable by the pilot. Since the CBS stick and rudder pedal deadzone is small, a .490 second dropout delay was necessary to allow for beep trimming in the opposing direction of the stick.

COMMAND CHANNEL EQUALIZATION AND SYNCHRONIZATION

The equalization circuit forces the CBS trim rates for the three channels to follow each other within the boundaries of the equalization deadzone ($\pm$.393 V dc error between channel command and MVL output). It does this by adding or subtracting from the trim rates up to a maximum of $\pm$.189 V dc per second.

Synchronization has a faster rate and is used to reduce surface transients when downmoding from DCS to CBS. Equalization is used to null out any steady-state errors between channels while in the CBS mode. Synchronization is always engaged for a DCS mode regardless of whether the CBS/DCS switch is in the DCS or CBS position. Equalization is always engaged for the CBS mode.

One of the tradeoffs involved in using the synchronization scheme to reduce surface transients is that the voltage in the synchronization equalization circuit needed to reduce surface transients, when downmoding from DCS to CBS becomes a steady-state voltage for the CBS. For instance, if the pilot is using the side stick when a downmode occurs, the side stick command as seen at the time of the downmode remains in the synchronization/equalization circuit. This command must then be trimmed out by the pilot in order to return to a state of equilibrium.

COMMAND CHANNEL VOTING

Each MVL has three modes of operation which are controlled by logic. These modes are: middle value selection, single channel operation, and output disable.

The purpose of middle value selection for triplex operation is to select a single value for the three channels of the closed loop servo drive. Without a single value being selected, any difference between channels could cause reduced servo performance, pressure force fights, and possibly servo actuator disengagement (depending upon the level of pressure force fights). The MVL also reduces single channel spiking and prevents passive and hardover failures from getting through to the closed loop servo drive. It can do this because as soon as a faulty channel, if it is middle value, crosses over another channel value, it is voted out and the other channel is selected.

Single channel operation occurs if manual engage is selected for that channel, or if power fails in the other two channels.

The output is disabled if two or three electronic comparators for that channel do not track.

COMMAND CHANNEL REDUNDANCY MANAGEMENT AND MODE CONTROL

All comparators were designed with adjustable trip levels and time delays. These trip levels and time delays are given in table 2. The electronic and PEM operators readily detect single channel hardover conditions, passive failures are more difficult to detect since the comparators require a level of mismatch. This mismatch corresponds to a surface command equivalent to the trip level given in table 2. Presently, the trip levels are the same as they were for phase I, except for that for the roll electronic comparator, which was reduced by 15 percent.

A 40-millisecond delay was designed into the logic when downmoding from DCS to CBS to reduce nuisance downmoding. A 15-millisecond delay was designed into the system prior to the one-shot resetting of the electronic comparators. This delay was necessary to prevent a possible CBS-DCS mode mismatch after downmoding from DCS to CBS. For example, assume that channel A and channel B electronic comparators have just tripped while in the yaw DCS mode. Normal operation would call for all three channels to downmode to CBS in 40 milliseconds. However, due to tolerance effects, assume that channel A downmodes first and resets its electronic comparator before channel B and channel C complete their downmodes to CBS. For this case, channel B and channel C would stop downmoding, since now only channel B has an electronic comparator tripped (it takes two tripped electronic comparators to downmode to CBS). The modes would now be channel A = CBS, channel B = channel C = DCS. Because of this situation, a 15-millisecond delay was added to ensure that all channels are downmoded to CBS before the electronic comparators are reset.

TABLE 2. ELECTRONIC AND PEM COMPARATOR TRIP LEVELS

Comparator	Trip Level		Time Delay
	±V dc	±Surface degrees	
Roll Electronics, Left and Right	2.43 ± .17	10.94 ± .77 A	200 msec
Pitch Electronics, Left and Right	2.84 ± .20	6.82 ± .48 E	200 msec
Yaw Electronics	2.84 ± .20	5.96 ± .48 R	200 msec
Roll PEM	2.84 ± .20	12.78 ± .90 A	400 msec
Pitch PEM	2.84 ± 20	6.82 ± .48 E	400 msec
Yaw PEM	2.84 ± .20	5.96 ± .42 R	400 msec

SERVO ENGAGE INTERLOCKS

Each channel of the secondary actuator has its own hydraulic supply: the two primary and single utility hydraulic systems. It should be pointed out that the primary (surface) actuator is supplied from only the two primary hydraulic systems standard for this aircraft. Therefore, failure of the two primary hydraulic systems would disable all primary surface actuators.

The original phase II design allowed a tolerance of up to 50 milliseconds of disengagement time between the two on channels. It was found during ground check that when a second failure was detected by the servo comparators, timing was critical for disengaging the remaining two channels. Excessive disengagement time caused large surface transients. For example, if a hardover failure occurs in one of two remaining channels, and the valid channel is disengaged first [assuming a roll actuator with a slew rate of 38.1 cm (15 inches) per second], the following transient could be expected:

$$(.05 \text{ second}) \frac{(38.1 \text{ cm})}{\text{second}} \frac{(60 \text{ degrees})}{5.08 \text{ cm}} = 22.5 \text{ degree transient}$$

This tolerance was tightened up for a maximum of 5 milliseconds of delay between disengagements.

The normal procedure for the first servo-comparator-detected failure is for the pilot to attempt to reset this comparator. If the reset is unsuccessful, he switches that channel off. If a second failure occurs, all channels for that actuator are switched off. <u>No attempt should be made to engage manual, or to reset the comparators after two failures, since this could cause a hardover state</u>.

SERVO CONTROL SYSTEM DESIGN

The design of the secondary actuator servo control system was arrived at via the analysis, simulation, and testing of the flyable hardware in the F-8 iron bird mock-up. Analysis was used to establish and verify the adequacy of position and pressure loop gain and phase margins. Simulation using a linear model for the servovalve was used to perform system time history responses and arrive at the desirable initial values for the monitoring system design. Testing in the iron bird mock-up provided the opportunity for performance verification and optimization in a realistic operating environment. Under these conditions the impact of servovalve nonlinearities and aircraft hydraulic characteristics was ascertained, and refinements to the design were introduced as performance requirements dictated. These investigations resulted in the secondary actuator servo system design shown in the transfer function block diagram in figure 15.

FOLDOUT FRAME

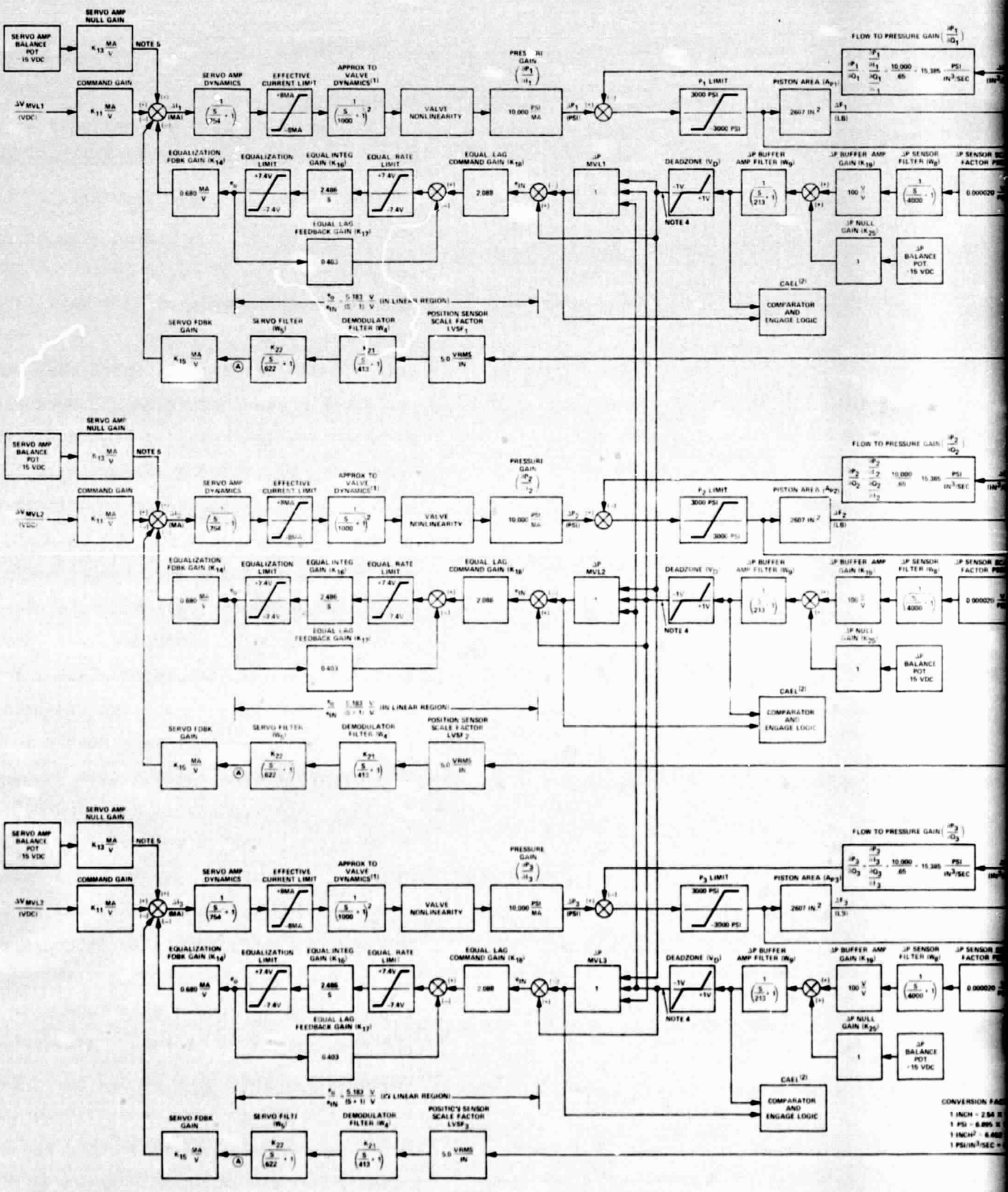
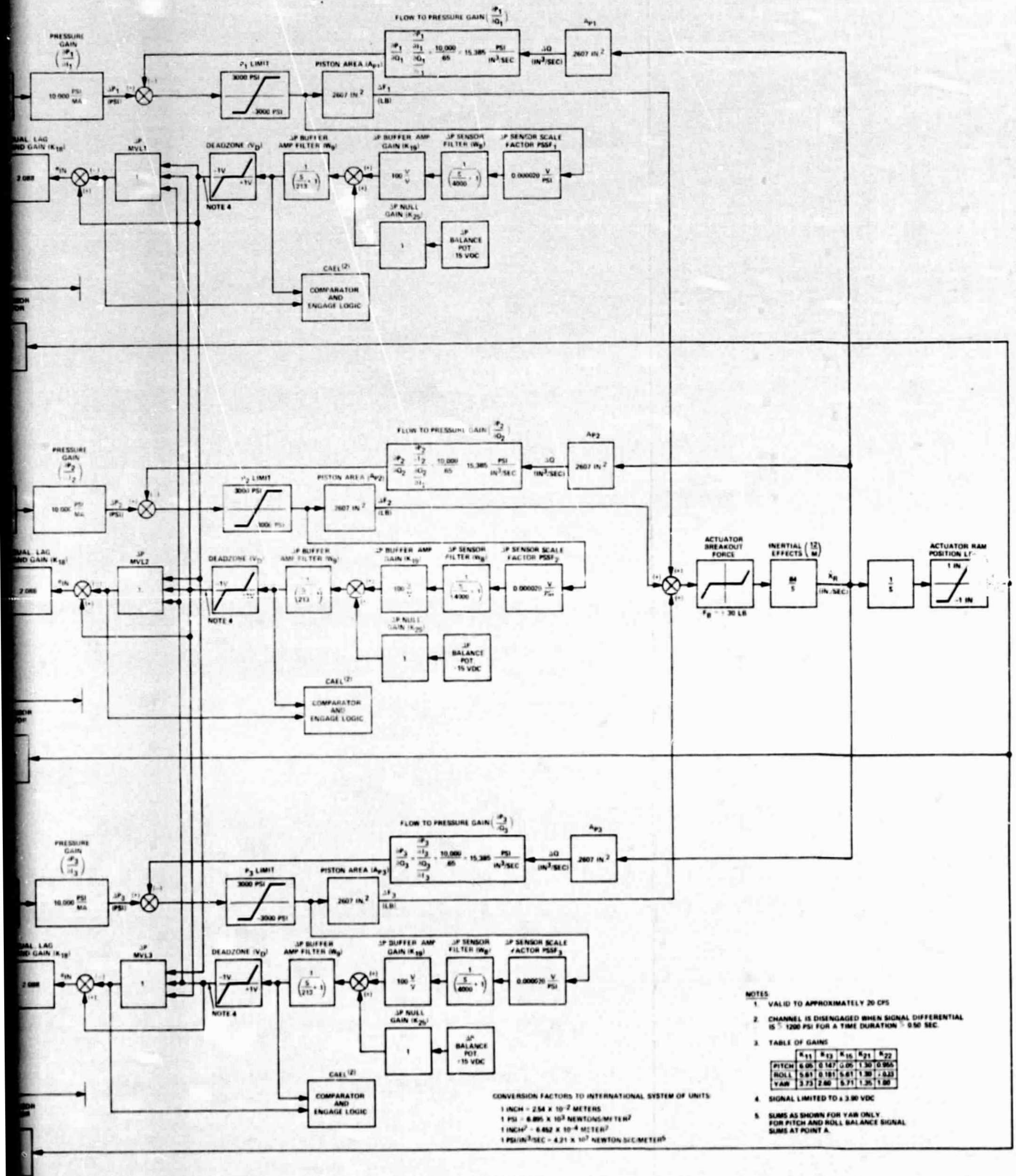


Figure 15. F-8 DFBW secondary actuator transfer function



. F-8 DFBW secondary actuator transfer function block diagram.

Position Loop

The position loop was designed so that at 20 Hz the closed loop response had a normalized gain of ±3 decibels and a phase lag of less than 90 degrees. Given a valve flow gain of $1.07 \times 10^{-5} \frac{m^3/\text{second}}{ma}$, $\frac{.65 \text{ inch}^3/\text{second}}{ma}$ a piston area of approximately 1.68×10^{-4} m^2 (.26 $inch^2$), and an LVDT scale factor of 197 volts rms/m (5 volts rms/inch); a maximum feedback gain of 1476 ma/m (37.5 ma/inch) from RAM position to servo amplifier output was found to be acceptable. This resulted in a total loop gain of 93.5 per second, which when combined with the dynamic effects around the position loop produced the Bode plot in figure 16. The position loop dynamics, as indicated in figure 15, includes a two stage roll-off filter to reduce 800-hertz demodulator ripple, a servo amplifier lag filter associated with stabilizing the servo amplifier, and the second order approximation to the servovalve dynamics. The gain and phase margins measured from figure 16 are 10 db and 53 degrees, respectively. The gain margin meets the typical servo design objective, while the phase margin is 7 degrees less than the normal design goal of 60 degrees. However, since the phase II hardware was derived from the "in-place" phase I F-8 secondary servo actuator design, it was concluded that this slight reduction in phase margin did not warrant the cost of making any hardware module design modifications. A 2 db reduction in the electronic feedback gain would have yielded the desired phase margin. However, this would have been accomplished at the expense of increased sensitivity to valve and actuator nonlinearities when the pressure loop was closed. This was deemed to be an undesirable trade-off, and hence the lower phase margin value was considered to be acceptable. This is supported by the actuator step responses illustrated in figure 17 where it is seen that the peak overshoots are well within desirable limits (less than 10 percent). A corresponding closed loop Bode plot is presented in figure 18. This plot and step responses of figure 17 were obtained on the F-8 iron bird mock-up. The relative gain of 0 db and -73 degrees of phase at 20 Hertz are well within the design requirements. On-line performance evaluation as well as off-line analysis has shown that the basic servo loop stability and step response is essentially independent of pressure equalization loop closure.

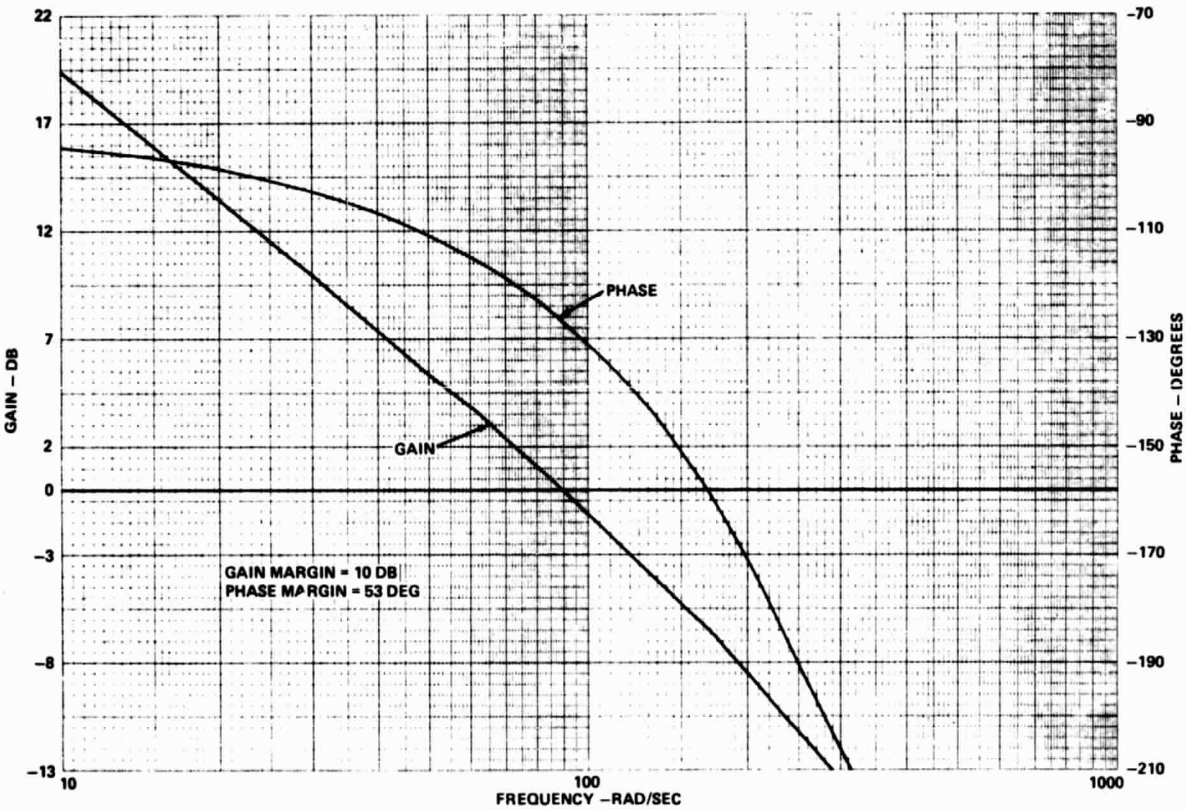


Figure 16. Secondary actuator open loop Bode plot of position loop.

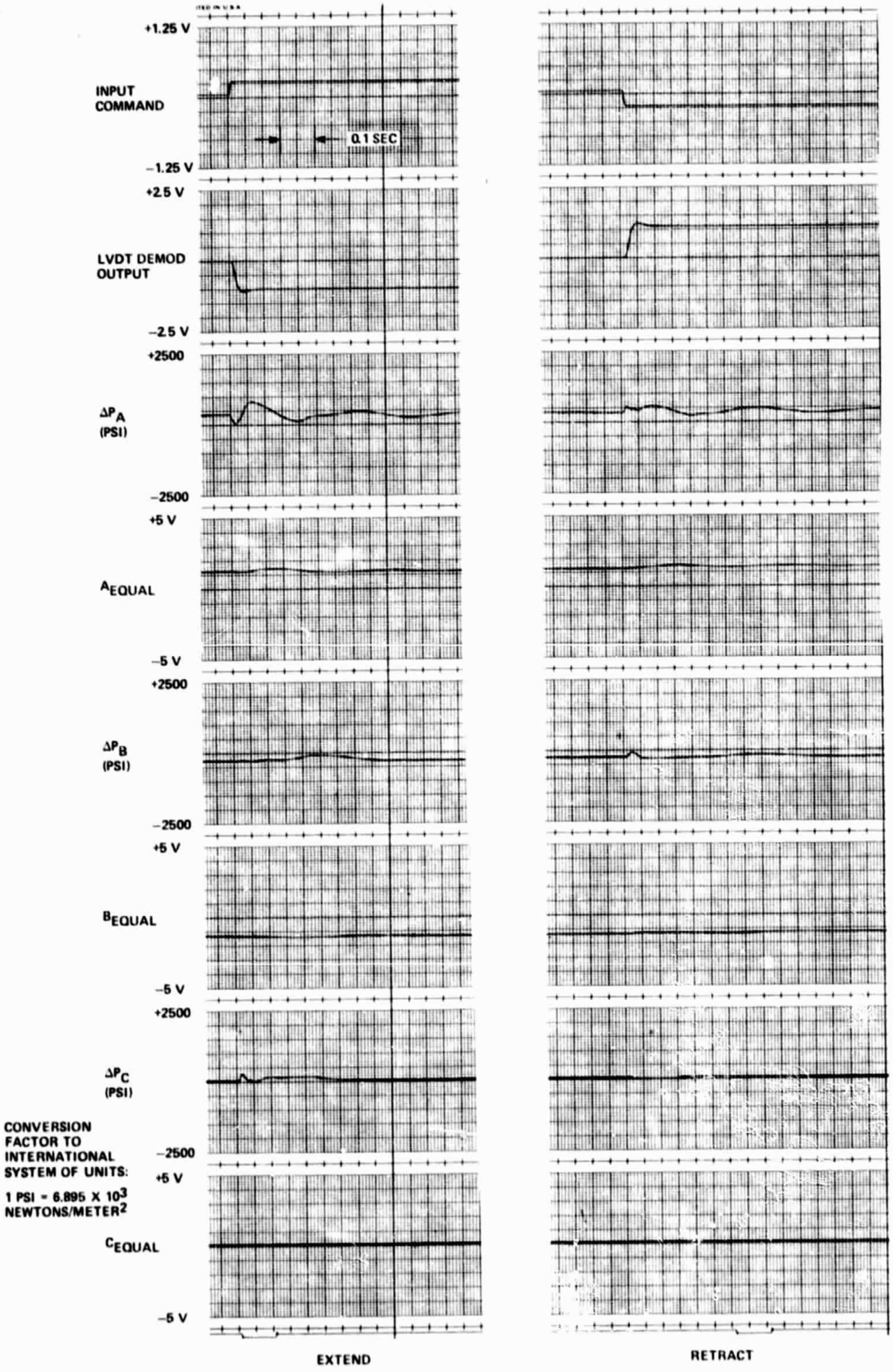


Figure 17. Response of triplex system, actuator S/N 001 to an equivalent ±2 degrees δ_R step.

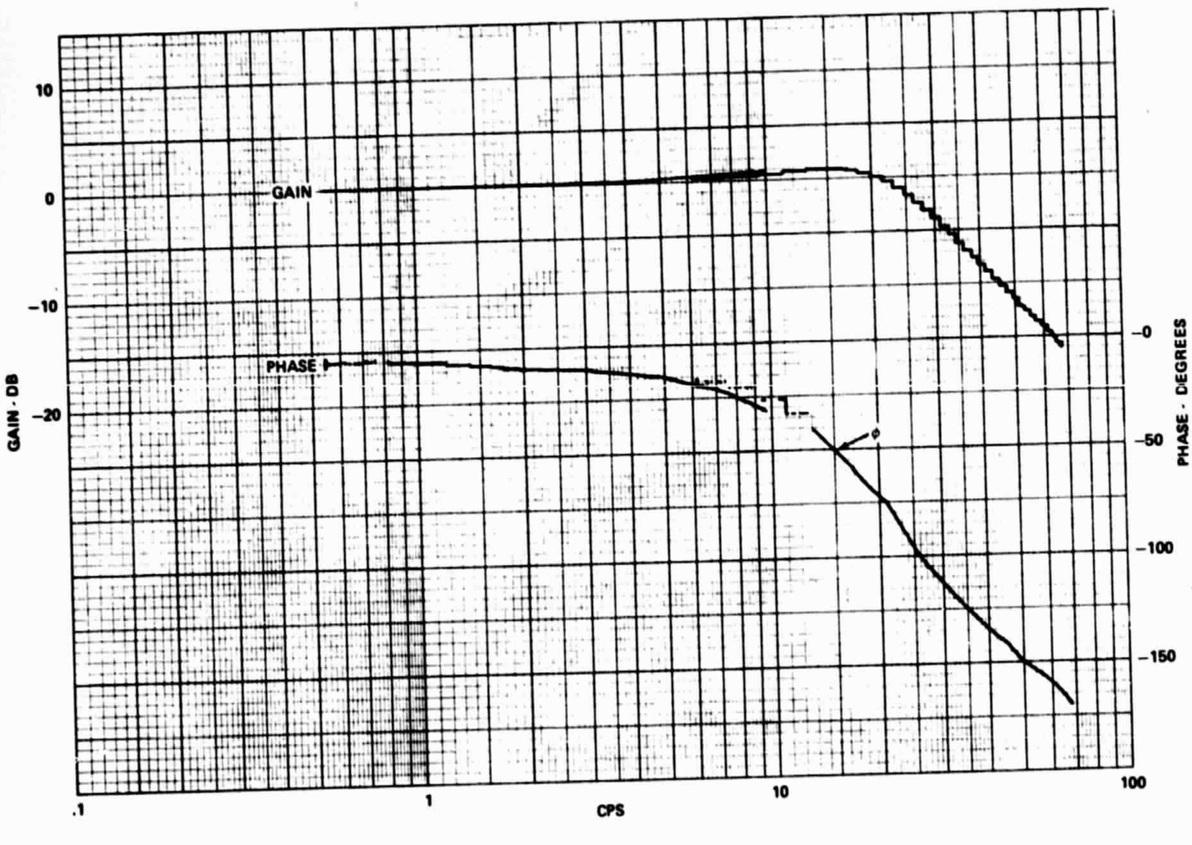


Figure 18. Closed loop Bode plot for actuator (input amplitude = 0.61 cm peak-to-peak).

The step responses of figure 17 support this conclusion, and correlate well with a relative damping ratio projection of .6 to .65 that is predictable from the Bode plots of figures 16 and 17.

Servo Equalization and Monitor Design

Description.- The equalization and monitor functions for the secondary actuator system, described previously in Servo Actuator Engagement, are shown in figure 15, which is an idealized representation of the actual servo electronic circuitry. This figure represents the functional characteristics of the servo system which were included in the simulation. For each of the force summed servos, the pressure sensor signal is filtered, amplified, limited and finally fed to a deadzone circuit. The signal from the deadzone circuit is applied to a MVL input terminal, then subtracted from the MVL output to provide the input to an equalization lag circuit. Following filtering (one-second lag), amplification and limiting in the equalization lag circuit, the equalization signal is summed with the servo position command. The monitor consists of a comparator circuit which compares the difference between the limited and buffered pressure sensor signal and the MVL output.

These equalization circuit functions provide the means to change any servo channel pressure towards the mid-value pressure corresponding to the MVL output. The channel or channels corresponding to the mid-value pressures are not affected; only those channels differing from the mid-value pressure have non-zero equalization feedback signals. Similarly, the monitor reading will be zero for channels corresponding to the mid value pressures since in such a case MVL outputs will equal input. For a sudden change in the position signal to only one servo, piston pressures will first increase, then reduce in magnitude with time. This is due to the equalization of the differences between servo channel pressures. The exact characteristic will depend upon the time history of the position signal, pressure and position loop gains, and the lag time constant in the equalization feedback path.

Objectives.- The major objective of servo equalization is to make fault monitoring feasible. Other advantages include improved tracking performance in two-channel operation.

Unless some means exists to automatically detect and isolate a failure, the triplex secondary actuator configuration cannot be justified. Only if automatic disengagement of the faulty channel occurs will the reliability and safety requirements be met. Experience has shown that most failures in a force summed servo system will cause unequal and opposing forces across the individual hydraulic pistons. This fact has encouraged widespread use of a fault monitoring technique based on servo piston force measurements, utilizing simple pressure sensors. The drawback has been that, without equalization, typical servo and electronic circuit tolerances can also result in the force-opposition condition, even when no failure is present. The equalization path, as described above, has the property of forcing the pressures to the same level for normal tolerance spread. Changes in signal levels due to faults (changes in excess of the normal system tolerance) will result in exceeding the comparator trip limit and disengagement of the faulty channel.

The requirement to equalize pressures for monitorability is of particular importance in a secondary actuator design such as the F-8, which uses servovalves having a high nominal pressure gain of 68.9×10^6 N/m^2 per milliampere (10,000 psi per milliampere). For example, a channel unbalance equivalent to only .3 ma of the servo amplifier could cause a maximum force to be realized at the piston of this particular channel.

Lack of equalization tends to degrade performance during two-channel operation. Given the case where the aircraft is in a trimmed flight condition and the two servo position commands lie on opposite sides of zero, the output will remain at zero until either of the commands passes through zero. Low level oscillations can result. Addition of equalization essentially eliminates this problem, since the effect is to force both servo amplifier output signals together.

Final Adjustments.- The selected values of the equalization circuit parameters were established with the actual hardware, under operating conditions. Parameters, varied to provide optimum fault detection capability, included equalization loop gain, lag time constant, monitor trip level, monitor trip time delay, equalization rate and authority, and equalization circuit deadzone.

Two system nonlinearities tended to degrade monitoring performance: the three hydraulic supply systems exhibit severe pressure/flow mismatch under large dynamic flow conditions, and the secondary actuator servovalve is characterized by extremely high gain about the zero flow region. Ratcheting and granularity effects resulted from the servovalve nonlinearities for low level dynamic signals. Poor pressure tracking between pistons resulted from the supply pressure mismatch condition.

The solution to the low-level ratcheting problem was to increase the deadzone region to one-half the channel trip level. This opens the equalization path for low-level signals which are below monitor thresholds. Incorporation of the deadzone results in improvements in the position response for low-level inputs over the zero to two hertz frequency range, without compromising fault detection capability.

Increases in equalization loop gain and selection of a .4 second monitor trip time sufficed to resolve the nuisance trip problem resulting from hydraulic system mismatch. This ensured that no nuisance trips would result when cycling the servo, stop-to-stop, at a frequency of one hertz, with a peak current mismatch of one milliampere between channels. Pressure loop bandwidth is illustrated by the Bode plot in figure 19. This is a plot of open loop amplitude/phase versus frequency for the total linearized pressure loop in figure 15. The zero decibel crossover at 67 radians per second attests to the large pressure loop bandwidth required. Equalization feedback without lag filtering would tend to mask failures; the lag filter is

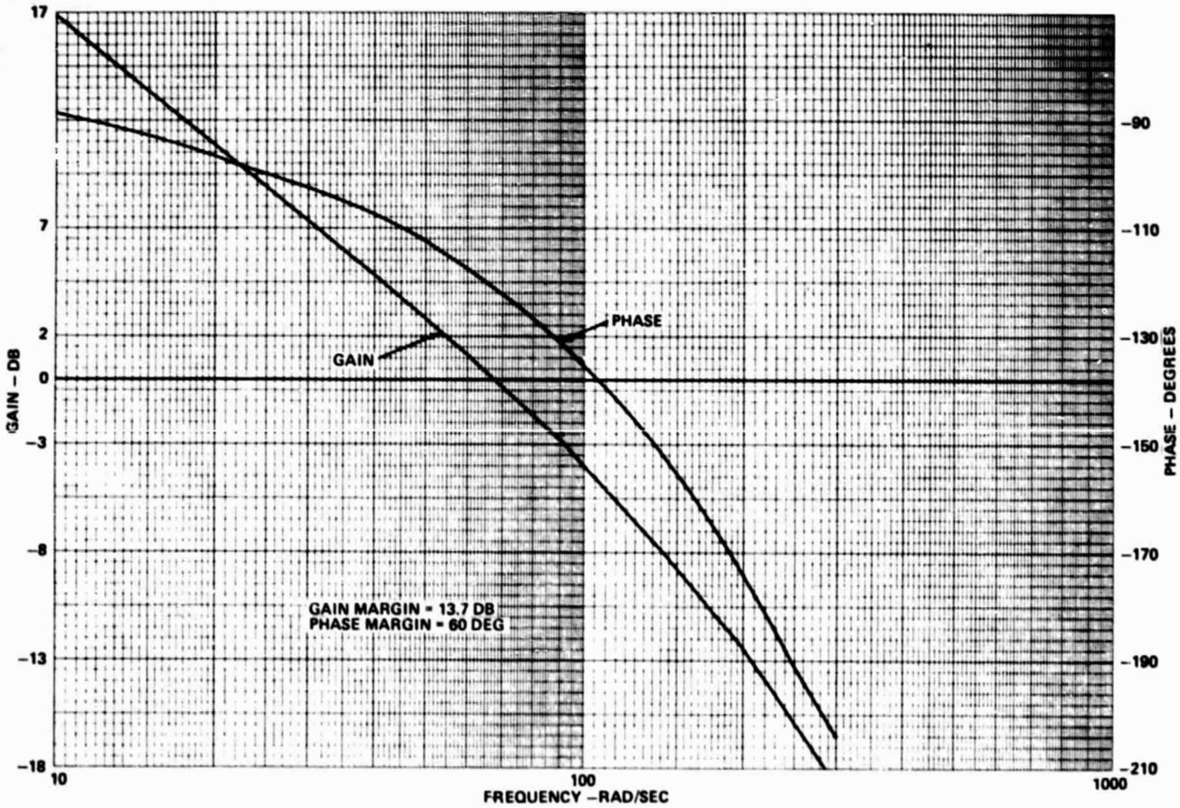


Figure 19. Open loop Bode plot of pressure loop.

rate limited to ensure that high frequency faults will not be equalized out. Lag time constant values in excess of one second were found to aggravate the nuisance disengage problem.

It may be noted from figure 19 that the gain and phase margins are 13.7 decibels and approximately 60 degrees, respectively. These margins are consistent with a stable pressure loop design, and were found to be necessary for minimizing pressure-to-position loop coupling.

Equalization feedback authority is limited to 13.5 percent. This value is a compromise between the requirement to minimize aircraft transients due to failures, and the need to provide equalization of anticipated static nulls, offsets, and dynamic channel mismatches.

In order to detect a fail-to-zero condition of any LVDT position sensor, it is required that the loss of the LVDT signal create a pressure change sufficient to trip the monitor. The critical condition corresponds to a trimmed surface, which is the most likely situation. Rather than rig the LVDT to produce zero volts for this trimmed condition, which could result in a passive failure, the LVDT is rigged to provide a small signal for no command input. The resulting servo amplifier bias signal is balanced out by appropriate adjustment of the servo amplifier balance potentiometer.

SELF-TEST DESIGN PHILOSOPHY

The design philosophy of the preflight self-test for the phase II digital fly-by-wire computer bypass system is different from that used in phase I. In the phase I system, if a self-test in the yaw axis was desired, it was necessary to first complete the pitch and the roll axes before initiating self-test in the yaw axis. In phase II, this feature was redesigned so that each axis could be started independently from the other two.

In addition, the monitoring points were increased from eight per channel (pitch electronic status, pitch left servo status, pitch right servo status, roll electronic status, yaw electronic status, and yaw servo status) as in phase I to sixteen per channel. The pitch and roll electronic status in phase I was changed to pitch left, pitch right, roll left, and roll right electronic status. Furthermore, pre-engage monitor and trim monitor for pitch, roll, and yaw were added in phase II.

Finally, a self-test binary counter indicator was added on the panel for easier failure isolation.

In every test, the first half cycle is used to reset all the comparators (monitoring points) in all three channels unless there is a zero (0) in the self-test stimulus No. 6 Inhibit Master reset column (appendix A). A zero in the Inhibit Master Reset column for that test means that the test result from the previous test shall be carried to the present test. Combining the carryover test result with self-test stimuli No. 1 through No. 4 (zero means apply stimulus) to be applied to the appropriate channel during the second half cycle of the test, the expected test results could be yielded at the examination period at the end of the test.

At the beginning of the second half of each test, the stimuli stored in the read-only memory are enabled and applied to the appropriate places. After a short delay, the responses of all the monitoring points (comparators), are selected through a two-stage multiplexer (the first stage selects the axis, the second stage selects the desired monitoring point) to pick up the desired monitoring points. These points are then compared with the predicted test results. If the status of the selected monitoring points agrees with the predicted test results, the test counter passes the examination gate at the end of that test and advances to the next test. If the status of one or more of the selected monitoring points disagrees with the predicted test results, the failure result with the highest priority (PEM first, then channel A, B, and C) is strobed into a latch. The amber self-test on light is turned off and the counter stops. The proper red diagnosis lights come on based on the information strobed into the latch, the test number in the counter, and the monitoring points selecting information stored in the read-only-memory.

The magnitude of the self-test stimuli for all three axes in the DCS mode is equivalent to 40 percent of the maximum authority. In the CBS mode, the magnitude of the stimuli is 32.9 percent in the pitch axis, 28 percent in the roll axis and 50.7 percent in the yaw axis.

A table which lists the status check for every comparator is included as appendix B.

HARDWARE DESCRIPTION

BYPASS AND SERVO ELECTRONICS

The CBS components that provide the buffering, bypass, monitoring, and servo electronic functions consist of three identical BSE assemblies. Each assembly measures 10.4 x 27.2 x 11.4 cm (4.1 x 10.9 x 4.5 inches) and weighs 3.45 kg (7.6 pounds).

Each LRU's components are located in 11 plug-in cards; one card is fastened in the power supply. Each unit was subjected to operational, vibration and temperature tests before being shipped. The operational test was conducted with SFS computerized automatic test equipment.

Twenty-eight-V dc power is used in the logic for this system and is converted to ±15 V dc and 5 V dc for the analog portion of the system. A current of 1.4 amperes is drawn from the 28-V dc bus. The 26-V ac supply is routed into the BASE box and back out again for primary power to the control stick and rudder pedal LVDT's and secondary actuator position LVDT's. The 26-V ac supply is converted to in- and out-of-phase 10 V ac for reference use in the BASE box.

One dedicated connector on each unit is provided to bring out critical test points. These test points are used for telemetry monitoring, system preflight, and maintenance testing.

The BASE system was packaged in such a way as to split up left and right pitch and roll functions onto separate cards. This was done to prevent two failures from disabling both left and right aileron or elevator surfaces. Single functions common to the pitch and roll axes, such as CBS and DCS input points, were also split up onto separate cards. Other single functions such as the power monitor had logic designed so that no two primary power failures would disable all surfaces. There is no dual function in the yaw axis, and it was stipulated by NASA that the aircraft be flown safely with a failed rudder actuator.

Figure 20 is an outline of the bypass and servo electronics assembly.

STATUS/ENGAGE PANEL

Figure 21 shows the front layout and the inside view of the status/engage panel. The panel measures 14.5 x 38.1 x 15.2 cm (5.7 x 15 x 6 inches) and weighs approximately 6.35 kg (14 pounds). The status/engage panel used in the phase II DFBW is a modification of the panel used in phase I. The modification consisted of removing all the indicators and switches related to channel 1, changing all the legends from channels 2, 3, and 4 to channels A, B, and C, adding PEM and TRIM lighted pushbutton switches and the associated lamp drivers and telemetry interface circuitries, adding a self-test counter indicator to the preflight self-test section, modifying the self-test circuitries to accept 10 monitoring points per channel, and modifying the circuitry such that the self-test for each axis could be started independently of the other two axes.

The status indication and engage function for channels A, B, and C in phase II uses the same circuitry as was used in phase I for channels 2, 3, and 4. The status indication and reset function for trim in phase II uses

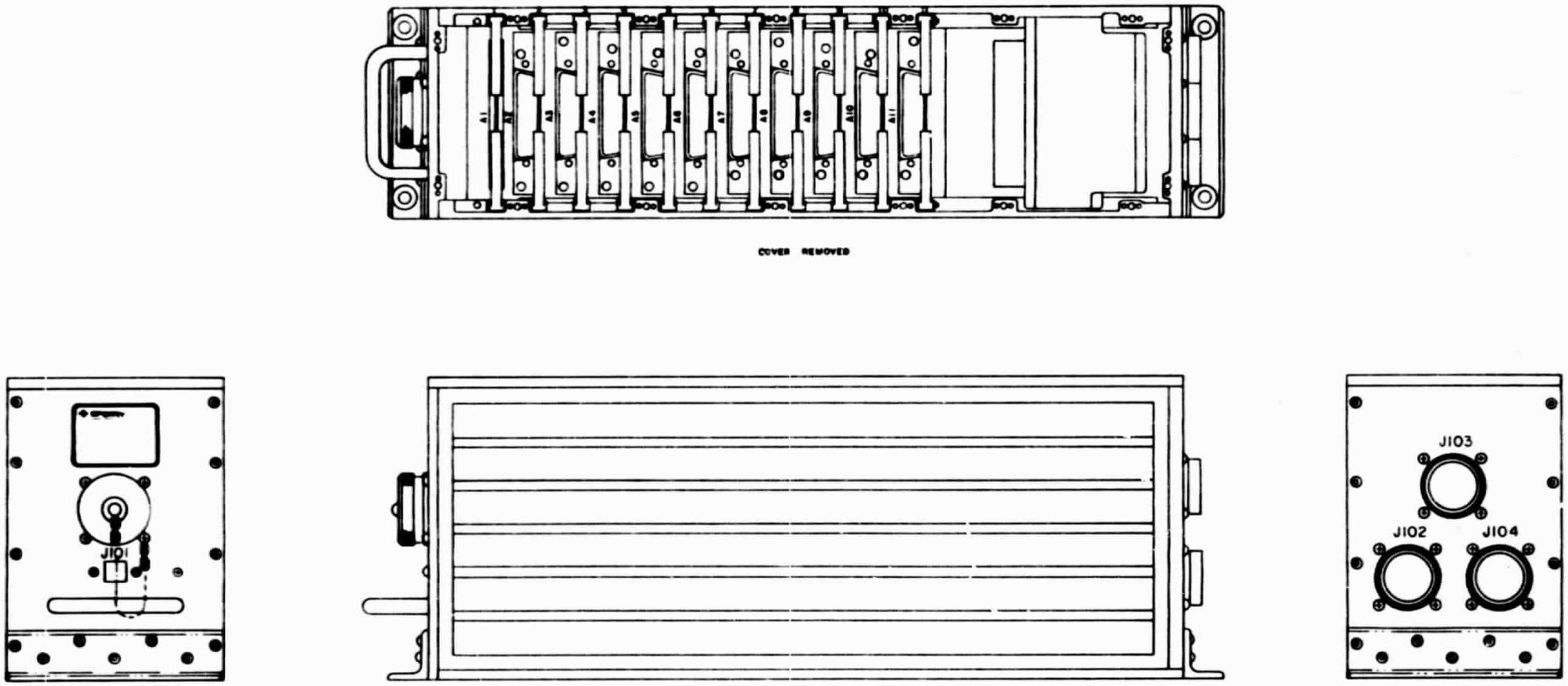


Figure 20. Bypass and servo electronics assembly.

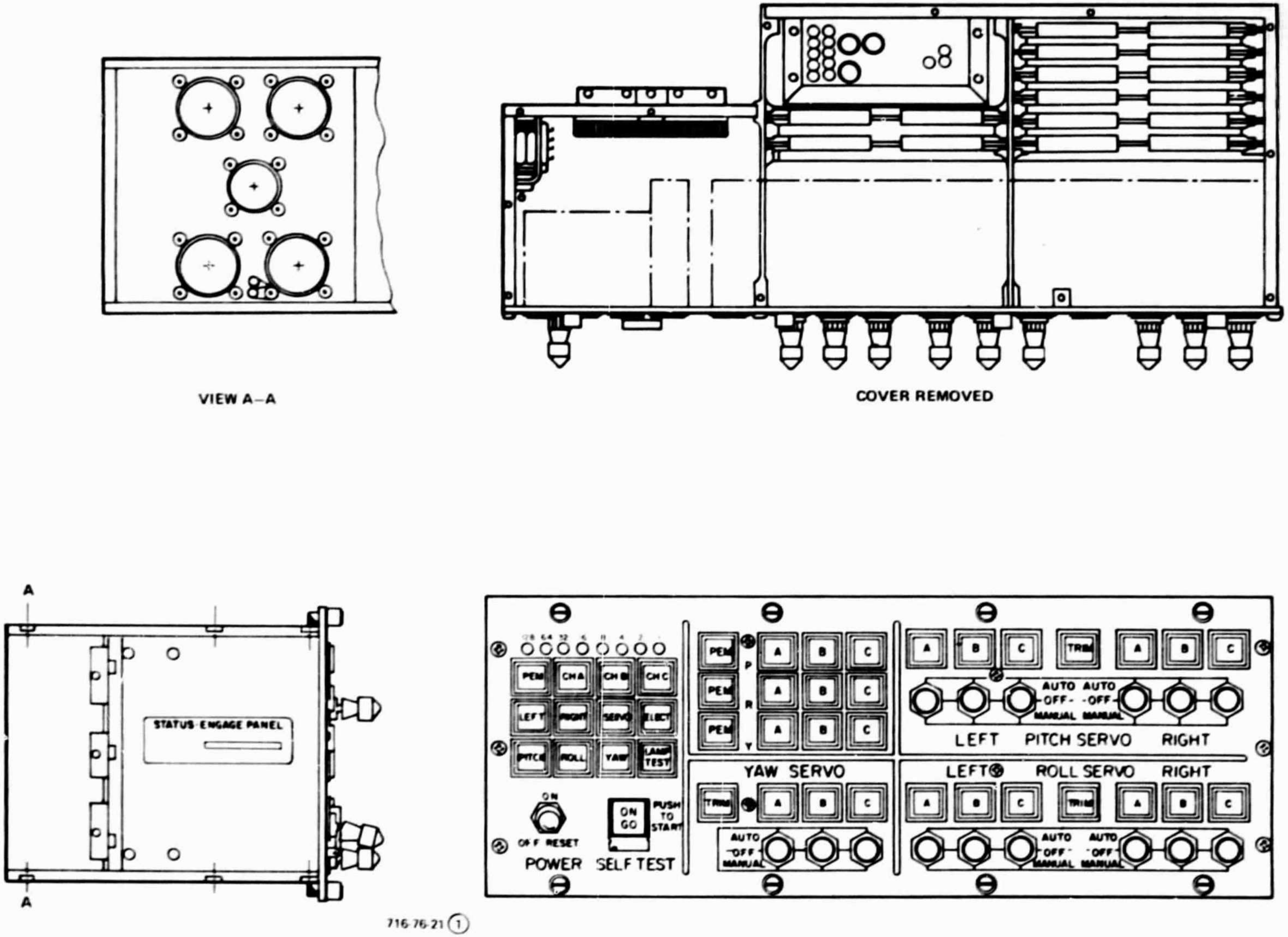


Figure 21. Status/engage panel.

the circuitry originally for channel 1 in phase I. PEM status indicators and reset functions for all three axes are newly added to the panel, and all of the associated lamp drivers and telemetry interface circuitries have been added to card A7. Due to the interchangeability of cards A7 and A8, the same circuitries were added to card A7 that also have been added to card 8.

Everything related to channel 1 in phase I was modified or removed, leaving connector J1 as a dummy only.

As mentioned in SELF-TEST DESIGN PHILOSOPHY, every change related to self-test from phase I to phase II was incorporated into card A2 through A5 by modifying those cards used in phase I, which includes the self-test independence among axes; two stage multiplexing to select the desired signals from an increased number of monitoring points; and the added circuitries for self-test binary counter indicators.

APPENDIX A

PREFLIGHT SELF-TEST TABLE

The CBS self-test tables that follow provide a means for the crew to systematically follow the test sequence and gain an increased understanding of the actual operations involved. However, reference to these tables is not required to interpret the test results since failures are reported-out on the status/engage panel.

As an aid in self-test table interpretation, a column-by-column description for test no. 1 (pitch) in table A-1 is presented below. Remaining entries follow the same format.

TABLE A-1. SELF-TEST TABLE, TEST NO. 1 EXPLANATION

1. Test numbers 0 through 39 are for the pitch axis only; test numbers 40 through 79 are for the roll axis only; and test numbers 80 through 111 are for the yaw axis only.

2. First column identifies the test number (no. 1 in this example).

3. Second column lists channels A, B, and C.

4. Under the "S.T. Stimuli" (Self-Test Stimuli) heading, a 0 under #2 in row A means self-test stimulus #2 in channel A is activated; a 0 under #5 Sync Disable means the synchronization between the digital primary system and the analog backup systems is disabled.

5. A 1 under the "Comparator" heading means those comparators shall be failed based on the stimuli applied for that test. For test no. 1, PEM shall fail, and both the "LE" (left electronics) and the "RE" (right electronics) in channel A shall fail.

6. Under the heading "Monitoring Point" are the comparators that were selected in the corresponding channel for monitoring. "CLE" is used in channel A, "CLE" is used in channel B and "CRS" (right servo comparator) is selected in channel C for monitoring in test no. 1.

7. Under "Function Under Test," the first row states that the monitored point, PEM, shall fail in both channels A and B due to the stimuli applied for this particular test.

In test no. 1, #2 S.T. Stimuli was applied only to channel A, and the synchronization (S.T. Stimuli no. 5) was disabled. The 0 +, + 0, 0 - or - 0 after "F" shows the polarity which causes the comparator to fail.

8. If the status of the comparator under monitoring does not agree with the status predicted, the self-test will be stopped and the failure mode shall be reported-out on the control panel as indicated in the last column "Failure Report Out."

TABLE A-1. SELF-TEST TABLE

Test No.	Channel	S.T. STIMULI #1	#2	#3	#4	#5 Sync Disable	#6 Inhibit Master Reset	Comparator PEM	1 = FAIL L_E	R_E	L_S	R_S	Monitoring Point	Function Under Test	Failure Report Out
0						0	0						PEM	GO	P PEM
	A												CLE	GO	LE
	B												CRE	GO	RE
	C												CLE	GO	LE
1						0		1					PEM	F 0 + (A) + 0 (B)	PEM
	A		0						1	1			CLE	F + 0	LE
	B												CLE	GO	LE
	C												CRS	GO	RS
2						0		1					PEM	F 0 + (B) + 0 (C)	PEM
	A												CRS	GO	RS
	B		0						1	1			CLE	F + 0	LE
	C												CLE	GO	LE
3						0		1					PEM	F 0 + (C) + 0 (C)	PEM
	A												CLE	GO	LE
	B												CRS	GO	RS
	C		0						1	1			CLE	F + 0	LE
4						0		1					PEM	F 0 − (A) − 0 (B)	PEM
	A	0							1	1			CRE	F − 0	RE
	B												CRE	GO	RE
	C												CLS	GO	P LS

NOTE: C = Comparator Status, E = Electronics, F = Failure, G = Good, L = Left, P = Pitch, R = Right, S = Servo, PEM = Pre-Engage Monitor, C_{LE} = Comparator Status of Left Electronics

TABLE A-2. SELF-TEST TABLE

Test No.	Channel	S.T. STIMULI #1	#2	#3	#4	#5 Sync Disable	#6 Inhibit Master Reset	Comparator PEM	1 = FAIL L_E	R_E	L_S	R_S	Monitoring Point	Function Under Test	Failure Report Out
5						0		1					PEM	F 0 - (A) - 0 (B)	P PEM
	A												CLS	GO	LS
	B	0							1	1			CRE	F - 0	RE
	C												CRE	GO	RE
6						0		1					PEM	F 0 - (C) - 0 (A)	PEM
	A												CRE	GO	RE
	B												CLS	GO	LS
	C	0							1	1			CRE	F - 0	RE
7													PEM	G Sync+(B)(C)	PEM
	A								1	1			CLE	F 0 +	LE
	B		0										CRE	G +	RE
	C		0										CLE	G +	LE
8													PEM	G Sync+(C)(A)	PEM
	A		0										CLE	G +	LE
	B								1	1			CLE	F 0 +	LE
	C		0										CRE	G +	RE
9													PEM	G Sync+(A)(B)	PEM
	A		0										CRE	G +	RE
	B		0										CLE	G +	LE
	C								1	1			CLE	F 0 +	P LE

NOTE: C = Comparator Status, E = Electronics, F = Failure, G = Good, L = Left, P = Pitch, R = Right, S = Servo, PEM = Pre-Engage Monitor, C_{LE} = Comparator Status of Left Electronics

TABLE A-3. SELF-TEST TABLE

Test No.	Channel	S.T. STIMULI #1	#2	#3	#4	#5 Sync Disable	#6 Inhibit Master Reset	Comparator PEM	1 = FAIL L_E	R_E	L_S	R_S	Monitoring Point	Function Under Test	Failure Report Out
10													PEM	G Sync-(B)(C)	P PEM
	A								1	1			CRE	F 0 -	RE
	B	0											CLE	G -	LE
	C	0											CRE	G -	RE
11													PEM	G Sync-(C)(A)	PEM
	A	0											CRE	G -	RE
	B								1	1			CRE	F 0 -	RE
	C	0											CLE	G -	LE
12													PEM	G Sync-(A)(B)	PEM
	A	0											CLE	G -	LE
	B	0											CRE	G -	RE
	C								1	1			CRE	F 0 -	RE
13						0	0						PEM	G INHIBIT	PEM
	A		C						1	1			CLE	F	LE
	B												CLE	G	LE
	C												CRE	G	RE
14					0								PEM	G	PEM
	A	0											CLE	G +	LE
	B	0											CLE	G +	LE
	C	0											CLE	G +	P LE

NOTE: C = Comparator Status, E = Electronics, F = Failure, G = Good, L = Left, P = Pitch, R = Right, S = Servo, PEM = Pre-Engage Monitor, C_{LE} = Comparator Status of Left Electronics

TABLE A-4. SELF-TEST TABLE

Test No.	Channel	S.T. STIMULI #1	#2	#3	#4	#5 Sync Disable	#6 Inhibit Master Reset	Comparator PEM	1 = FAIL L_E	R_E	L_S	R_S	Monitoring Point	Function Under Test	Failure Report Out
15						0							PEM	G	P PEM
	A								1	1			CLE	F 0 +	LE
	B	0											CLS	G +	LS
	C	0		0							1		CLS	F + 0	LS
16						0	0						PEM	G	PEM
	A			0					1	1	1		CRS	G +	RS
	B	0									1		CLS	F 2/3 +	LS
	C	0			0						1	1	CRS	F + 0	RS
17						0	0						PEM	G	PEM
	A				0				1	1	1	1	CRE	F 0 +	RE
	B	0									1	1	CRS	F 2/3 +	RS
	C	0									1	1	CRE	G +	RE
18						0							PEM	G	PEM
	A		0						1	1			CLE	F - 0	LE
	B												CLS	GO	LS
	C	0							1	1			CRE	F + 0	RE
19						0							PEM	G	PEM
	A	0		0							1		CLS	F + 0	LS
	B								1	1			CLE	F 0 +	LE
	C	0											CLS	G +	P LS

NOTE: C = Comparator Status, E = Electronics, F = Failure, G = Good, L = Left, P = Pitch, R = Right, S = Servo, PEM = Pre-Engage Monitor, C_{LE} = Comparator Status of Left Electronics

TABLE A-5. SELF-TEST TABLE

Test No.	Channel	S.T. STIMULI #1	#2	#3	#4	#5 Sync Disable	#6 Inhibit Master Reset	Comparator PEM	1 = FAIL L_E	R_E	L_S	R_S	Monitoring Point	Function Under Test	Failure Report Out
20						0	0						PEM	G	P PEM
	A	0			0						1	1	CRS	F + 0	RS
	B			0					1	1	1		CRS	G +	RS
	C	0									1		CLS	F 2/3 +	RS
21						0	0						PEM	G	PEM
	A	0									1	1	CRE	G +	RE
	B				0				1	1	1	1	CRE	F 0 +	RE
	C	0									1	1	CRS	F 2/3 +	RS
22						0							PEM	G	PEM
	A	0							1	1			CRE	F + 0	RE
	B		0						1	1			CLE	F - 0	LE
	C												CLS	GO	LS
23						0							PEM	G	PEM
	A	0											CLS	G +	LS
	B	0		0							1		CLS	F + 0	LS
	C								1	1			CLE	F 0 +	LE
24						0	0						PEM	G	PEM
	A	0									1		CLS	F 2/3 +	LS
	B	0			0						1	1	CRS	F + 0	RS
	C			0					1	1	1		CRS	G +	P RS

NOTE: C = Comparator Status, E = Electronics, F = Failure, G = Good, L = Left, P = Pitch, R = Right, S = Servo, PEM = Pre-Engage Monitor, C_{LE} = Comparator Status of Left Electronics

TABLE A-6. SELF-TEST TABLE

Test No.	Channel	S.T. STIMULI #1	#2	#3	#4	#5 Sync Disable	#6 Inhibit Master Reset	Comparator PEM	1 = FAIL L_E	R_E	L_S	R_S	Monitoring Point	Function Under Test	Failure Report Out
25						0	0						PEM	G	P PEM
	A	0									1	1	CRS	F 2/3 +	RS
	B	0									1	1	CRE	G +	RE
	C				0				1	1	1	1	CRE	F 0 +	RE
26						0							PEM	G	PEM
	A												CLS	GO	LS
	B	0							1	1			CRE	F + 0	RE
	C		0						1	1			CLE	F - 0	LE
27						0							PEM	G	PEM
	A		0										CRE	G -	RE
	B		0										CRE	G -	RE
	C		0										CRE	G -	RE
28						0							PEM	G	PEM
	A								1	1			CLE	F 0 -	LE
	B		0										CLS	G -	LS
	C		0	0							1		CLS	F - 0	LS
29						0	0						PEM	G	PEM
	A			0					1	1	1		CRS	G -	RS
	B		0								1		CLS	F 2/3 -	LS
	C		0		0						1	1	CRS	F - 0	P RS

NOTE: C = Comparator Status, E = Electronics, F = Failure, G = Good, L = Left, P = Pitch, R = Right, S = Servo, PEM = Pre-Engage Monitor, C_{LE} = Comparator Status of Left Electronics

TABLE A-7. SELF-TEST TABLE

Test No.	Channel	S.T. STIMULI #1	#2	#3	#4	#5 Sync Disable	#6 Inhibit Master Reset	Comparator PEM	1 = FAIL L_E	R_E	L_S	R_S	Monitoring Point	Function Under Test	Failure Report Out
30						0	0						PEM	G	P PEM
	A				0				1	1	1	1	CRE	F 0 -	RE
	B		0								1	1	CRS	F 2/3 -	RS
	C		0								1	1	CLE	G -	LE
31						0							PEM	G	PEM
	A		0						1	1			CRE	F - 0	RE
	B												CRS	GO	RS
	C	0							1	1			CLE	F + 0	LE
32						0							PEM	G	PEM
	A		0	0							1		CLS	F - 0	LS
	B								1	1			CLE	F 0 -	LE
	C		0										CLS	G -	LS
33						0	0						PEM	G	PEM
	A		0		0						1	1	CRS	F - 0	RS
	B			0					1	1	1		CRS	G -	RS
	C		0								1		CLS	F 2/3 -	LS
34						0	0						PEM	G	PEM
	A		0								1	1	CLE	G -	LE
	B				0				1	1	1	1	CRE	F 0 -	RE
	C		0								1	1	CRS	F 2/3 -	P RS

NOTE: C = Comparator Status, E = Electronics, F = Failure, G = Good, L = Left, P = Pitch, R = Right, S = Servo, PEM = Pre-Engage Monitor, C_{LE} = Comparator Status of Left Electronics

TABLE A-8. SELF-TEST TABLE

Test No.	Channel	S.T. STIMULI #1	#2	#3	#4	#5 Sync Disable	#6 Inhibit Master Reset	Comparator PEM	1 = FAIL L_E	R_E	L_S	R_S	Monitoring Point	Function Under Test	Failure Report Out
35						0							PEM	G	P PEM
	A	0							1	1			CLE	F + 0	LE
	B		0						1	1			CRE	F - 0	RE
	C												CRS	GO	RS
36						0							PEM	G	PEM
	A		0										CLS	G -	LS
	B		0	0							1		CLS	F - 0	LS
	C								1	1			CLE	F 0 -	LE
37						0	0						PEM	G	PEM
	A		0								1		CLS	F 2/3 -	LS
	B		0		0						1	1	CRS	F - 0	RS
	C			0					1	1	1		CRS	G -	RS
38						0	0						PEM	G	PEM
	A		0								1	1	CRS	F 2/3 -	RS
	B		0								1	1	CLE	G -	LE
	C				0				1	1	1	1	CRE	F 0 -	RE
39						0							PEM	G	PEM
	A												CRS	GO	RS
	B	0								1	1		CLE	F + 0	LE
	C		0							1	1		CRE	F - 0	P RE

NOTE: C = Comparator Status, E = Electronics, F = Failure, G = Good, L = Left, P = Pitch, R = Right, S = Servo, PEM = Pre-Engage Monitor, C_{LE} = Comparator Status of Left Electronics

TABLE A-9. SELF-TEST TABLE

Test No.	Channel	S.T. STIMULI #1	#2	#3	#4	#5 Sync Disable	#6 Inhibit Master Reset	Comparator PEM	1 = FAIL L_E	R_E	L_S	R_S	Monitoring Point	Function Under Test	Failure Report Out
40						0	0						PEM	GO	R PEM
	A												CLE	GO	LE
	B												CRE	GO	RE
	C												CLE	GO	LE
41						0	1						PEM	F 0 + (A) + 0 (B)	PEM
	A		0		0					1			CLE	Symm G Cancel Asym	LE
	B												CLE	GO	LE
	C												CRS	GO	RS
42						0	1						PEM	F 0 + (B) + 0 (C)	PEM
	A												CRS	GO	RS
	B		0		0					1			CLE	G	LE
	C												CLE	GO	LE
43						0	1						PEM	F 0 + (C) + 0 (A)	PEM
	A												CLE	GO	LE
	B												CRS	GO	RS
	C		0		0					1			CLE	G	R LE

NOTE: C = Comparator Status, E = Electronics, F = Failure, G = Good, L = Left, R = Roll or Right, S = Servo, PEM = Pre-Engage Monitor, RRS = Roll Right Servo

TABLE A-10. SELF-TEST TABLE

Test No.	Channel	S.T. STIMULI #1	#2	#3	#4	#5 Sync Disable	#6 Inhibit Master Reset	Comparator PEM	1 = FAIL L_E	R_E	L_S	R_S	Monitoring Point	Function Under Test	Failure Report Out
44						0	1						PEM	F 0 - (A) - 0 (B)	R PEM
	A	0		0						1			CLE	G	LE
	B												CRE	GO	RE
	C												CLS	GO	LS
45						0	1						PEM	F 0 - (B) - 0 (C)	PEM
	A												CLS	GO	LS
	B	0		0						1			CLE	G	LE
	C												CRE	GO	RE
46						0	1						PEM	F 0 - (C) - 0 (A)	PEM
	A												CRE	GO	RE
	B												CLS	GO	LS
	C	0		0						1			CLE	G	LE
47													PEM	G Sync+(B) (C)	PEM
	A								1	1			CLE	F 0 +	LE
	B	0											CRE	G +	RE
	C	0											CLE	G +	LE
48													PEM	G Sync+(C) (A)	PEM
	A	0											CLE	G +	LE
	B								1	1			CLE	F 0 +	LE
	C	0											CRE	G +	R RE

NOTE: C = Comparator Status, E = Electronics, F = Failure, G = Good, L = Left, R = Roll or Right, S = Servo, PEM = Pre-Engage Monitor, RRS = Roll Right Servo

TABLE A-11. SELF-TEST TABLE

Test No.	Channel	S.T. STIMULI #1	#2	#3	#4	#5 Sync Disable	#6 Inhibit Master Reset	Comparator PEM	1 = FAIL L_E	R_E	L_S	R_S	Monitoring Point	Function Under Test	Failure Report Out
49													PEM	G Sync+ (A) (B)	R PEM
	A	0											CRE	G +	RE
	B	0											CLE	G +	LE
	C								1	1			CLE	F 0 +	LE
50													PEM	G Sync- (B) (C)	PEM
	A								1	1			CRE	F 0 -	RE
	B		0										CLE	G -	LE
	C		0										CRE	G -	RE
51													PEM	G Sync- (C) (A)	PEM
	A		0										CRE	G -	RE
	B								1	1			CRE	F 0 -	RE
	C		0										CLE	G -	LE
52													PEM	G Sync- (A) (B)	PEM
	A		0										CLE	G -	LE
	B		0										CRE	G -	RE
	C								1	1			CRE	F 0 -	RE
53						0	0						PEM	G INHIBIT	PEM
	A												CRE	G	RE
	B	0							1	1			CRE	F	RE
	C												CRE	G	R RE

NOTE: C = Comparator Status, E = Electronics, F = Failure, G = Good, L = Left, R = Roll or Right, S = Servo, PEM = Pre-Engage Monitor, RRS = Roll Right Servo

TABLE A-12. SELF-TEST TABLE

Test No.	Channel	S.T. STIMULI #1	#2	#3	#4	#5 Sync Disable	#6 Inhibit Master Reset	Comparator PEM	1 = FAIL L_E	R_E	L_S	R_S	Monitoring Point	Function Under Test	Failure Report Out
54						0							PEM	G	R PEM
	A		0										CLE	G +	LE
	B		0										CLE	G +	LE
	C		0										CLE	G +	LE
55						0							PEM	C	PEM
	A								1	1			CLE	F 0 +	LE
	B		0										CLS	G +	LS
	C		0	0							1		CLS	F + 0	LS
56						0	0						PEM	G	PEM
	A			0					1	1	1		CRS	G +	RS
	B		0								1		CLS	F 2/3 +	LS
	C		0		0						1	1	CRS	F + 0	RS
57						0	0						PEM	G	PEM
	A				0				1	1	1	1	CRE	F 0 +	RE
	B		0								1	1	CRS	F 2/3 +	RS
	C		0								1	1	CRE	G +	RE
58						0							PEM	G	PEM
	A	0							1	1			CLE	F - 0	LE
	B												CLS	GO	LS
	C		0						1	1			CRE	F + 0	R RE

NOTE: C = Comparator Status, E = Electronics, F = Failure, G = Good, L = Left, R = Roll or Right, S = Servo, PEM = Pre-Engage Monitor, RRS = Roll Right Servo

TABLE A-13. SELF-TEST TABLE

Test No.	Channel	S.T. STIMULI #1	#2	#3	#4	#5 Sync Disable	#6 Inhibit Master Reset	Comparator PEM	1 = FAIL L_E	R_E	L_S	R_S	Monitoring Point	Function Under Test	Failure Report Out
59						0							PEM	G	R PEM
	A		0	0							1		CLS	F + 0	LS
	B								1	1			CLE	F 0 +	LE
	C		0										CLS	G +	LS
60						0	0						PEM	G	PEM
	A		0		0						1	1	CRS	F + 0	RS
	B			0					1	1	1		CRS	G +	RS
	C		0								1		CLS	F 2/3 +	LS
61						0	0						PEM	G	PEM
	A		0								1	1	CRE	G +	RE
	B				0				1	1	1	1	CRE	F 0 +	RE
	C		0								1	1	CRS	F 2/3 +	RS
62						0							PEM	G	PEM
	A		0						1	1			CRE	F + 0	RE
	B	0							1	1			CLE	F - 0	LE
	C												CLS	GO	LS
63						0							PEM	G	PEM
	A		0										CLS	G +	LS
	B		0	0							1		CLS	F + 0	LS
	C								1	1			CLE	F 0 +	R LE

NOTE: C = Comparator Status, E = Electronics, F = Failure, G = Good, L = Left, R = Roll or Right, S = Servo, PEM = Pre-Engage Monitor, RRS = Roll Right Servo

TABLE A-14. SELF-TEST TABLE

Test No.	Channel	S.T. STIMULI #1	#2	#3	#4	#5 Sync Disable	#6 Inhibit Master Reset	Comparator PEM	1 = FAIL L_E	R_E	L_S	R_S	Monitoring Point	Function Under Test	Failure Report Out
64						0	0						PEM	G	R PEM
	A		0								1		CLS	F 2/3 +	LS
	B		0		0						1	1	CRS	F + 0	RS
	C			0					1	1	1		CRS	G +	RS
65						0	0						PEM	G	PEM
	A		0								1	1	CRS	F 2/3 +	RS
	B		0								1	1	CRE	G +	RE
	C				0				1	1	1	1	CRE	F 0 +	RE
66						0							PEM	G	PEM
	A												CLS	GO	LS
	B		0						1	1			CRE	F + 0	RE
	C	0							1	1			CLE	F - 0	LE
67						0							PEM	G	PEM
	A	0											CRE	G -	RE
	B	0											CRE	G -	RE
	C	0											CRE	G -	RE
68						0							PEM	G	PEM
	A								1	1			CLE	F 0 -	LE
	B	0											CLS	G -	LS
	C	0		0							1		CLS	F - 0	R LS

NOTE: C = Comparator Status, E = Electronics, F = Failure, G = Good, L = Left, R = Roll or Right, S = Servo
PEM = Pre-Engage Monitor, RRS = Roll Right Servo

TABLE A-15. SELF-TEST TABLE

Test No.	Channel	S.T. STIMULI #1	#2	#3	#4	#5 Sync Disable	#6 Inhibit Master Reset	Comparator PEM	1 = FAIL L_E	R_E	L_S	R_S	Monitoring Point	Function Under Test	Failure Report Out
69						0	0						PEM	G	R PEM
	A			0					1	1	1		CRS	G -	RS
	B	0									1		CLS	F 2/3 -	LS
	C	0			0						1	1	CRS	F - 0	RS
70						0	0						PEM	G	PEM
	A				0				1	1	1	1	CRE	F 0 -	RE
	B	0									1	1	CRS	F 2/3 -	RS
	C	0									1	1	CLE	G -	LE
71						0							PEM	G	PEM
	A	0							1	1			CRE	F - 0	RE
	B												CRS	G)	RS
	C		0						1	1			CLE	F + 0	LE
72						0							PEM	G	PEM
	A	0		0							1		CLS	F - 0	LS
	B								1	1			CLE	F 0 -	LE
	C	0											CLS	G -	LS
73						0	0						PEM	G	PEM
	A	0			0						1	1	CRS	F - 0	RS
	B			0					1	1	1		CRS	G -	RS
	C	0									1		CLS	F 2/3 -	R LS

NOTE: C = Comparator Status, E = Electronics, F = Failure, G = Good, L = Left, R = Roll or Right, S = Servo, PEM = Pre-Engage Monitor, RRS = Roll Right Servo

TABLE A-16. SELF-TEST TABLE

Test No.	Channel	S.T. STIMULI #1	#2	#3	#4	#5 Sync Disable	#6 Inhibit Master Reset	Comparator PEM	1 = FAIL L_E	R_E	L_S	R_S	Monitoring Point	Function Under Test	Failure Report Out
74						0	0						PEM	G	R PEM
	A	0									1	1	CLE	G -	LE
	B				0				1	1	1	1	CRE	F 0 -	RE
	C	0									1	1	CRS	F 2/3 -	RS
75						0							PEM	G	PEM
	A		0						1	1			CLE	F + 0	LE
	B	0							1	1			CRE	F - 0	RE
	C												CRS	GO	RS
76						0							PEM	G	PEM
	A	0											CLS	G -	LS
	B	0		0							1		CLS	F - 0	LS
	C								1	1			CLE	F 0 -	LE
77						0	0						PEM	G	PEM
	A	0									1		CLS	F 2/3 -	LS
	B	0			0						1	1	CRS	F - 0	RS
	C			0					1	1	1		CRS	G -	RS
78						0	0						PEM	G	PEM
	A	0									1	1	CRS	F 2/3 -	RS
	B	0									1	1	CLE	G -	LE
	C				0				1	1	1	1	CRE	F 0 -	R RE

NOTE: C = Comparator Status, E = Electronics, F = Failure, G = Good, L = Left, R = Roll or Right, S = Servo, PEM = Pre-Engage Monitor, RRS = Roll Right Servo

TABLE A-17. SELF-TEST TABLE

Test No.	Channel	S.T. STIMULI						Comparator					Monitoring Point	Function Under Test	Failure Report Out
		#1	#2	#3	#4	#5 Sync Disable	#6 Inhibit Master Reset	PEM	1 = FAIL L_E	R_E	L_S	R_S			
79						0							PEM	G	R PEM
	A												CRS	GO	RS
	B		0						1	1			CLE	F + 0	LE
	C	0							1	1			CRE	F - 0	R RE

NOTE: C = Comparator Status, E = Electronics, F = Failure, G = Good, S = Servo, Y = Yaw, PEM = Pre-Engage Monitor

TABLE A-18. SELF-TEST TABLE

Test No.	Channel	S.T. STIMULI #1	#2	#3	#4	#5 Sync Disable	#6 Inhibit Master Reset	Comparator PEM	1 = FAIL E		S		Monitoring Point	Function Under Test	Failure Report Out
80						0	0						PEM	GO	Y PEM
	A												CE	GO	E
	B												CE	GO	E
	C												CE	GO	E
81						0		1					PEM	F 0 + (A) + 0 (B)	PEM
	A		0						1				CE	F + 0	E
	B												CE	GO	E
	C												CS	GO	S
82						0		1					PEM	F 0 + (B) + 0 (C)	PEM
	A												CS	GO	S
	B		0						1				CE	F + 0	E
	C												CE	GO	E
83						0		1					PEM	F 0 + (C) + 0 (A)	PEM
	A												CE	GO	E
	B												CS	GO	S
	C		0						1				CE	F + 0	Y E

NOTE: C = Comparator Status, E = Electronics, F = Failure, G = Good, S = Servo, Y = Yaw,
PEM = Pre-Engage Monitor

TABLE A-19. SELF-TEST TABLE

Test No.	Channel	S.T. STIMULI #1	#2	#3	#4	#5 Sync Disable	#6 Inhibit Master Reset	Comparator PEM	1 = FAIL E		S		Monitoring Point	Function Under Test	Failure Report Out
84						0		1					PEM	F 0 - (A) - 0 (B)	Y PEM
	A	0							1				CE	F - 0	E
	B												CE	GO	E
	C												CS	GO	S
85						0		1					PEM	F 0 - (B) - 0 (C)	PEM
	A												CS	GO	S
	B	0							1				CE	F - 0	E
	C												CE	GO	E
86						0		1					PEM	F 0 - (C) - 0 (A)	PEM
	A												CE	GO	E
	B												CS	GO	S
	C	0							1				CE	F - 0	E
87													PEM	G Sync+(B)(C)	PEM
	A								1				CE	F 0 +	E
	B		0										CE	G +	E
	C		0										CS	G +	S
88													PEM	G Sync+(C)(A)	PEM
	A		0										CS	G +	S
	B								1				CE	F 0 +	E
	C		0										CE	G +	Y E

NOTE: C = Comparator Status, E = Electronics, F = Failure, G = Good, S = Servo, Y = Yaw, PEM = Pre-Engage Monitor

TABLE A-20. SELF-TEST TABLE

Test No.	Channel	S.T. STIMULI #1	#2	#3	#4	#5 Sync Disable	#6 Inhibit Master Reset	Comparator PEM	1 = FAIL E		S		Monitoring Point	Function Under Test	Failure Report Out
89													PEM	G Sync+$^{(A)}_{(B)}$	Y PEM
	A		0										CE	G +	E
	B		0										CS	G +	S
	C								1				CE	F 0 +	E
90													PEM	G Sync-$^{(B)}_{(C)}$	PEM
	A								1				CE	F 0 -	E
	B	0											CE	G -	E
	C	0											CS	G -	S
91													PEM	G Sync-$^{(C)}_{(A)}$	PEM
	A	0											CS	G -	S
	B								1				CE	F 0 -	E
	C	0											CE	G -	E
92													PEM	G Sync-$^{(A)}_{(B)}$	PEM
	A	0											CE	G -	E
	B	0											CS	G -	S
	C								1				CE	F 0 -	E
93						0	0						PEM	G INHIBIT	PEM
	A		0						1				CE	F	E
	B												CE	G	E
	C												CE	G RESET	Y E

NOTE: C = Comparator Status, E = Electronics, F = Failure, G = Good, S = Servo, Y = Yaw, PEM = Pre-Engage Monitor

TABLE A-21. SELF-TEST TABLE

Test No.	Channel	S.T. STIMULI						Comparator					Monitoring Point	Function Under Test	Failure Report Out
		#1	#2	#3	#4	#5 Sync Disable	#6 Inhibit Master Reset	PEM	1 = FAIL E		S				
94						0							PEM	G	Y PEM
	A								1				CE	F 2/3	E
	B	0							1				CE	F + 0	E
	C		0						1				CE	F − 0	E
95						0							PEM	G	PEM
	A								1				CE	F 0 +	E
	B	0											CS	G +	S
	C	0			0						1		CS	F $^{+}_{+}$ +	S
96						0	0						PEM	G	PEM
	A				0				1		1		CS	F $^{+}_{+}$ +	S
	B	0									1		CS	F 2/3 +	S
	C	0									1		CE	G +	E
97						0							PEM	G	PEM
	A	0			0						1		CS	F $^{+}_{+}$ +	S
	B								1				CE	F 0 +	E
	C	0											CS	G +	S
98						0	0						PEM	G	PEM
	A	0									1		CE	G +	E
	B				0				1		1		CS	F $^{+}_{+}$ +	S
	C	0									1		CS	F 2/3 +	Y S

NOTE: C = Comparator Status, E = Electronics, F = Failure, G = Good, S = Servo, Y = Yaw
PEM = Pre-Engage Monitor

TABLE A-22. SELF-TEST TABLE

Test No.	Channel	S.T. STIMULI #1	#2	#3	#4	#5 Sync Disable	#6 Inhibit Master Reset	Comparator PEM	1 = FAIL E		S		Monitoring Point	Function Under Test	Failure Report Out
99						0							PEM	G	Y PEM
	A	0											CS	G +	S
	B	0			0						1		CS	F + + +	S
	C								1				CE	F 0 +	E
100						0	0						PEM	G	PEM
	A	0									1		CS	F 2/3 +	S
	B	0									1		CE	G +	E
	C				0				1		1		CS	F + + +	S
101						0							PEM	G	PEM
	A		0						1				CE	F + 0	E
	B								1				CE	F 2/3	E
	C	0							1				CE	F + 0	E
102						0							PEM	G	PEM
	A								1				CE	F 0 -	E
	B		0										CS	G - -	S
	C		0		0						1		CS	F - - -	S
103						0	0						PEM	G -	PEM
	A				0				1		1		CS	F - - -	S
	B		0								1		CS	F 2/3 -	S
	C		0								1		CE	G -	Y E

NOTE: C = Comparator Status, E = Electronics, F = Failure, G = Good, S = Servo, Y = Yaw,
PEM = Pre-Engage Monitor

TABLE A-23. SELF-TEST TABLE

Test No.	Channel	S.T. STIMULI #1	#2	#3	#4	#5 Sync Disable	#6 Inhibit Master Reset	Comparator PEM	1 = FAIL E		S		Monitoring Point	Function Under Test	Failure Report Out
104						0							PEM	G	Y PEM
	A		0		0						1		CS	F = -	S
	B								1				CE	F 0 -	E
	C		0										CS	G -	S
105						0	0						PEM	G	PEM
	A		0								1		CE	G =	E
	B				0				1		1		CS	F = -	S
	C		0								1		CS	F 2/3 -	S
106						0							PEM	G	PEM
	A		0										CS	G =	S
	B		0		0						1		CS	F = -	S
	C								1				CE	F 0 -	E
107						0	0						PEM	G	PEM
	A		0								1		CS	F 2/3 -	S
	B		0								1		CE	G =	E
	C				0				1		1		CS	F = -	S
108						0							PEM	G	PEM
	A	0							1				CE	F + 0	E
	B		0						1				CE	F - 0	E
	C								1				CE	F 2/3	Y E

NOTE: C = Comparator Status, E = Electronics, F = Failure, G = Good, S = Servo, Y = Yaw, PEM = Pre-Engage Monitor

REPRODUCIBILITY OF THE ORIGINAL PAGE IS POOR

TABLE A-24. SELF-TEST TABLE

Test No.	Channel	S.T. STIMULI #1	#2	#3	#4	#5 Sync Disable	#6 Inhibit Master Reset	Comparator PEM	1 = FAIL E		S		Monitoring Point	Function Under Test	Failure Report Out
109						0							PEM	G	Y PEM
	A												CE	GO	E
	B												CE	GO	E
	C												CE	GO	E
110						0							PEM	G	PEM
	A												CS	GO	S
	B												CS	GO	S
	C												CS	GO	S
111						0							PEM	G	PEM
	A												CE	GO	E
	B												CE	GO	E
	C												CE	GO	Y E

NOTE: C = Comparator Status, E = Electronics, F = Failure, G = Good, S = Servo, Y = Yaw, PEM = Pre-Engage Monitor

APPENDIX B

COMPARATOR STATUS CHECK TABLE FOR SELF-TEST

The comparator status check tables that follow summarize the results of the tests in Appendix A. For an example, the case for "Pitch Fail +0" is interpreted in the following paragraphs. Other items of the table have a similar interpretation.

1. "Pitch Fail +0" means that the comparator should indicate a failure due to a positive polarity at the MVL input with respect to the voltage at the MVL output.

2. "MVL Input +, Output 0" means that a single + polarity voltage is applied and a 0 voltage is generated at the output of the MVL.

"MVL Input ++, Output +" means that a positive hardover voltage was applied and the output was a lower level positive voltage.

3. The notation "Channel A Pitch DCS 3" means that the subject comparator trip occurred in test 3 of Appendix A, resulting in a PEM failure indication. Similarly, the notation "Channel A Pitch DCS C" means that the test is not performed in the DCS mode, since it is already covered in CBS mode.

TABLE B-1. COMPARATOR STATUS CHECK TABLE - PITCH AXIS

COMPARATOR STATUS CHECK TABLE - PITCH AXIS

Pitch	MVL		Channel A Pitch									Channel B Pitch									Channel C Pitch								
	Input	Output	DCS					CBS				DCS					CBS				DCS					CBS			
			C_{PEM}	C_{LE}	C_{RE}	C_{LS}	C_{RS}	C_{LE}	C_{RE}	C_{LS}	C_{RS}	C_{PEM}	C_{LE}	C_{RE}	C_{LS}	C_{RS}	C_{LE}	C_{RE}	C_{LS}	C_{RS}	C_{PEM}	C_{LE}	C_{RE}	C_{LS}	C_{RS}	C_{LE}	C_{RE}	C_{LS}	C_{RS}
Good 0	0	0	0	3	6	5	2	D	D	26	39	0	1	4	6	3	D	D	18	31	0	2	5	4	1	D	D	22	35
Good +	+	+	9	8	9	C	C	14	21	23	16	7	9	7	C	C	14	25	15	20	8	7	8	C	C	14	17	19	24
Good -	-	-	12	12	11	C	C	34	27	36	29	10	10	12	C	C	38	27	28	33	11	11	10	C	C	30	27	32	37
Sync +	N/A	N/A	8	← N/A →								9	← N/A →								7	← N/A →							
Sync -	N/A	N/A	11	← N/A →								12	← N/A →								10	← N/A →							
Fail +0	+ ++	0 +	3	1	C	C	C	35	22	19	20	1	2	C	C	C	39	26	23	24	2	3	C	C	C	31	18	15	16
Fail 0+	0	+	1	7	C	N/A		15	17	N/A		2	8	C	N/A		19	21	N/A		3	9	C	N/A		23	25	N/A	
Fail -0	- --	0 -	6	C	4	C	C	18	31	32	33	4	C	5	C	C	22	35	36	37	5	C	6	C	C	26	39	28	29
Fail 0-	0	-	4	C	10	N/A		28	30	N/A		5	C	11	N/A		32	34	N/A		6	C	12	N/A		36	38	N/A	
2 out of 3 Failure +	N/A	N/A	← N/A →			C	C	N/A		24	25	← N/A →			C	C	N/A		16	17	← N/A →			C	C	N/A		20	21
2 out of 3 Failure -	N/A	N/A	← N/A →			C	C	N/A		37	38	← N/A →			C	C	N/A		29	30	← N/A →			C	C	N/A		33	34
NOTE: C = Checked in CBS Mode, D = Checked in DCS Mode, N/A = Not Applicable																													

TABLE B-2. COMPARATOR STATUS CHFCK TABLE - ROLL AXIS

COMPARATOR STATUS CHECK TABLE - ROLL AXIS

Roll	MVL		Channel A Roll									Channel B Roll									Channel C Roll								
	Input	Output	DCS					CBS				DCS					CBS				DCS					CBS			
			C_{PEM}	C_{LE}	C_{RE}	C_{LS}	C_{RS}	C_{LE}	C_{RE}	C_{LS}	C_{RS}	C_{PEM}	C_{LE}	C_{RE}	C_{LS}	C_{RS}	C_{LE}	C_{RE}	C_{LS}	C_{RS}	C_{PEM}	C_{LE}	C_{RE}	C_{LS}	C_{RS}	C_{LE}	C_{RE}	C_{LS}	C_{RS}
Good 0	0	0	40	43	46	45	42	D	D	66	79	40	41	44	46	43	D	D	58	71	40	42	45	44	41	D	D	62	75
Good +	+	+	49	48	49	C	C	54	61	63	56	47	49	47	C	C	54	65	55	60	48	47	48	C	C	54	57	59	64
Good -	-	-	52	52	51	C	C	74	67	76	69	50	50	52	C	C	78	67	68	73	51	51	50	C	C	70	67	72	77
Sync +	N/A	N/A	48	◄— N/A —►								49	◄— N/A —►								47	◄— N/A —►							
Sync -	N/A	N/A	51	◄— N/A —►								52	◄— N/A —►								50	◄— N/A —►							
Fail +0	+ ++	0 +	43	C	C	C	C	75	62	59	60	41	C	C	C	C	79	66	63	64	42	C	C	C	C	71	58	55	56
Fail 0+	0	+	41	47	C	N/A		55	57	N/A		42	48	C	N/A		59	61	N/A		43	49	C	N/A		63	65	N/A	
Fail -0	- --	0 -	46	C	C	C	C	58	71	72	73	44	C	C	C	C	62	75	76	77	45	C	C	C	C	66	79	68	69
Fail 0-	0	-	44	C	50	N/A		68	70	N/A		45	C	51	N/A		72	74	N/A		46	C	52	N/A		76	78	N/A	
2 out of 3 Failure +	N/A	N/A	◄— N/A —►			C	C	N/A		64	65	◄— N/A —►			C	C	N/A		56	57	◄— N/A —►			C	C	N/A		60	61
2 out of 3 Failure -	N/A	N/A	◄— N/A —►			C	C	N/A		77	78	◄— N/A —►			C	C	N/A		69	70	◄— N/A —►			C	C	N/A		73	74

NOTE: C = Checked in CBS Mode, D = Checked in DCS Mode, N/A = Not Applicable

TABLE B-3. COMPARATOR STATUS CHECK TABLE - YAW AXIS

COMPARATOR STATUS CHECK TABLE - YAW AXIS

Yaw	MVL		Channel A - Yaw					Channel B - Yaw					Channel C - Yaw				
	Input	Output	DCS			CBS		DCS			CBS		DCS			CBS	
			C_{PEM}	C_E	C_S	C_E	C_S	C_{PEM}	C_E	C_S	C_E	C_S	C_{PEM}	C_E	C_S	C_E	C_S
Good 0	0	0	80	83	82	109	110	80	81	83	109	110	80	82	81	109	110
Good +	+	+	89	89	88	98	99	87	87	89	100	95	88	88	87	96	97
Good -	-	-	92	92	91	105	106	90	90	92	107	102	91	91	90	103	104
Sync +	N/A	N/A	88	← N/A →				89	← N/A →				87	← N/A →			
Sync -	N/A	N/A	91	← N/A →				92	← N/A →				90	← N/A →			
Fail +0	+ ++	0 +	83	81	C	108	96	81	32	C	94	98	82	83	C	101	95
Fail 0+	0	+	81	87	N/A	95	N/A	82	88	N/A	97	N/A	83	89	N/A	99	N/A
Fail -0	- --	-	86	84	C	101	103	84	85	C	108	105	85	86	C	94	102
Fail 0-	0	-	84	90	N/A	102	N/A	85	91	N/A	104	N/A	86	92	N/A	106	N/A
2 out of 3 Failure +	N/A	N/A	← N/A →		C	94	100	← N/A →		C	101	96	← N/A →		C	108	98
2 out of 3 Failure -	N/A	N/A	← N/A →		C	94	107	← N/A →		C	101	103	← N/A →		C	108	105
NOTE: C = Checked in CBS Mode, N/A = Not Applicable																	

1. Report No.	2. Government Accession No.	3. Recipient's Catalog No.
NASA CR-144865		

4. Title and Subtitle	5. Report Date
DESCRIPTION AND THEORY OF OPERATION OF THE COMPUTER BY-PASS SYSTEM FOR THE NASA F-8 DIGITAL FLY-BY-WIRE CONTROL SYSTEM	May 1978
	6. Performing Organization Code
7. Author(s)	8. Performing Organization Report No.
	10. Work Unit No.
9. Performing Organization Name and Address	512-51-04
Sperry Flight Systems P.O. Box 21111 Phoenix, Arizona 85036	11. Contract or Grant No. NAS4-2248
	13. Type of Report and Period Covered Contractor Report - Final
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D.C. 20546	14. Sponsoring Agency Code

15. Supplementary Notes

NASA Technical Monitor: Wilton P. Lock, Jr., Dryden Flight Research Center

16. Abstract

A triplex digital flight control system was installed in a NASA F-8C airplane to provide fail-operate, full authority control. The triplex digital computers and interface circuitry process the pilot commands and aircraft motion feedback parameters according to the selected control laws, and they output the surface commands as an analog signal to the servoelectronics for position control of the aircraft's power actuators.

A triplex analog backup control system (computer by pass) was also installed to provide additional redundancy and flight safety in case of a generic problem with the computer or software. The computer by-pass control system contains the servodrive electronics utilized for either digital computer or backup control.

This report describes the system and theory of operation of the computer by-pass and servoelectonics. It includes an automated ground test for each axis.

The F-8 secondary actuators are triplex force-summing analog servos having both large force capability and large system bandwidth. The servovalves are of the two-stage electrohydraulic flapper type. The secondary actuator position loop is the dominant control loop and includes pressure equalization utilizing the differential pressure developed across each piston.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
Fly-by-wire systems Triplex servoactuators Pressure equalization F-8 digital fly-by-wire aircraft	Unclassified - Unlimited STAR Category: 05

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price*
Unclassified	Unclassified	102	

*For sale by the National Technical Information Service, Springfield, Virginia 22161