Synthesis of a FBW System for an F-4

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The control laws for use in a fly-by-wire (FBW) flight control system to be flight tested in an F-4 aircraft have been developed. The synthesis of the FBW system required defining the control laws on a single channel basis to achieve the desired handling qualities. Frequency response, root locus, and transient response techniques were used as appropriate. The linear and nonlinear mathematical models used to establish the system design, including the flexible airframe models used to define structural coupling, are described. The results of the analyses and stimulations used to help define the system are presented.

Introduction

THE Survivable Flight Control System (SFCS) program is an advanced development program being conducted by McDonnell Aircraft Company (MCAIR) under contract to the Air Force Flight Dynamics Laboratory. The principal objective of this program is the development and flight test demonstration of an F-4 aircraft with a Survivable Flight Control System utilizing fly-by-wire techniques.

The SFCS is a three-axis, fly-by-wire primary flight control system which functions to command aircraft motion, instead of surface position, as a direct function of pilot applied inputs, as illustrated in Fig. 1. Improved stability characteristics are provided through the use of feedback control, proper placement of aircraft motion sensors, and the application of structural mode filters to attenuate aircraft resonant frequencies. The initial flight test of the SFCS included a mechanical back-up control system in the pitch and yaw axes. This back-up system was removed from the aircraft as soon as adequate system performance and reliability were demonstrated.

Quadruplex (four channel) redundancy is used in all feasible system components, including power supplies, to obtain improved system reliability and increased mission completion probability. The system is designed to sustain two similar failures per axis without significant degradation in performance.

The paper is limited to a discussion of the longitudinal and lateral-directional control law development studies, manned simulation results, and conclusions relative to these studies and simulations. Details of these studies and simulations are presented in Refs. 1 and 2.

The longitudinal and lateral-directional control laws were initially implemented to comply with the requirements of MIL-F-8785B(ASG) and the C^* criterion. As part of the contractive effort, a handling qualities criteria development program, reported in Ref. 3, was conducted concurrently with the control law development effort. Time history criteria were developed for each axis of control, including a modified C^* criteria with a dC^*/dt addition, a similar criteria for yaw, termed D^* and dD^*/dt, and a roll rate-roll oscillation criterion. The control law mechanization was subsequently evaluated against these criteria for handling qualities acceptance. The results of this evaluation, which show compliance in all essential parameters, are included in this paper.

A six-degree-of-freedom, man-in-the-loop simulation program was conducted to evaluate the control law implementation. This simulation included the capability to maneuver the aircraft through the F-4 flight envelope including stall and post stall conditions. As a result of this simulation, seven design modifications were identified and evaluated. These modifications were subsequently implemented into the SFCS design.

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Longitudinal Control Law Development

The functional block diagram presented in Fig. 2 depicts the SFCS longitudinal control system. A summary of the control law features as presently conceived is presented below.

Normal Mode

The longitudinal control system is a fly-by-wire system in which aircraft motion is the controlled parameter. This is accomplished by utilizing normal acceleration and pitch rate feedbacks which are subtracted from either the center stick force transducer commands or from the side stick position transducer commands to obtain an error signal. The error signal is used as a position command for the stabilator actuator.

The Normal mode of operation provides neutral speed stability (NSS) for all nonterminal flight phases. Since the speed stability is neutral, no steady-state, pilot-applied stick force or trim input is required to compensate for the change in stabilator position required to trim the airplane because of changes in airspeed and/or altitude. The NSS characteristic is obtained by generating an integration in the forward path of the control system. The integration maintains zero steady-state error between force command and the blended pitch rate and normal acceleration feedback. The airplane is kept in trim since any uncommanded pitch rate and normal acceleration is automatically reduced to zero by the action of the integrator. For nonterminal flight conditions, only occasional trim inputs initiated by the pilot are required to offset any electrical biases which may be present. A signal from the nose gear door switch removes the integration function when the landing gear is extended, and nominal trimming ac-

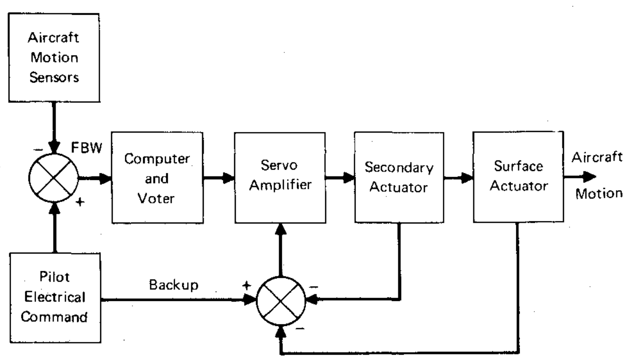

Fig. 1 FBW system mechanization.

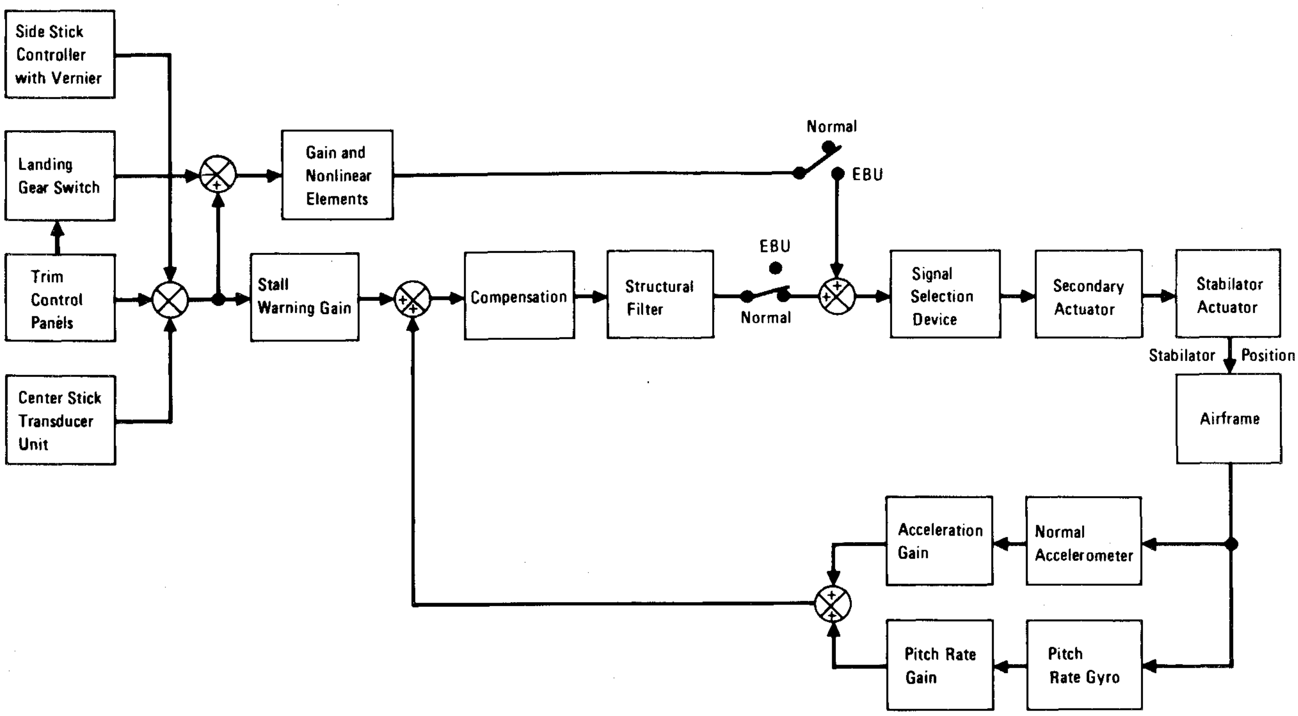

Fig. 2 Simplified functional block diagram single channel of longitudinal axis.

tion by the pilot is required during terminal flight phases. The pilot may also override the NSS Function with a switch on the trim panel.

The secondary actuator is used to provide the integration function. This is implemented by incorporating washout circuits in the secondary actuator ram position and main actuator ram position feedback signals, which transform the position feedbacks into velocity feedbacks for low frequency inputs ($\omega < 1$ rad/sec).

The washout network for the secondary actuator position feedback is mechanized immediately after the demodulator. For the Normal mode with gear up, the transfer function for the feedback circuitry is:

$$\frac{\text{Output}}{\text{Input}} = \frac{1}{1 + 1/S} = \frac{S}{S+1}$$

The demodulated stabilator actuator position feedback is passed through an identical washout of $S/(S + 1)$. Since the washout circuits reduce both electrical feedbacks to zero for steady-state secondary actuator and stabilator actuator position, any electrical input to the actuator loop is unopposed and is integrated by the secondary actuator. The feedback paths can attain nonzero, steady-state values only in the presence of constant secondary actuator and stabilator actuator rates. Thus the steady-state response to a constant input to the actuator loop is a constant stabilator rate. This method of implementing the NSS function was selected since the inherent integration characteristic of the secondary actuator can be utilized and the need for including a separate electronic integrator in the forward loop is avoided.

The input to the actuator loop must be zero to achieve a steady-state stabilator deflection in response to a steady-state pilot applied input. For center stick input, the steady-state control law is:

$$(\text{Pilot Force})\,(0.06)\,(2)\,[K(C_s)] + N_z(0.8)\,K(N_z) + q\,(0.12)\,K_q = 0$$

Substituting the values for $K(c_s)$, $K(N_Z)$ and K_q and using the steady-state equation for pitch rate, $q = (1845/V)\,N_Z$ gives the following expression:

$$(0.45 + \frac{184.5}{V})N_Z - 0.12(\text{Pilot Force}) = 0$$

In order for the above identity to hold, N_Z must be controlled to zero in the absence of pilot applied force. When an out-of-trim stabilator condition occurs, uncommanded N_Z is sensed, and the forward loop integration acts to reposition the stabilator, thereby re-establishing the above equality.

Electrical Back-Up (EBU) Mode

The control system includes an electric back-up (EBU) mode which consists of a simple forward path connecting the inputs from the longitudinal force pre-filters to the signal selection device (SSD). To minimize transients, selection of EBU by the pilot causes the direct electric path to be faded-in as the normal SFCS forward path is faded-open. The actuator main ram electrical feedback path is also simultaneously faded-open. Aircraft motion feedbacks, forward loop compensation, structural filters, adaptive gain changing, stabilator position feedback, and stall warning circuitry are all disabled in the EBU mode. However, the EBU is still a quadruplex system and the signal selection devices continue to function.

Adaptive Gain Changing

Pilot selection of adaptive gain changing or fixed gain operation is available. The adaptive gain changer provides three gain states in the longitudinal axis which change the forward loop gain, K_F, over a range of 4 to 1 as the aircraft stabilator effectiveness parameter, M_δ, varies with flight condition and aircraft configuration changes.

The adaptive gain computer provides "hysteresis loops" at the three gain change boundaries, and all gain changes are faded-in over a 2 sec period. After selecting the fixed gain position on the Master Control and Display Panel, the pilot may manually select any of the three gain states. The system is configured to provide stable operation over the entire flight envelope for the low gain value. By selecting fixed gain operation, the pilot may manually select any of the adaptive gains.

Stall Warning System

The normal F-4 audio tone generator provides pilot warning of impending one g and accelerated stalls. An additional stall warning system is provided with the SFCS to warn the pilot of impending accelerated stalls.

The stall warning system provides stick force cues to the pilot when the region of impending accelerated stall is entered. The warning becomes increasingly more pronounced with progression into the stall region. This is accomplished by decreasing the electrical gradient through which the stick force transducer and Side Stick Controller (SSC) outputs are passed prior to the point at which they are summed with the sensor feedback signals. Thus, the pilot-applied maneuvering force for either center stick or SSC must be increased to command a given maneuver if

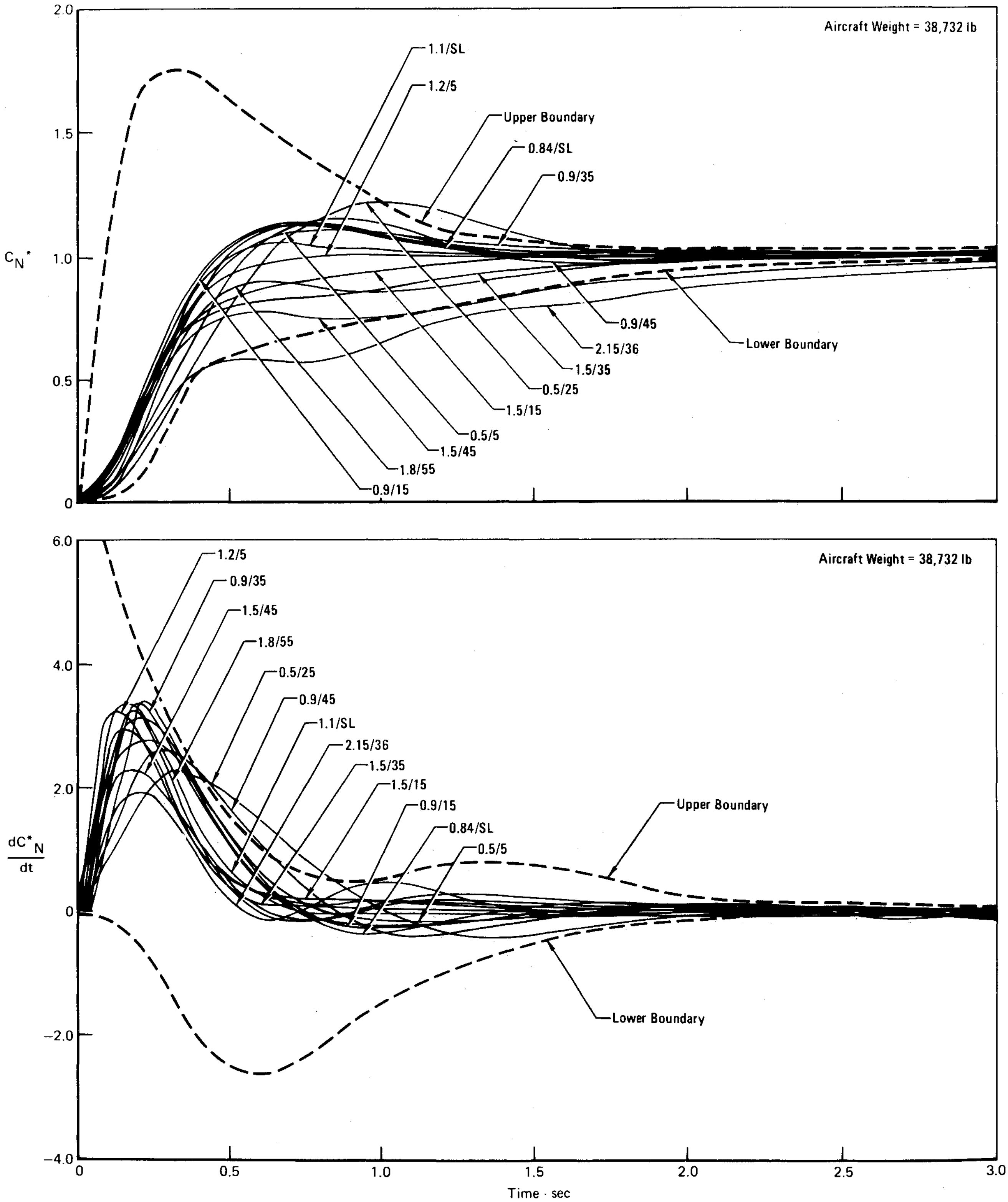

Fig. 3 Longitudinal SFCS time history C* criteria compliance adaptive gains–NSS.

the maneuver causes entry into the stall region. The warning signal is initiated at 20 units angle-of-attack and increased linearily to a maximum of 22.5 units angle-of-attack, at which point the stick force electrical gradient, K_{cs}, is correspondingly decreased to one-third its nominal value. Cancelled pitch rate is used to generate a stall anticipatory signal. Appropriate filtering is applied to the summed acceleration and angle-of-attack signal to prevent transient variations in signal such as can be caused by air turbulence from actuating the stall warning system.

Structural Filter

A second-order notch and first-order lag filter for structural mode attenuation are included in the longitudinal SFCS forward loop. The open loop peak gain amplitudes corresponding to each of the three structural modes were used to obtain the minimum attenuation at each structual mode. The attenuation is at least 6 db for all flight conditions investigated.

Stability and Response Characteristics

Aircraft Short Period and Control System Oscillatory Modes

Stability margins for the aircraft short period and control system oscillatory modes were determined by the Bode frequency response method. Open loop frequency response plots were obtained for 15 flight conditions. Two aircraft weights were used for each flight condition documented. The lowest gain margin value computed is 13.2 db, which is well within the gain margin guideline of 10 db. The lowest value of phase margin computed is 41.8° which is considered acceptable. Calculated gain and phase margins using the fixed low gain state at these same flight conditions indicated acceptable gain margins in all cases, and acceptable phase margins at all normally expected flight conditions.

Structural Mode Stability

Bode frequency response and root locus techniques were employed in determining the stability characteristics of the longitudinal SFCS structural modes. Structural mode data were obtained for 11 flight conditions and two aircraft weight configurations. The open loop frequency responses were computed for all flight conditions. The open loop peak gain amplitudes corresponding to each of the three structural modes were used to obtain the minimum attenuation at each structural mode. The structural mode attenuation is at least -6 db for all flight conditions investigated. Also, the desired attenuation of -9 db for the stabilator bending and first vertical bending modes for subsonic flight conditions has been maintained. Since the adaptive gain changer will select MED or HIGH gain as appropriate for some flight conditions, the structural mode attenuation will be greater for fixed low gain operation by 6 to 12 db at these conditions.

C and dC*/dt Criteria*

Time history responses of C^* and dC^*/dt, as presented in Fig. 3, for the nonterminal flight conditions compare favorably with these evaluation criteria. The responses for Mach 0.5, alt = 25,000 ft and Mach 2.15, alt = 36,000 ft do not fall entirely within the envelopes. These two flight conditions represent extreme points on the flight envelope which were included in the analysis primarily to investigate achievable stability margins. The significance of these two response shapes is decreased since, at these conditions, any maneuver from one g flight produces speed and/or altitude changes. This effect tends to mask any undesirable characteristics in the load factor and pitch rate response.

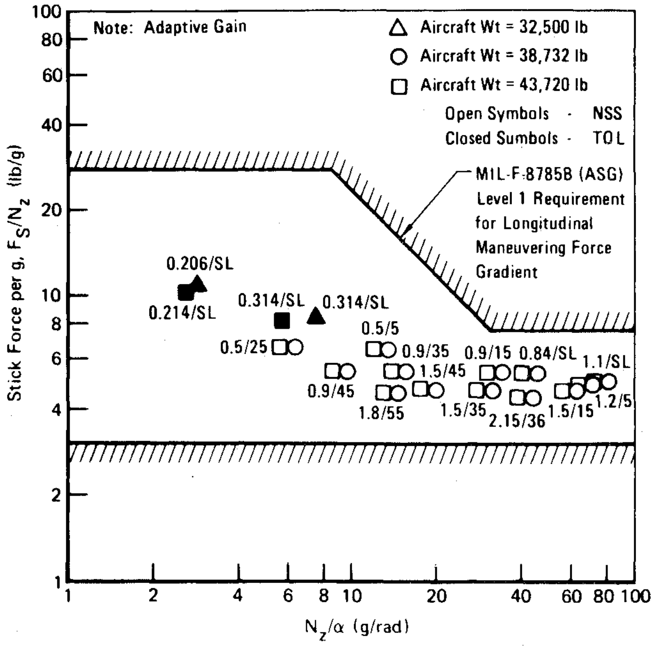

Fig. 4 Stick force per g.

Frequency and Damping

Root locus data for the longitudinal SFCS were computed for 15 flight conditions and 2 aircraft weights for the purpose of documenting the frequency and damping of the aircraft short period and phugoid modes.

The frequency and damping of the closed loop roots which emanate from the basic aircraft short period poles were compiled for the adaptive gain expected at each flight condition. These roots represent the predominant oscillatory modes.

The SFCS short period damping ratios at 29 of the 30 conditions documented are within the Level 1 range, $0.35 \leq \zeta \leq 1.3$ as specified in MIL-F-8785B (ASG), Paragraph 3.2.2.1.2 for Category A and C flight phases. The one point at which the damping ratio is below the required 0.35 value is at a flight condition (Mach 2.15, alt = 36,000 ft, wt = 38,732 lb) on the edge of the flight envelope. Sustained, maneuvering flight is virtually impossible at this flight condition since the drag created when a maneuver is initiated results in rapid loss of airspeed.

Maneuvering Force Gradient

The center stick sensitivity was set to provide a minimum maneuvering force gradient of 4 lb/g at the high q flight conditions. The electrical gains were appropriately distributed to allow pilot commanded limit load factor without voltage saturation in the SFCS. The center stick maneuvering force gradients for the longitudinal SFCS are presented in Fig. 4 for 15 flight conditions and 2 aircraft weights. All flight conditions investigated exhibited stick force per g characteristics in compliance with the Level 1 requirements of MIL-F-8785B (ASG) Paragraph 3.2.2.2.1.

Provisions were made in the SFCS to vary the center stick sensitivity through a range of 50% to 150% of the nominal values. The sidestick controller longitudinal sensitivity was selected and evaluated during the man-in-the-loop simulation program. Figure 5 shows a comparison of the center stick and sidestick output commands as a function of pilot applied force. The sidestick has a breakout of

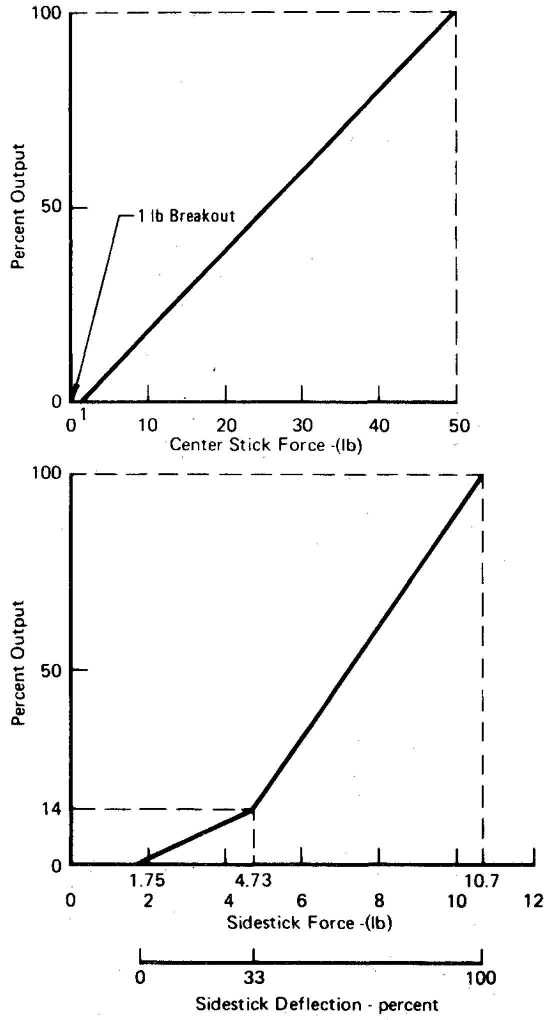

Fig. 5 Longitudinal control system command vs stick force.

1.75 lb followed by dual linear gradients. The breakout was originally set at 2.5 lb; however, it was found during the man-in-the-loop simulation that this value was too high. Consequently, the breakout was adjusted down to 1.75 lb, at which point favorable pilot comments were received for simulated flight with the sidestick. Dual gradients were provided in the SSC to permit fine control of pitch attitude or normal acceleration with low force inputs while retaining limit load factor capability with higher forces. Evaluation of the SSC during the flight simulation program indicated that the gradients chosen are satisfactory. A capability to adjust the location of the gradient break point has been included in the system design.

Lateral-Directional Control Law Development

The lateral and directional axes functional block diagrams are presented in Figs. 6 and 7 respectively. The features of these mechanizations are presented below.

Lateral Axis

The lateral control system provides Normal and EBU modes as in the longitudinal axis. The lateral control system was designed to comply with the criteria in MIL-F-8785B (ASG), and provide a nearly constant roll rate time constant and roll rate to stick force sensitivity throughout the flight envelope. In addition, the system was evaluated in terms of the lateral and directional criteria of Ref. 3. A fixed gain in the roll rate feedback loop suffices for satisfactory compliance with all applicable roll criteria. However, provisions have been made for a future adaptive gain function to enhance system flexibility.

The salient features of the SFCS lateral axis are the improved roll rate time constant and roll rate to stick force gradient, lower roll rate overshoot and a reduction in roll oscillation. Figure 8 shows a comparison between the roll mode time constant of the basic F-4 and the F-4 with the SFCS. The SFCS values are the time to reach 65% of the average roll rate in response to a step of stick force as measured from time history responses. The SFCS time constants vary between 0.35 and 0.7 sec with the more typical values being 0.5 sec. Although the response does not follow the 0.33 sec prefilter time constant exactly the improvement over the unaugmented F-4 is substantial.

At 15,000 ft altitude, the variation in roll rate to stick force with Mach number is about 3 to 1 for the F-4 with yaw SAS. The F-4 with SFCS has about a 1.35 to 1 variation under the same conditions. In addition, the SFCS provides a lower sensitivity for small roll maneuvers for precise tracking, and higher sensitivity for more rapid roll rate maneuvers. The relatively high SFCS roll rate feedback gain effectively suppresses the Dutch Roll oscillations in the roll rate response to a step of stick force. The parameter, P_{OSC}/P_{AV}, is a measure of the ratio of the oscillatory component of the roll rate to the average com-

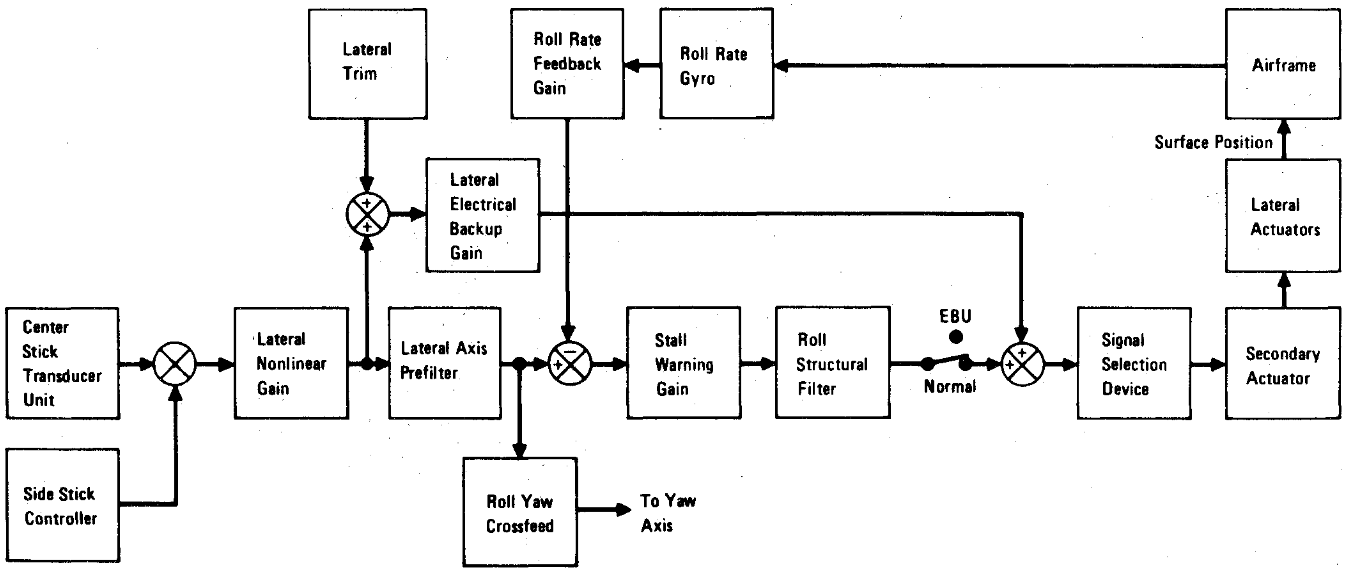

Fig. 6 Simplified functional block diagram single channel of lateral axis.

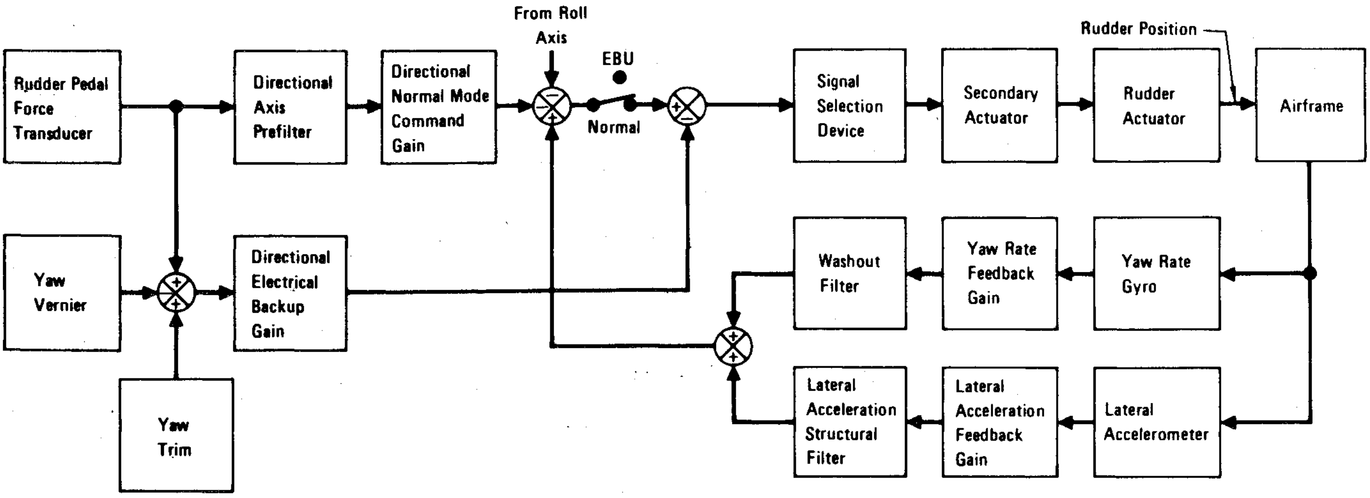

Fig. 7 Simplified functional block diagram single channel of directional axis.

ponent of roll rate following a rudder-pedal-free step of lateral stick force. This parameter is well below the MIL-F-8785B(ASG) Level 1 requirements and near zero for most flight conditions. P_{OSC}/P_{AV} was calculated as per MIL-F-8785B(ASG).

An evaluation of lateral response in terms of the normalized roll rate response criteria is presented in Fig. 9 for three flight conditions. These curves show the improvement in the F-4 SFCS lateral response over that of the F-4 with its production yaw stability augmentation system (SAS).

The EBU mode command path remains a part of the Normal mode with the gains selected to provide for maximum control surface deflection capability at maximum stick force.

The use of ailerons at or near stall on a high performance aircraft can precipitate spin. To avoid an aileron induced spin, a signal from the stall warning computer reduces effective roll rate feedback to zero as a function of cancelled pitch rate and angle-of-attack at high angles-of-attack. The aileron to stick force gain is also reduced to that present in the unaugmented airplane or EBU mode. This design feature was evaluated on the six-degree-of-freedom, man-in-the-loop simulator. The results showed that the chances of inducing a spin are lower when the above design feature is incorporated into the lateral axis.

Directional Axis

The major requirements for the directional axis are to provide acceptable Dutch Roll mode damping and to keep the sideslip excursions in rolling maneuvers sufficiently small so as to provide good tracking performance.

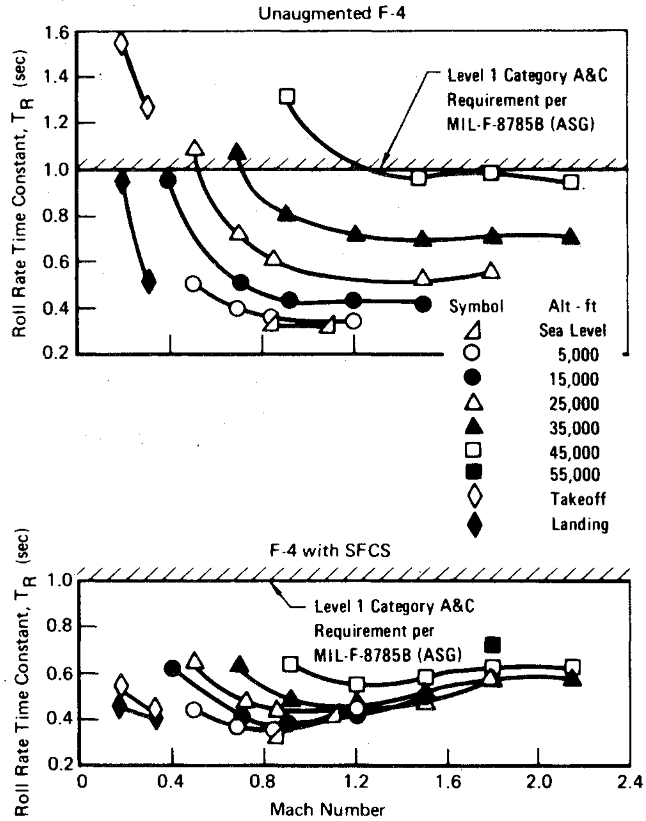

Fig. 8 Roll rate time constant.

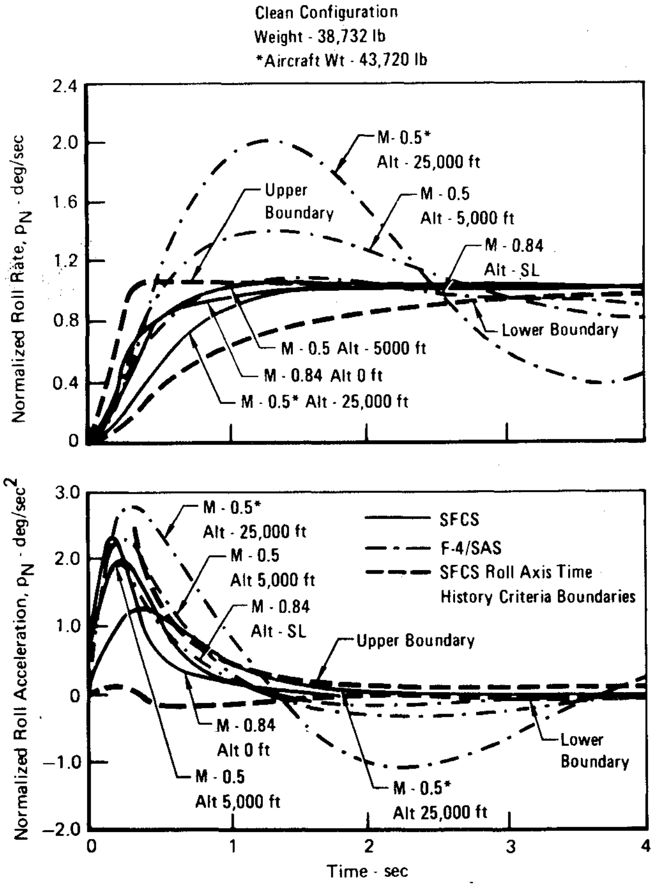

Fig. 9 Normalized roll response.

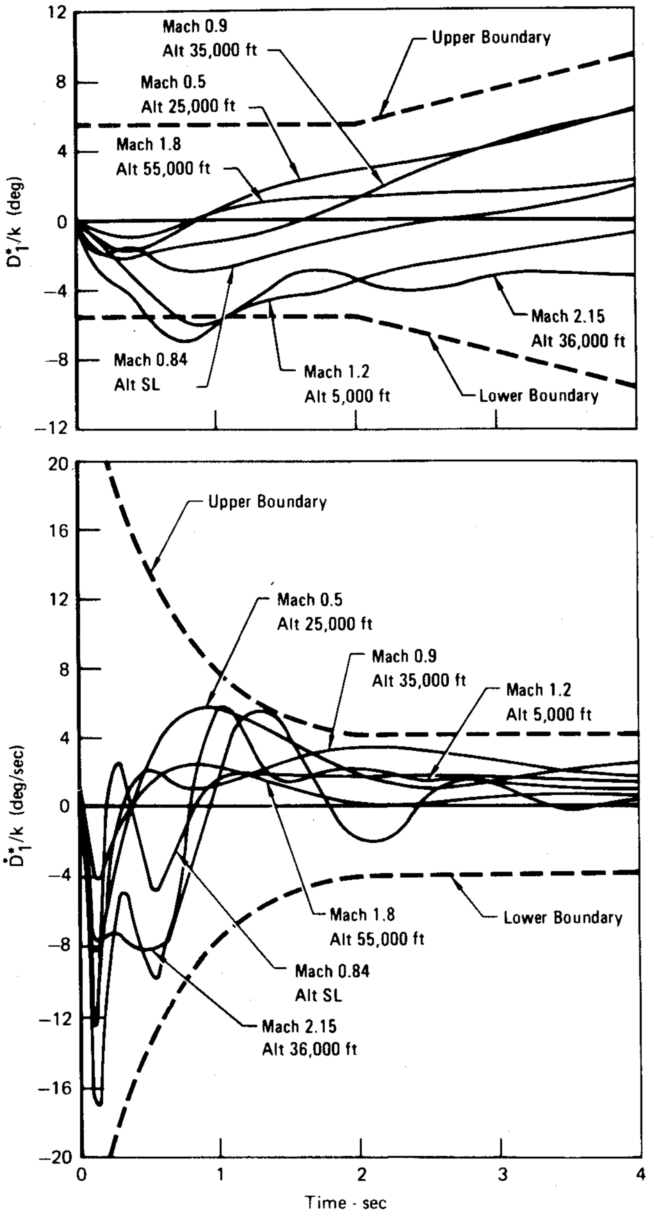

Fig. 10 SFCS lateral directional D^*_1 response.

To achieve good damping and minimize sideslip excursions a combination of yaw rate and lateral acceleration feedback is employed in the directional axis. The high lateral loop gain alone prevents the roll to sideslip coupling in the Dutch Roll mode from exceeding maximum allowables. Yaw rate to rudder feedback augments the airframe Dutch Roll damping. A washout network in this feedback loop prevents opposing rudder deflection during steady-state turn maneuvers. Lateral acceleration feedback aids in reducing sideslip and provides turn coordination especially at high q flight conditions. Turn coordination at low and mid q flight conditions is augmented by the roll to yaw crossfeed network.

A fixed gain is provided in the lateral acceleration feedback loop and a three-value variable gain is included in the yaw rate feedback loop. This variable gain is both pilot selectable and controllable by the adaptive gain changer as a function of the aircraft stabilator effectiveness parameter, M_δ. This is the same parameter used to control the adaptive gain changer in the pitch axis of control.

The variable gain is required to minimize the effects of yaw rate feedback on adverse yaw at low M_δ flight conditions, while providing sufficient Dutch Roll damping at high M_δ flight conditions. The gear down switch adjusts the yaw rate gain for the landing configuration. The F-4 SFCS Dutch Roll frequency and damping characteristics are within MIL-F-8785B(ASG) requirements.

For the high M_δ flight conditions, the SFCS generally has a damping ratio between 0.3 and 0.5. At the low M_δ flight conditions, damping ranges between 0.3 and 0.4. The mid-range M_δ flight conditions show a 0.75 to 0.95 damping ratio for subsonic flight and 0.2 to 0.35 for supersonic flight. These data were obtained from results of a digital computer analysis of the Dutch Roll root only. However, in a highly augmented system there are generally other complex roots which may reduce overall damping slightly at the highly damped flight conditions.

As in the lateral axis, the directional axis EBU mode is utilized during Normal mode operation with overall gains selected to provide full rudder authority at maximum rudder pedal force. The Normal mode command gain is set to offset the feedback signals at low q where full rudder deflection may be desired. A three radian filter is included in this command path to reduce the initial command gradient and prevent over sensitivity of the rudder pedals because of the added gain.

Roll to Yaw Crossfeed

The directional axis yaw rate and lateral acceleration feedbacks provide satisfactory Dutch Roll mode damping, but do not provide for sufficient improvement in turn coordination. Severe sideslip excursions resulting from rolling maneuvers can exist at several flight conditions. Increasing lateral acceleration feedback is not feasible because of its degrading effect on Dutch Roll mode damping, the higher rms g environment experienced by the pilot during random gusts, and structural mode feedback. Therefore, a crossfeed from the roll rate command signal to the directional axis was selected for the SFCS.

An analysis of the F-4's roll induced sideslip shows that sideslip varies with flight condition, from highly adverse at high angles-of-attack to proverse at low angles-of-attack. A fixed network crossfeed therefore would not be sufficient. The SFCS roll to yaw crossfeed is relatively uncomplicated and adequate to provide a significant reduction in sideslip. The network incorporates two variable gains which are changed automatically as a function of M_δ by the adaptive gain changer, or manually by the directional axis manual gain select switch. Four sets of gains, varying from high gains at low M_δ to zero gain at high M_δ, are provided through the adaptive gain changer. A fifth set of gains is used with gear down. A limiter is required to avoid excessive crossfeed commands at low roll power flight conditions.

The turn coordination performance for the F-4 SFCS with the adaptive roll to yaw crossfeed gains was calculated in terms of MIL-F-8785B(ASG) sideslip requirement. With the SFCS, the side-slip is within the requirements except at the very low q and very high q flight conditions.

An evaluation of the SFCS with respect to the MCAIR developed lateral-direction $D_1{}^*$ criteria is presented in Fig. 10 for three subsonic and supersonic flight conditions.

The $D_1{}^*$ parameter recorded in Fig. 10 is in response to a step of lateral stick force and is a combination of sideslip and lateral acceleration at the pilot seat as expressed by the following equation:

$$D_1{}^*/K = \beta - 0.513a(y_p)(\text{deg})$$

where

β(sideslip) in deg

ay_p (lateral acceleration at the pilot seat) in ft/sec^2

K = Ratio of "command roll performance" to "applicable roll performance requirement"

Structural Modes

Three F-4 structural modes have been identified as having to be considered in the design of the lateral-directional control system. An analysis of these modes shows that the lateral loop is primarily sensitive to the fuselage first torsion mode. The relatively low frequency and the high sensitivity of this mode to aileron deflection, coupled with the high SFCS lateral loop gain, required that a 10 db, 40 rad notch filter be provided in the lateral loop. The directional axis is primarily sensitive to the fuselage first lateral bending mode through the lateral accelerometer pickup. Sufficient attenuation of this structural mode in the SFCS is provided by the low lateral acceleration loop gain and a 40 rad lag filter in the lateral acceleration feedback loop. No filter for structural modes is required in the yaw rate loop.

Adequate phase margins, gain margins, and structural mode attenuation have been achieved in the design of the lateral-directional SFCS. The minimum lateral axis stability margins are 61 deg of phase and 15 db of gain. The minimum directional axis stability margins are 58 deg of phase and 13 db of gain. The gains at the structural frequencies evaluated are such as to assure a 10 db gain margin regardless of the phase shift present in the system caused by any undefined nonlinearities.

Evaluation Simulation

A fixed base flight simulator program was conducted to evaluate the selected SFCS to reveal any design problems or necessary modifications. A complete six-degree-of-freedom, three-axis, single-channel SFCS math model was programmed on the hybrid computer along with aircraft flight dynamics representing the F-4E. The entire flight envelope could be flown including power approach, stall entry, and post stall recovery. The simulator crew station was equipped with all necessary controls for the pilot to fly the aircraft and change the SFCS operating mode during flight.

The SFCS manned simulator program was performed using the McDonnell Automation Company hybrid simulation facility. Use of digital computing techniques on the CDC 6600 permitted large variations in flight parameters while retaining the precision required for simulation accuracy. An analog computer served as a buffer to the cockpit equipment, visual display equipment and time history recorders. Peripheral equipment, consisting of closed circuit TV, terrain map, gimballed target model and horizon generator, provided a very realistic environment in which to evaluate the selected SFCS.

Seven modifications to the SFCS resulted from the subject simulation. These modifications were included in the system design to achieve reduced pilot workload, improved SFCS effectiveness, and reduced possibility of catastrophic transients induced by inadvertant pilot action. These system modifications were to:

1) Reduce the maximum lateral roll force from ±20 lb to ±12 lb.

2) Eliminate the high passed stabilator position feedback for the EBU mode.

3) Disable the circuit in the secondary actuator feedback loop which returns the integrator to the takeoff trim position on the engagement of electrical back-up mode when the landing gear is extended.

4) Reduce the roll electrical trim authority from ±33-⅓% of full lateral surface to ±16 ⅔% of full authority.

5) Reduce the pitch electrical trim authority in the EBU mode to ±1° δ_s with an additional ±4° of stabilator gear down. The ±1° portion is also used as the fly-by-wire trim authority and allows trimming into a 45° banked level turn at 360 knots at sea level.

6) Reduce the side stick breakout forces from ±2.5 lb to ±1.75 lb.

7) Increase the easy-on-easy-off switching time in the pitch axis from 2.5 sec to 5.0 sec.

Conclusions

A flight control system has been synthesized for use in a fly-by-wire flight test program. Analysis of this system indicates that the stability and control characteristics of the F-4 test aircraft will be improved throughout its flight envelope when using this system. Specific conclusions drawn from these studies are:

1) The SFCS will provide adequate stability and good performance characteristics in the Normal mode.

2) The electrical back-up mode will provide for safe return and landing in the event of total failure of the Normal mode.

3) The longitudinal SFCS, when operating in the Normal mode, provides airframe responses which compare favorably with the C^* and dC^*/dt criteria throughout the F-4E flight envelope.

4) The lateral SFCS, operating in the Normal mode, provides improved airframe roll response throughout the F-4E flight envelope.

5) The directional SFCS, operating in the Normal mode, provides Dutch Roll damping which meets the MIL-F-8785B(ASG) requirements throughout the F-4E flight envelope.

6) The roll to yaw crossfeed provides improved turn coordination and meets the MIL-F-8785B(ASG) requirements except for some very low q and very high Mach number flight conditions.

7) The stall warning mechanization provides good indication of the approach to stall.

8) Adequate structural mode stability margins for stable operation are provided by the three axis SFCS through the correct placement of feedback sensors and the utilization of structural filters.

Flight testing will provide data to determine the validity of the design analyses and to verify the handling qualities of the SFCS equipped F-4.

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