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FLIGHT TEST EXPERIENCE WITH THE F-8 DIGITAL FLY-BY-WIRE SYSTEM

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SUMMARY

Flight test results of the F-8 digital fly-by-wire (DFBW) control system are presented and the implications for application to active control technology (ACT) are discussed. The F-8 DFBW system has several of the attributes of proposed ACT systems, so the flight test experience is helpful in assessing the capabilities of those systems. Topics of discussion include the predicted and actual flight performance of the control system, assessments of aircraft flying qualities and other piloting factors, software management and control, and operational experience.

INTRODUCTION

In May 1972 the flight testing of the F-8 DFBW aircraft began. This aircraft, which used Apollo guidance and navigation system hardware, was the first to rely on a DFBW system for primary flight control. The design and development of the F-8 DFBW control system are described in references 1 to 3. This paper presents the major flight test results for the control system. A detailed description of the system's software development and verification is given in reference 4, and the backup control actuation systems are described in reference 5.

The primary objectives of the flight tests were to evaluate the performance of the digital flight control system and to acquire operating experience with it. The program also served to determine whether the long-advertised advantages and capabilities of DFBW control systems could be realized. Many of these advantages, such as software flexibility, system reliability, and computational ability, make a DFBW system a logical candidate for active control technology applications. The F-8 DFBW control system had characteristics in common with systems proposed for ACT applications. Specifically, it was a highly reliable, full authority system that was committed for use from the first takeoff and landing. An analog control system was the only backup to the DFBW system. The mechanical controls of the basic F-8C airplane were removed before the first flight.

This approach parallels that taken toward the development of an active control system, both in terms of the importance attributed to the design of the control system and the reliability and management of hardware and software, and in terms of the requirement for detailed preflight testing. This paper emphasizes the aspects of the flight test program that relate to the broader considerations of an active control system.

SYMBOLS

$a_1, a_2, a_3, b_1, b_2, b_3,$	digital filter coefficients
$C^* = n_Z - \frac{V_{co}}{57.3g} q$, g	
G(s)	general s-plane filter
G(w)	general w-plane filter
G(z)	general digital filter
K	general gain constant
K_{C^*}	C* feedback gain, deg/g
K_p	roll rate feedback gain, deg/deg/sec
K_q	pitch rate feedback gain, deg/deg/sec
K_r	yaw rate feedback gain, deg/deg/sec
L_{δ_a}	roll acceleration due to aileron deflection, $deg/sec^2/deg$
M	Mach number
M_{δ_e}	pitch acceleration due to elevon deflection, $deg/sec^2/deg$
N_{δ_r}	yaw acceleration due to rudder deflection, $deg/sec^2/deg$
n_Z	acceleration along positive Z-body axis, g
p	roll rate, deg/sec

q	pitch rate, deg/sec
r	yaw rate, deg/sec
s	Laplace transform variable
T	sample period, sec
V	velocity, KIAS
V_{co}	crossover velocity, m/sec
w	sampled-data system frequency domain variable
z	sampled-data domain transform variable
Δ	incremental change
δ	general surface command, deg
δ_{a_p}	pilot roll stick deflection, cm
δ_e	horizontal stabilizer deflection, deg
ζ	damping ratio
θ	pitch attitude, deg
$\tau_{r_{eff}}$	effective roll mode time constant, sec
φ	roll attitude, deg
ψ	heading angle, deg
ω	natural frequency, Hz

Subscripts:

d	Dutch roll mode
n	current sample
n-1	last sample
p	pilot
sp	longitudinal short period mode

ss	steady state
Z	component along aircraft Z-body axis in positive (down) direction
$\hat{( )}$	derived quantity ( )

ABBREVIATIONS

ACT	active control technology
A/D	analog to digital
CAS	command augmentation system
D/A	digital to analog
DFBW	digital fly-by-wire
DSKY	display and keyboard
KIAS	knots indicated airspeed
PCM	pulse code modulation
PIO	pilot-induced oscillation
SAS	stability augmentation system

CONDUCT OF FLIGHT TEST PROGRAM

Figure 1 illustrates the nature and sequence of the phases of the flight test program. The first three flights were made by using the proportional control, or direct, digital mode. The fourth flight culminated in a landing during which three-axis DFBW stability augmentation was used. The evaluation of the DFBW control system progressed rapidly from then on, and by the eighth flight all modes had been flown. The airplane was then evaluated in a variety of tasks, including ground-controlled approaches, gunsight tracking, mild aerobatics, and formation flight. The latter portion of the flight program concentrated on flying qualities assessments by additional pilots and on an evaluation of a minimum-displacement side stick that operated through the backup control system only (ref. 5). In total, 58 hours were accumulated by six pilots during 42 flights.

The F-8 DFBW system was flight tested within the flight envelope shown in figure 2. Most of the closed-loop evaluations were made at speeds between 250 knots indicated airspeed (KIAS) and 400 KIAS and altitudes from 6000 meters to

10,700 meters. Tests at low speeds (below 200 KIAS) were made with the variable-incidence wing of the F-8C airplane in the up position. Pilot ratings were given in accordance with the Cooper-Harper scale (ref. 6).

All flights were conducted during the daytime under VFR conditions. They averaged 80 minutes in duration. Each flight was monitored in a control room in which 36 airplane parameters were displayed. In addition, duplicates of the pilot's mode panel and servo status panel showed the state of the fly-by-wire control system. All parameters were telemetered from the aircraft's pulse code modulation (PCM) data acquisition system.

CONTROL SYSTEM PERFORMANCE

The digital flight control system consisted of pilot-selectable modes in each axis. The mode panel layout is described in reference 3. The available modes are shown by axis in the table below:

	Axis		
	Pitch	Roll	Yaw
Modes available	Direct SAS CAS	Direct SAS Test	Direct SAS - - - -

The direct mode, which had no augmentation, and a stability augmentation system (SAS) mode were provided in each axis. A command augmentation system (CAS) mode was also available in the pitch axis. The roll test mode was used to facilitate comparisons between various SAS mode configurations. Block diagrams of the digital control modes are shown in figures 3(a) to 3(c).

Direct Mode

The direct mode provided proportional control with no augmentation. Figure 3(a) shows the direct mode mechanization, which was similar in all axes. Analog-to-digital (A/D) quantization of the stick outputs, effective quantization on trim due to sample rate, and digital-to-analog (D/A) output quantization are aspects of digital flight control that were apparent in this mode. Linear and nonlinear stick shaping were used during the flight program. In the pitch axis, linear and parabolic shaping were used (fig. 4). The Apollo A/D interface allowed a maximum of 45 quantization levels for full stick or pedal deflection in one direction. The Apollo computer D/A converter output quantization, which had ±384 levels, was approximately an order of magnitude finer than the stick A/D converter. The linear gearing mechanization resulted in a quantization level of 0.59° of horizontal stabilizer deflection when full pitch control authority was retained. During early flights, various linear gearing gains were evaluated. Table 1 summarizes the pitch quantization effects found with linear gearing. The threshold of quantization detection appeared to be from 0.15g to 0.2g and 1.2 degrees per second to 1.5 degrees per second of peak pitch rate. Figure 5 shows an example of the thumping that the pilot detected at 365 KIAS as he attempted to increase pitch rate smoothly. This small airplane excitation was characteristic of the quantization effect in the pitch and roll axes resulting from control surface actuator response to staircase commands.

The parabolic stick shaping resulted in a nonlinear quantization. The step size is shown in table 2. This shaping greatly improved the fine pitch control of the airplane, while retaining nearly full stabilizer authority. With this mechanization, pilots reported that quantization was not apparent at speeds up to approximately 400 KIAS. In the roll axis, stick quantization had to be reduced by changing the linear gearing about the center stick position. The initial value of 1.04° of total aileron command was changed to 0.36°. This reduced the minimum commanded roll rate from 8.32 degrees per second to 2.90 degrees per second at 250 KIAS and yielded acceptable roll control around trim. The only noticeable effect of quantization in the yaw axis was in random 1-bit commands that were observed at 400 KIAS. Lateral acceleration peaks of 0.03g due to 1-bit or 0.38° rudder surface commands were observed. This problem was corrected by writing software in erasable memory to allow a 1-bit deadband in the rudder pedal command. No other rudder pedal quantization effects were seen.

It should be noted that the ±45 quantization steps available represented less than a 6-bit A/D conversion. A 12-bit (11 bits plus sign) A/D capability is available today. This yields a resolution nearly 50 times as fine as that in the F-8 DFBW system. At the most sensitive F-8C flight condition, which was Mach 0.86 at sea level, a 12-bit A/D interface would have allowed digital commands as small as 0.001g, assuming linear gearing and full surface authority. Therefore it is safe to assume that the quantization effects of a modern A/D interface would be negligible and undetectable by the pilot.

Quantization of pilot trim inputs due to sample rate also became apparent in the flight program. In the F-8 DFBW mechanization, trim command discretes were sampled every 90 milliseconds. Based on the pitch trim rate value of 1.25 degrees per second, the minimum software command was 0.11°. This command is nearly twice as coarse as the D/A converter quantization steps of 0.069° for the horizontal stabilizer. This effective trim quantization was a factor in making precise trim of the F-8 DFBW aircraft difficult at a target speed and altitude.

The pitch trim discrete inputs should have been sampled at the major cycle sample period of 30 milliseconds, which would have resulted in a trim quantization of 0.0375°. This would have taken full advantage of the output D/A quantization. This points out the need to sample beep trim discrete inputs at a high enough rate to yield acceptable output quantization. In some cases, trim discretes may have to be sampled at rates higher than the major cycle sample rate, if fine trim resolution is required.

Stability Augmentation System Mode

The nominal SAS configurations flown are shown in figure 3(b). Body axis rate was estimated by filtering the transformed inertial attitude from the Apollo inertial platform. Compensation filtering and gain were placed in the feedback path. There was an aileron-to-rudder interconnect in the yaw SAS mode only. The stick and trim processing were identical to those in the direct mode. A rate reasonability check was applied to the final command, and an automatic transfer to the direct mode resulted if the reasonability threshold was exceeded.

The digital SAS modes operated as expected. This is important from the point of view of the sampled-data design process. The acceptance of digital control systems depends in large part on the ability to predict system performance accurately.

The digital SAS loops were designed by using sampled-data analysis methods, especially the z-plane root locus method. The linear system model used in the pitch axis is shown in figure 6. An ideal pitch rate signal was assumed. At first, the rate estimation filter that acted on pitch attitude was used in the model, but the resulting pitch rate signal was found to be nearly identical to that for the ideal case at the F-8C short period frequencies. Neither the highly nonlinear A/D conversion of gimbal angles nor the axis transformation steps were modeled. Four symmetrical bending modes were included in the analysis.

The z-plane root locus for the pitch SAS mode without lead-lag compensation is shown in figure 7(a). A lead-lag filter was designed to improve the performance of the pitch rate loop in increasing the short period damping ratio. A w-plane frequency response was used to select the compensation root locations. The w-plane compensation,

$$G(w) = \frac{w/0.1 + 1}{w^2/0.16 + w/0.286 + 1}$$

was transformed to the z-plane by $w = \frac{z - 1}{z + 1}$ and yielded a discrete filter,

$$G(z) = \frac{1.023(1 + z^{-1})(1 - 0.818z^{-1})}{1.0 - 0.976z^{-1} + 0.349z^{-2}}$$

The root locus for the compensated system is shown in figure 7(b). Higher short period damping ratios were achieved by using the lead-lag filter, as one would expect in a continuous system. A comparison between the predicted effects of the compensation filter and those measured in flight is shown in figure 8, where the increment in short period damping ratio is shown for three flight conditions. The sampled-data system prediction is good.

The improvement in airplane response with the pitch SAS is evident in the flight time histories in figure 9. Figures 10(a) to 10(c) show a comparison of predicted with measured damping in the three airplane axes. Agreement is good for the longitudinal short period (fig. 10(a)) and Dutch roll (fig. 10(b)) modes. At low gains, rate estimation quantization and actuator friction restricted surface motion at the angular rates tested, and, as a result, the SAS loop was less effective.

The flight performance of the digital roll SAS mode is illustrated in figure 10(c). Since the roll rate response that resulted from a step lateral stick command was contaminated slightly by the Dutch roll, an effective roll mode time constant corresponding to the time between the initial roll rate response and the time when 63 percent of steady state was achieved was used. Yaw SAS was engaged on all runs to reduce the Dutch roll contamination. The predicted trend, which was for decreasing roll mode time constant with increasing roll SAS gain, is clear, although a bias of approximately 0.05 second is apparent. One factor that contributed to this bias was the nonideal pilot step input, which resembled a rapid ramp. This resulted in a slightly higher than predicted effective time constant, since the predicted value was based on a perfect step input.

To further evaluate the sampled-data analysis method, the pitch rate feedback gain was increased in flight until the compensation root approached neutral stability. Figure 11 shows the z-plane root locus prediction of the neutral stability point to be in good agreement with the flight-measured results.

The SAS modes also operated well at low speeds. Pitch SAS results are shown in figure 12(a). A washout filter was designed for low speed operation in the s-plane as

$$G(s) = \frac{s}{s + 1}$$

The discrete washout filter formed by using the bilinear transformation for real roots was

$$G(z) = \frac{0.98522(1 - z^{-1})}{1 - 0.9704z^{-1}}$$

The results of the washout filter addition to the feedback loop on aircraft response was as expected (fig. 12(b)). The highest loop gains used in flight were $|K_q M_{\delta_e}| = 3.8$ in pitch, $|K_p L_{\delta_a}| = 3.2$ in roll, and $|K_r N_{\delta_r}| = 1.2$ in yaw. One further observation is appropriate. The Apollo inertial platform was designed for precise navigation. It had an A/D interface, the coupling data unit, that was not designed to facilitate rate estimation. Even so, the derived body rate provided a signal that could be used satisfactorily for the F-8 DFBW damper modes.

Command Augmentation System Mode

The pitch CAS mode block diagram is shown in figure 3(c). Derived normal acceleration is blended with derived pitch rate to form the feedback signal, C* (ref. 7). A forward loop integrator and bypass path provided zero steady state error and resulted in neutral aircraft speed stability. The $\cos \theta$ correction term eliminated acceleration feedback in a steady climb or descent. The pilot stick and trim interface with this mode was the same as in the direct and SAS modes.

As was the case in the pitch SAS mode, the performance of the digital CAS mode was essentially as predicted by linear sampled-data systems analysis. However, gain values selected for the C* feedback gain during the preliminary design could not be used in flight. The reasons for this are traceable to the noise problems associated with using rates and accelerations derived from the Apollo inertial measurement unit and interface hardware. These problems are not inherent in a digital mechanization. For acceptable noise levels at the horizontal stabilizer, the C* feedback gain was too low for optimum response. The flight performance of the CAS mode was reasonable at low speeds, however. Figures 13(a) and 13(b) compare the F-8 DFBW C* response in the direct and CAS modes at 180 KIAS and 250 KIAS, respectively. These responses, normalized to the final value, are shown with respect to the C* power approach and cruise design envelopes, respectively. The improvement in airplane response is substantial. The 250-KIAS response illustrates the problem encountered in CAS with insufficient loop gain. The short period response was satisfactory, but the aircraft exhibited drift in the 3- to 8-second time period that was actually the first-order mode resulting from the forward loop integrator. This effect was apparent to the pilots.

The CAS mode provided the expected neutral speed stability. Figures 14(a) and 14(b) show the phugoid response of the F-8 DFBW aircraft in the direct and CAS modes, respectively. The aircraft, trimmed at 180 KIAS, was slowed approximately 10 KIAS, where the stick was again centered. The CAS mode held zero pitch rate while the aircraft slowed to a new steady state speed of approximately 138 KIAS. Normal acceleration (not shown) remained constant at nearly 1g during the maneuver, while angle of attack (not shown), which started at 3.5°, stabilized at 10°.

The effectiveness of the CAS mode in suppressing transient effects is shown in figure 15, where the response of the F-8C airplane is compared in the direct and CAS modes during a wing transition (wing incidence changes from -1° to 7°). Both responses were without pilot inputs.

Although the performance of the CAS mode was degraded by the limitations of the Apollo hardware, the control system design was relatively straightforward, and flight results again matched predictions quite closely.

Implications of Digital Fly-By-Wire Design for Active Control Systems

The flight verification of the F-8 DFBW control system design was encouraging from an active control technology standpoint. First, the body of continuous control system design experience is largely applicable. In fact, if there is a reasonable separation between the half sample frequency and modes of interest, the design can be accomplished in the continuous domain and then exactly transformed to the discrete domain by using the bilinear transform. Furthermore, direct z-plane design is also possible. The most serious difficulty about using the latter approach is lack of experience with direct digital design.

The entire F-8 DFBW three-axis digital flight control system problem could be solved by the Apollo computer in less than a 30-millisecond major cycle time period. The capabilities of a current high performance computer and those of the Apollo computer are:

	Apollo computer	Current computer
Memory cycle time, μsec	11.7	1.0
Add time, μsec	23.4	2.5
Multiply time, μsec	46.8	6.0

The table shows that a state-of-the-art computer can be expected to be an order of magnitude faster than the Apollo computer. This suggests a sample rate or job capacity increase of the same magnitude. Although computer sizing must await a specific ACT configuration, the capability of today's computers would appear to be more than adequate for the control system tasks envisioned.

PILOTING FACTORS

Considered in conjunction with the control system performance reported in the previous section, the handling qualities results confirmed the feasibility and utility of a digital fly-by-wire control system.

Handling Qualities Summary

The flying qualities of the F-8 DFBW were evaluated by the pilots in a variety of tasks, including simulated instrument cruise, large or abrupt maneuvers, ground-controlled approaches, gunsight tracking, and close formation flight (ref. 8).

Figure 16(a) summarizes the longitudinal handling qualities results for small instrument maneuvers, and figure 16(b) summarizes the results for large maneuvers. The piloting tasks and the comment guide used for these evaluations are given in the appendix. In figure 16(a) the comments and ratings are typical of the findings of pilots at low-to-moderate cruise speeds (less than 350 KIAS). For large maneuvers the pilot rating improvement with control system sophistication was evident. Pilot acceptance of the SAS and CAS modes was expected on the basis of the control system and vehicle response characteristics reported in the previous section. Some pilots did report a long period overshooting tendency in the CAS mode for certain maneuvers where steady state pitch rates had to be arrested. This correlated with the first-order integrator mode present in the CAS step response.

Figure 17 is characteristic of the improvement in pitch control with digital SAS as seen by the pilots in a wind-up turn. In the direct mode, the F-8C airplane displays its undesirable short period damping. The same maneuver could be performed easily and precisely in the pitch SAS mode.

Ground-controlled approaches were flown down to approximately 60 meters under simulated instrument flight conditions in the various digital modes. Figures 18(a) and 18(b) show typical pilot comments and ratings in the lateral-directional and longitudinal axes. The pilot ratings reflect the improvement in Dutch roll damping provided by the yaw SAS mode. In figure 18(b) pilot A objected to a slight long-term overshooting tendency in the CAS mode.

The tracking performance of the F-8C airplane with the digital control system was degraded by stick quantization problems in both the pitch and roll axes. The parabolic pitch stick shaping resulted in unacceptable quantization steps at large aft stick positions (table 2). This degraded the pitch control of the airplane so much that even augmentation did not significantly improve the tracking performance. Some improvement with roll and yaw SAS was evident in a 2g gunsight tracking maneuver, as the time histories in figure 19 and the associated pilot comments and ratings in figure 20 show. The augmented time histories in figure 19 correspond to a yaw SAS gain, K_r, of 0.4 deg/deg/sec.

Close formation flight revealed deficiencies in the flying qualities that were often not apparent in maneuvers where the pilot was not required to be "in the loop" as tightly. The improvement shown in figure 21 of the longitudinal flying qualities with digital augmentation is typical. Pilot comments reflected the decreased workload evident in the time history. Barrel rolls, aileron rolls, and wingovers were performed in all control modes. Pilots noted little difference in their ability to perform these maneuvers between the direct and augmented modes, perhaps because these maneuvers tended to be more open loop in nature.

Except in maneuvers where the coarse stick quantization problem was overriding, as in the gunsight tracking maneuver, the DFBW control system markedly improved the flying qualities of the unaugmented F-8C aircraft. Because of the control system performance described in the previous section, this was not unexpected. One pilot who flew F-8C airplanes regularly found the F-8 DFBW vehicle superior even to a standard F-8C airplane with normal augmentation. He noted in particular the lack of the usual mechanical control slop.

The results of the flying qualities evaluations, coupled with the control system performance previously described, indicate that a DFBW control system can perform as well as or better than a conventional control system. The only serious problems encountered were due to the limitations of the Apollo system hardware, which would not be factors in a current design.

Pilot Interface With the Digital-Fly-by-Wire System

The F-8 DFBW system was designed to permit a simple, yet flexible, interface with the pilot. The normal astronaut interface with the Apollo guidance and navigation system was a display and keyboard device (DSKY) that allowed the operator to display memory contents, load erasable memory, or initiate special programs. The versatility of this interface was important to the design and test engineers during the development and flight test program, but it was not made available to the pilot because of its complexity for a single place aircraft. The pilot's only interface with the digital computer was through a mode and gain panel, which is described in reference 3. The pilot's gain switch mechanization in software contributed to the rapid, safe flight checkout of the digital flight control system. Table 3 lists the different digital control system parameters that were tied to the gain switches during the flight test program. In all, 105 parameters could be connected via software to the three gain switches.

With this gain mechanization, different control system parameters could rapidly be selected and optimized during the research program. More important, the gain switches allowed the designer to make use of the pilot's capabilities. Nominal values of critical gains that were established during the simulation phase were placed on the gain switches along with larger and smaller values. The pilot could change the gain values at any time. For example, one of the gain switches was for pitch gearing. During the first flight, when the effects of the pitch quantization and sensitivity had not yet been established, the pilot took off in the nominal gain position. By 13 minutes after takeoff at 300 KIAS, he had reduced the gearing 10 percent because of pitch control sensitivity. Before landing he evaluated three gain positions, finally selecting the nominal gain value 2 1/2 minutes before touchdown. Apart from its research value, this type of gain selection and evaluation gave the pilot an important degree of freedom. Switch arrangements like this are not unique to digital flight control systems, but the ability to designate such a large number (105) of parameters for this use with virtually no hardware impact is unique to a digital system.

This kind of flexibility can be carried in a digital computer with only a small increase in software complexity. This mechanization approach would also be advantageous in an active control system design, because the F-8 DFBW experience showed that the pilot could rapidly and safely assess open- and closed-loop gain parameter variations about the nominal design point during flight.

Flight experience also showed the multimode digital flight control system to be safe and valuable for both research and proof testing phases of the flight program. The low mode of control in the primary digital system (direct) provided a fallback position for both the pilot and the system. Since the direct and augmented modes were fully synchronized, they could be switched manually or automatically under any dynamic conditions with a minimum and safe aircraft response transient. The pilots took advantage of this multimode mechanization to diagnose the cause of flying qualities deficiencies by comparing airplane response in each mode.

Like the gain switch arrangement, the multimode mechanization makes use of the online monitoring capabilities of the pilot. It too is a good candidate for active control mechanization, especially for the first few flights. One problem was encountered with this approach. Mode changes could occur without being commanded by the pilot due, for example, to a reasonability test. The mode panel display light configuration would change, but this was not easily detected by the pilot. A master caution and annunciator warning of any uncommanded mode change should have been incorporated.

In summary, software flexibility allowed the test pilot to use his real-time diagnostic capability and to make control system alterations. The alterations could be made with almost no hardware impact and with minimum additional software complexity. These concepts are applicable to early flight testing of full time active control systems.

MANAGEMENT OF FLIGHT SOFTWARE

The flexibility and versatility of digital flight control system software carries with it the need for software management and control. Perhaps no other area of digital fly-by-wire control raises as many questions and doubts as software reliability. The concern centers on whether it is possible to achieve reliable man-rated flight control software at a reasonable cost and whether software flexibility is compatible with software reliability in a practical application. The F-8 DFBW experience indicates that both questions can be answered yes.

Two aspects of the F-8 DFBW flight test program are of significance to full authority, man-rated digital flight control software. First, not a single software programing error was discovered during the flight test program. Much of the credit for this is due to the thorough verification procedures and facilities developed for the Apollo software, which were also used during the F-8 DFBW program, although on a smaller scale. The procedures are described in detail in reference 4. Secondly, not a single incorrect erasable memory constant propagated to a flight tape that was used to load the Apollo computer. These results are significant because an active control system must achieve the same level of reliability as the basic airframe. The software, in turn, is central to the active control system's reliability, because even though an active control system would have redundant digital channels, the software would be common to all, as it was in the F-8 DFBW system. For this reason, it is worthwhile to examine the software management procedures used in the F-8 DFBW program.

Figure 22(a) outlines the procedures established to control software programing changes during the flight program. These procedures were used three times after the hardwired memory was manufactured and before the first flight. The three special purpose programs written into the erasable memory consisted of pitch and roll parabolic stick shaping, yaw pedal deadband, and a special failure mode monitor.

The software control board in figure 22(a) consisted of representatives from control system engineering, project management, operations, and the pilots' office. Step 7 in figure 22(a) consisted not only of checking out the new code but rerunning former, documented tests on related code to insure proper program interaction, if any. Extensive files of detailed all-digital simulation runs generated during the initial verification phase were kept for comparison with identical runs with the modified code. This permitted short turnaround time for new additions to the code.

Figure 22(b) shows the steps taken in the alteration of control system constants in the erasable memory. In total, 394 erasable memory locations had to be loaded for each flight. Table 4 gives a breakdown of these constants. Sum checks and built-in data transmission checks in the Apollo computer made it possible to insure that the desired octal numbers were loaded into the computer.

Making sure that the 168 control system values loaded were those actually desired was less straightforward. A punched tape was used to load the computer. During the flight program six tapes were manufactured, each of which represented a different flight control system configuration.

Because the Apollo digital computer is a fixed-point machine, there were magnitude restrictions due to program scaling on most parameters. A variety of other restrictions combined to create a formidable set of rules for the set of control system constants.

An off-line diagnostic digital program (step 3 of fig. 22(b)), which ran on a data processing computer, was developed to ease the burden of verifying the correct content of the master load list, which was kept on standard punch cards. One task performed by the diagnostic program was to check each of the 394 constants against a previously drawn list of reasonable values. This reasonability list was constructed after considerable experience was gained from iron bird simulation, but before the first flight tape was made. The limits were set to encompass the expected or allowable operating range of each variable. Deviations from reasonability limits were flagged by the program as major errors and had to be corrected or signed off by the responsible engineer.

The program also reconstructed digital filter forms from their coefficients and computed their vital characteristics, such as root location, steady state gain, and absolute root magnitude in the z-plane. This was helpful in the case of digital filters, the characteristics of which are not as obvious as those of continuous filters.

One aspect of software control became apparent during the ground testing and simulation of various control system gain configurations. When many gain changes had been made and the precise configuration was in doubt, it was only necessary to dump the contents of erasable memory on magnetic tape to create a complete description of any given configuration. This capability proved to be extremely valuable in the control system refinement stage, and it is unique to a digital mechanization. It was also possible to revert to the baseline configuration merely by reloading memory with the baseline punched tape. This required approximately 3 minutes on the Apollo computer.

In summary, the F-8 DFBW flight experience indicates that highly reliable flight software can be generated and maintained, but that it requires thorough control.

Because the F-8 DFBW program was intended for research, the software program was made more flexible than would be necessary for a production airplane. Even with this flexibility, the software was easily managed with diagnostic digital programs, resulting in high overall system reliability. In fact, changes were made to the digital system more confidently than they were to the airplane's analog systems because there was no hardware impact.

Partly because of the built-in flexibility of the control system mechanization, only minor changes had to be made to the basic program during the flight test program. More program changes would be expected in a prototype system development, thus increasing the need for strict configuration management for software.

The F-8 DFBW flight results confirmed that a DFBW control system could be used in an active control application from the standpoint of software reliability and system flexibility.

OPERATIONAL FACTORS

Reasonability Checks

The software reasonability checks used in all augmented modes are surface command rate checks made over one sample period (30 milliseconds). Exceeding the threshold value in any axis resulted in an automatic downmode to the direct mode in that axis. The threshold values per sample period that were found to be usable in flight were 4.5° in pitch, 13.0° in roll, and 8.0° in yaw. These were the smallest values that allowed nearly any pilot input. Ten downmodes occurred in flight. All except four were directly related to sharp pilot step inputs that were made for test purposes. The other four were due to noise peaks that resulted from the angular rate derivation. At least one of these occurred in each axis.

The reasonability check was designed to detect abrupt command changes due to sensor failures or major software faults. Experience with the F-8 DFBW system indicated that the threshold rate limit could be reduced by at least 50 percent in all axes for an operational fighter. If unreasonable commands were allowed to exist for 100 milliseconds (approximately three sample periods), nuisance downmodes would be eliminated without sacrificing protection.

Digital System Reliability

The F-8 DFBW digital control system utilized a single highly reliable digital computer. This configuration would probably not be used in an active control system. However, the reliability requirements of the F-8 digital system are representative of the requirements of an ACT application. First, no single failure was permitted that would have resulted in the generation of a hazardous control surface command. Second, any serious failure within the digital system had to be detected. In the F-8 DFBW airplane, the failure warning signals were used to transfer control to the analog backup control system. In a redundant digital control system, operation would continue on the remaining good digital channels after a component failure. F-8 DFBW reliability experience is nevertheless applicable to active control technology in terms of failure detection and also in terms of the features of the digital mechanization that led to a high level of confidence in this system.

No hardware failures occurred in the primary digital flight control system on any flight. This is not surprising in view of the demonstrated in-service reliability of the Apollo guidance and navigation equipment. The discrepancies noted in the DFBW flight system, excluding the actuators and their drive electronics, are listed in table 5. Three component failures occurred in two systems during the 2500 hours of operation (items 4, 5, and 10). Item 4 would have had no impact on normal flight operation. The failure monitor in item 5 was added to the system during the flight program to protect against a potentially hazardous single-point failure mode in the Apollo computer output interface hardware. The monitor box failed before its first use in flight, although it failed in the proper "safe" mode (transfer to the backup control system). The roll stick circuit failure (item 10) would have caused a downmode to the backup control system in flight, as it did on the ground. There were no unresolved anomalies.

Preflight Procedures

Two preflight test procedures were used for the digital system. The first was a 1-hour test done on the system in the hangar the day before flight. Electrical and hydraulic power were external. The second procedure was part of the total aircraft preflight immediately before flight, with engine-supplied electrical and hydraulic power. The elements of the hangar and flight line preflight tests are listed in table 6. Virtually all the hangar tests except the specialized inertial measurement unit checks and the detailed surface deflection measurements were repeated. Although the digital system's flight line preflight was not optimized in the built-in software, it took only 10 to 15 minutes.

One sensitive preflight test was the computer activity check. A program in the erasable memory was used to measure computer duty cycle indirectly, by detecting idle time over a several second interval. In a given configuration, the duty cycle was consistent within a few percent over several time intervals. This test confirmed proper software operation to a high level of confidence.

During the investigation of the anomalies that occurred on both the iron bird and the F-8 DFBW airplane, it became apparent that it was possible to determine the health of the digital control system rapidly and confidently. The state of the digital control system could be determined in less than 5 minutes by running a self-test and by monitoring the internal control system parameters on the DSKY in the flight control modes. The monitor feature was indispensable during the flight test program. With half a dozen keystrokes, three control system parameters could be displayed in engineering units and in decimal format. The display was updated every second, so even under dynamic conditions the display was intelligible. This monitor format permitted the immediate checkout of virtually any part of the control system. Any future digital flight control system should incorporate such display software capability.

The ability to quickly and confidently assure proper control system performance is of paramount importance to active control systems. The repeatability of the test results of the F-8 DFBW program inspired enormous confidence in the operational readiness of the system before flight. Even personnel not thoroughly familiar with the digital control system were able to perform detailed tests of the system because of the well-designed display and monitor software. The component failures that did occur during ground operation were all detected by the system itself.

CONCLUDING REMARKS

The F-8 digital fly-by-wire (DFBW) flight program showed the feasibility and advantages of DFBW control for aircraft. Even with hardware designed a decade ago for space applications, an Apollo computer easily handled the F-8 DFBW flight control computation task. This demonstrated the inherent flexibility of a digital system.

The following conclusions can be drawn on the basis of the F-8 DFBW flight test program.

(1) Existing design tools, such as the w-plane frequency response and the z-plane root locus, are suitable for the synthesis of digital flight control.

(2) Flight performance of the digital flight control system verified the accuracy of the sampled-data design results for contemporary command and stability augmentation system modes.

(3) Pilot opinion correlated with that expected on the basis of the control system performance.

(4) A modern digital control system design would display no quantization effects noticeable to the pilot.

(5) The flexibility of the digital control system permits effective use of the pilot in configuration optimization in early flight test stages.

(6) Man-rated software can be safely managed while retaining a high degree of flexibility. The use of off-line diagnostic programs greatly reduced the engineering burden of software management.

(7) Digital system integrity can be rapidly and confidently determined in preflight tests by using flexible and extensive engineering interfaces.

The implications of these results for an active control application can be broadly stated as follows:

(1) A DFBW control system possesses the computational ability and flexibility necessary for advanced active control applications. Computer hardware advances are leading control system applications.

(2) Reliable software can be produced and is not an obstacle to an active control application.

(3) The fault detection and preflight test technology necessary for digital control systems exists. Full realization of DFBW potential awaits the successful demonstration of reconfiguration and normal operation after component failures in a practical redundant system.

There was no flight or ground experience that would indicate that a DFBW system could not be used in an active control technology application. In fact, the F-8 DFBW flight program achieved in practice the advantages so long attributed to a DFBW control system and confirmed the suitability of digital control for active control technology.

APPENDIX

PILOT COMMENT GUIDE FOR LONGITUDINAL HANDLING QUALITIES

Instrument Flight Maneuvering

(1) Trim the aircraft to desired speed at a zero rate of climb
(2) Make small heading changes of less than 30°
(3) Make air traffic control altitude changes
(4) Make air traffic control speed changes

Comment on:

(1) The ability to fine trim the aircraft
(2) The need to monitor the pitch axis during lateral-directional tasks
(3) The ability to make accurate changes in attitude
(4) Stick breakout and deadband forces
(5) The acceptability of these aircraft characteristics for fighter aircraft
(6) Overall longitudinal pilot rating

Large or Abrupt Maneuvers

(1) From trimmed flight, quickly establish a 1.5g to 2.5g turn
(2) Recover to trimmed, level flight
(3) Quickly set up a constant speed high performance climb by selecting a target pitch attitude and throttle setting
(4) Recover to trimmed, level flight at target altitude
(5) Increase speed 50 KIAS, and retrim

Comment on:

(1) The ability to control attitude and g. Tendencies to overshoot or for pilot-induced oscillations
(2) The ability to restore the aircraft to trimmed flight
(3) Stick breakout and deadband forces
(4) The lag in aircraft response to stick inputs
(5) Residual small-amplitude oscillations
(6) The acceptability of these characteristics for fighter aircraft
(7) Overall pilot rating for the large or abrupt maneuvers

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TABLE 1.—PITCH STICK QUANTIZATION EFFECTS WITH LINEAR GEARING

V, KIAS	Altitude, m	Dynamic pressure, N/m^2	Stabilizer quantization, deg	Response to quantization		Pilot comments
				n_Z peak, g	q peak, deg/sec	
300	5,500	13,885	0.53	0.2	1.5	Stepping noticeable, but easy to make small corrections.
365	6,100	19,870	0.5	0.3	2.0	Feel thumping in pitch control. Can see quantization.
400	6,100	22,980	0.3	0.15	1.2	Cannot see quantization.
[a]180	3,050	5,030	0.53	0.05	0.5	Cannot see any quantization. Smooth control.

[a]Wing up.

TABLE 2.—QUANTIZATION MAGNITUDE FOR FULL AUTHORITY PARABOLIC PITCH STICK SHAPING

Nominal stick position, cm	Quantization size, deg
0	0.1
5	0.3
10	0.7
15	1.2

TABLE 3.—DIGITAL CONTROL SYSTEM PARAMETERS TIED TO GAIN SWITCHES

Axis	Mode	Description
Pitch	Direct	Stick gearing
Pitch	SAS	Pitch rate feedback gain
Pitch	SAS	Type of digital filter
Pitch	CAS	Forward loop integrator gain
Pitch	CAS	C* feedback gain
Pitch	CAS	Pitch rate blending gain
Roll	Direct	Stick gearing – wing down
Roll	Direct	Stick gearing – wing up
Roll	SAS	Stick gearing
Roll	SAS	Nonlinear stick shaping
Roll	SAS	Roll rate feedback gain
Yaw	SAS	Yaw rate feedback gain
Yaw	SAS	Interconnect function slope
Yaw	SAS	Interconnect function intercept

TABLE 4.–ERASABLE MEMORY CONSTANTS LOADED FOR EACH F-8 DFBW FLIGHT

Description	Number
Control system constants	168
Computer downlink identity tags	100
Inertial subsystem	29
Erasable memory program (parabolic stick shaping)	87
Miscellaneous	10
	Total: 394

TABLE 5.—DIGITAL SYSTEM DISCREPANCIES DURING GROUND OPERATION

(a) Discrepancies.

Item	Discrepancy	Reason for discrepancy
1	Computer restarts	Procedural error
2	Computer time-of-day wrong	Procedural error
3	Inertial measurement unit test result out of specification	Inertial measurement unit degradation for navigation
[a]4	Yaw direct light cycling on-off	Failed transistor in mode panel
[a]5	Backup control system down-mode for rudder inputs	Failure in relay in external fail monitor
6	Computer locked in loop	Procedural error
7	Failure of preflight test	Damage to punched tape
8	Aileron offset	Procedural error
9	Roll D/A drift during backup control system self-test	Truncation during repeated primary/backup control system moding
[a]10	Backup control system down-mode for aileron inputs	Failed resistor in external stick electronics

[a]Primary electronics failures.

(b) Summary.

Component	Failures
Apollo hardware	0
Primary electronics	3

TABLE 6.—ELEMENTS OF F-8 DFBW PREFLIGHT TESTS

Element	Hangar	Flight line
Verify correct memory load	Yes	Yes
Computer self-test	Yes	Yes
Inertial measurement unit fail discretes	Yes	No
Inertial measurement unit turn-on sequence	Yes	Yes
Proper aline	Yes	Yes
Pilot gimbal angle indicator	Yes	Yes
Inertial measurement unit operational test (12 minutes)	Yes	No
Primary/backup control system moding	Yes	Yes
Gain switch discretes	Yes	Yes
Wing position discrete	Yes	Yes
Forced computer restart	Yes	Yes
Inertial measurement unit interface zero and reset	Yes	No
Forced computer fail discrete	Yes	Yes
Mode panel warning lights	Yes	Yes
Differential D/A output – backup control system downmode	Yes	No
Trim rate and trim fail detection	Yes	Yes
Stick-to-surface gearing measurements	Yes	No
Computer activity	Yes	Yes
Check failure monitor box	Yes	Yes
Maximum surface deflections	Yes	Yes
Load time-of-day	No	Yes
Load computer for flight	No	Yes

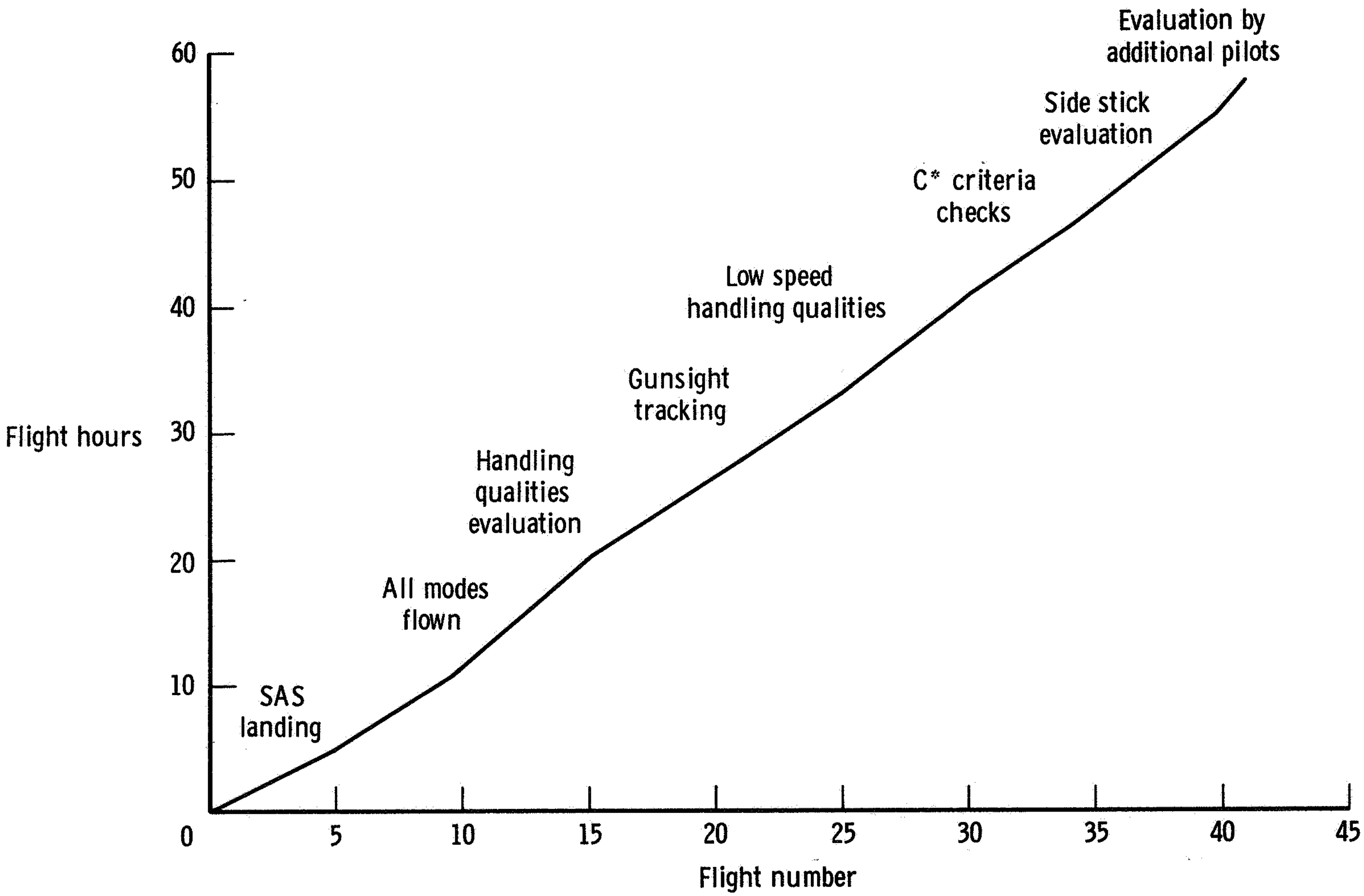


Figure 1. F-8 DFBW flight test summary.

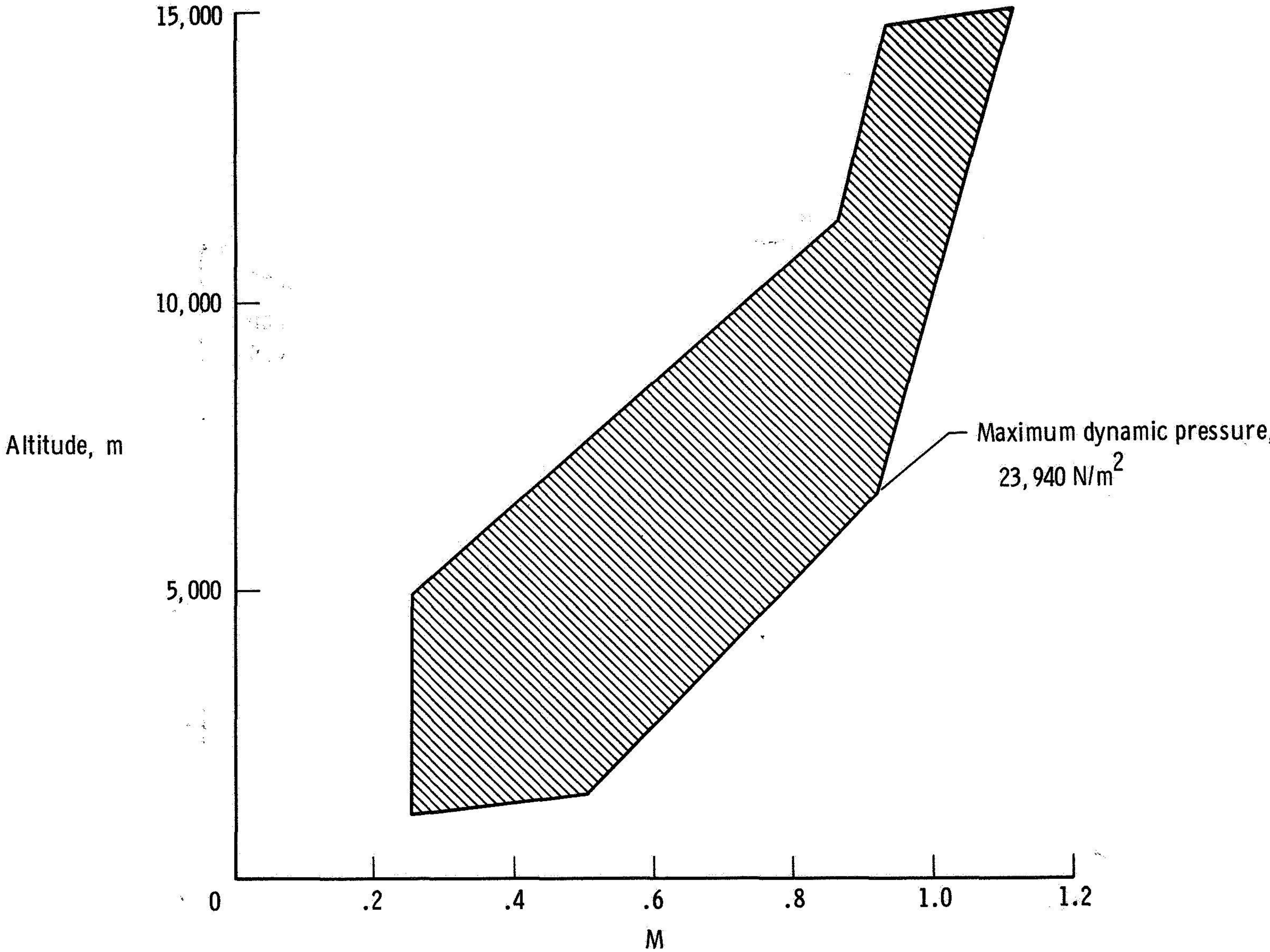
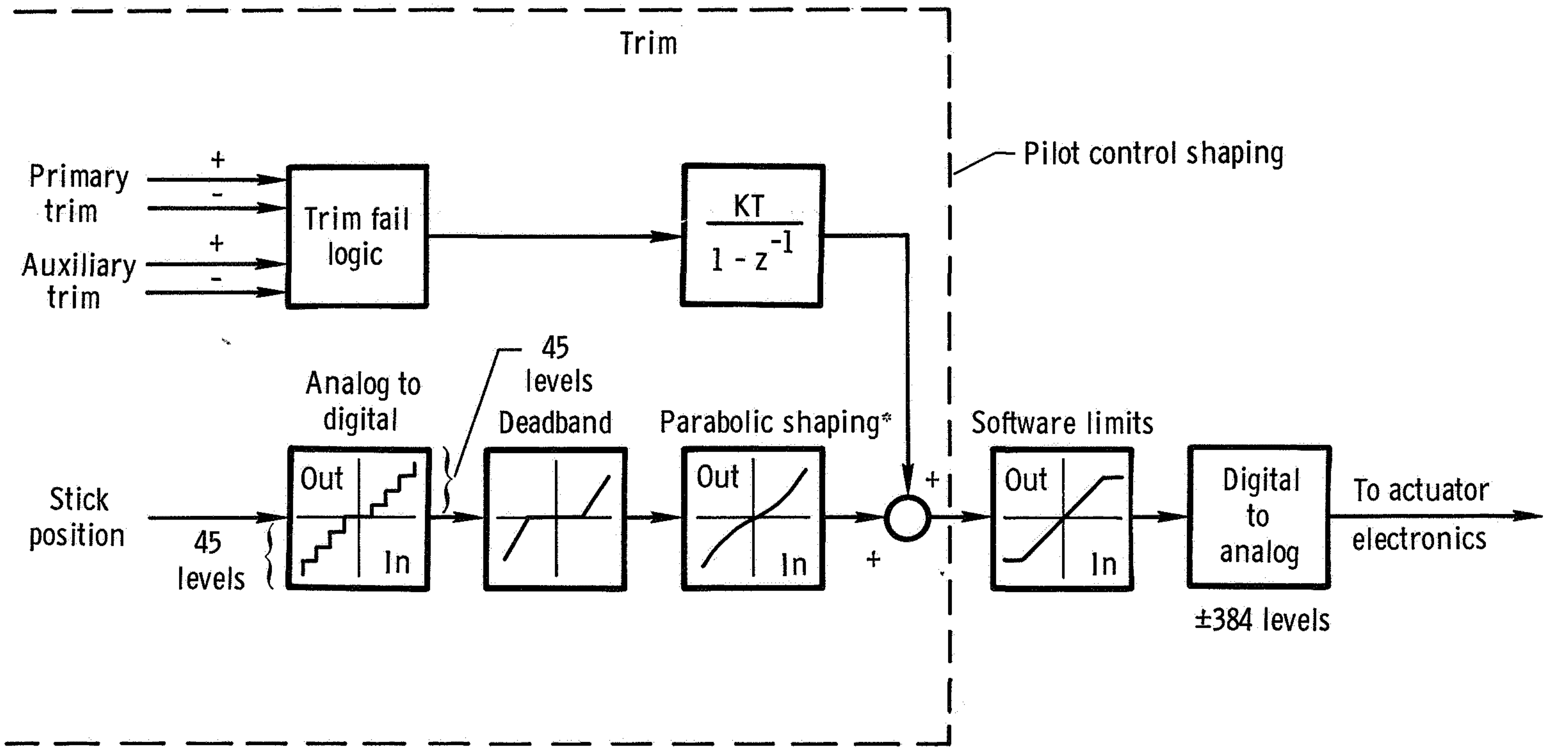


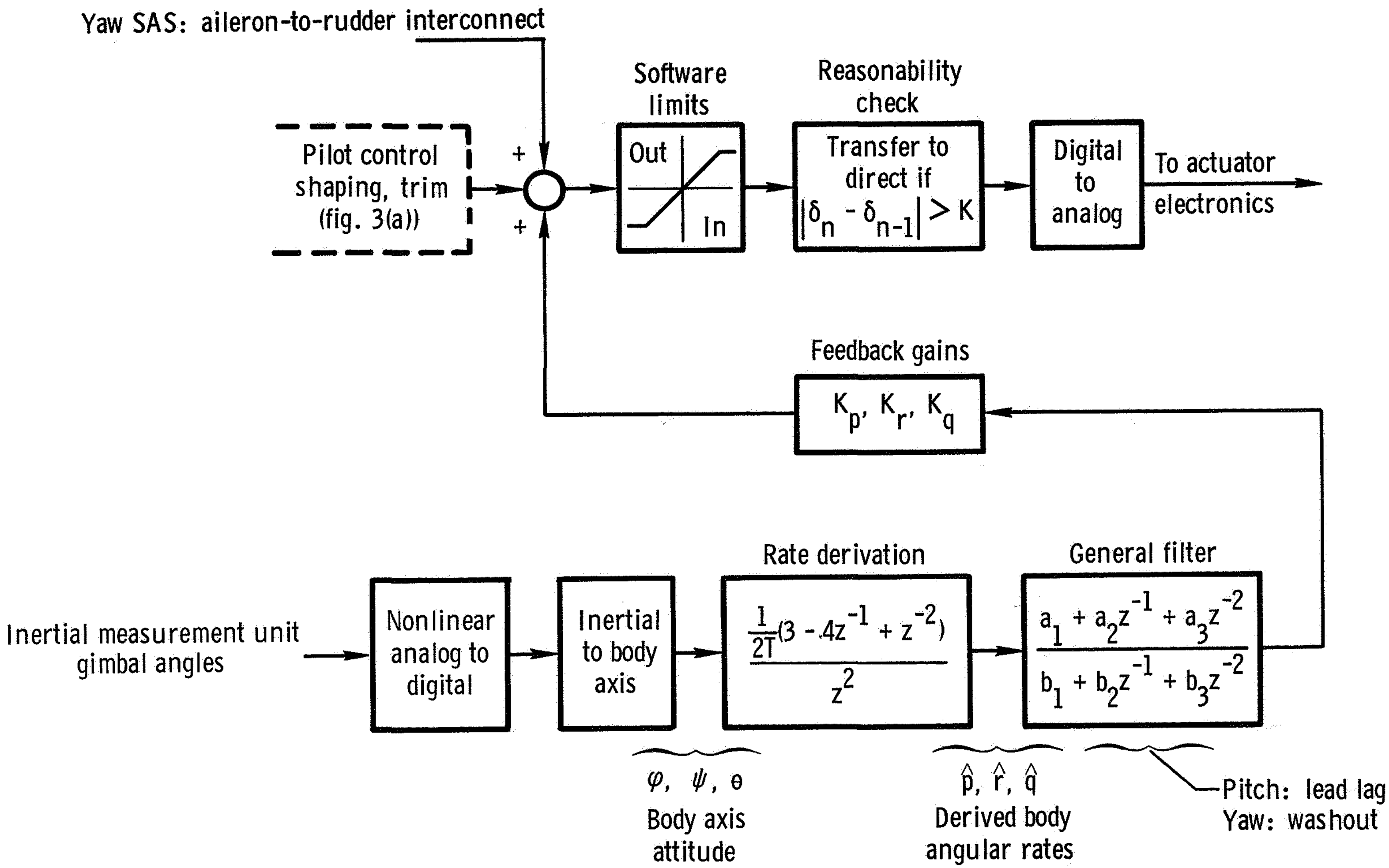
Figure 2. F-8 DFBW flight test envelope.



*Available in pitch and roll axes only

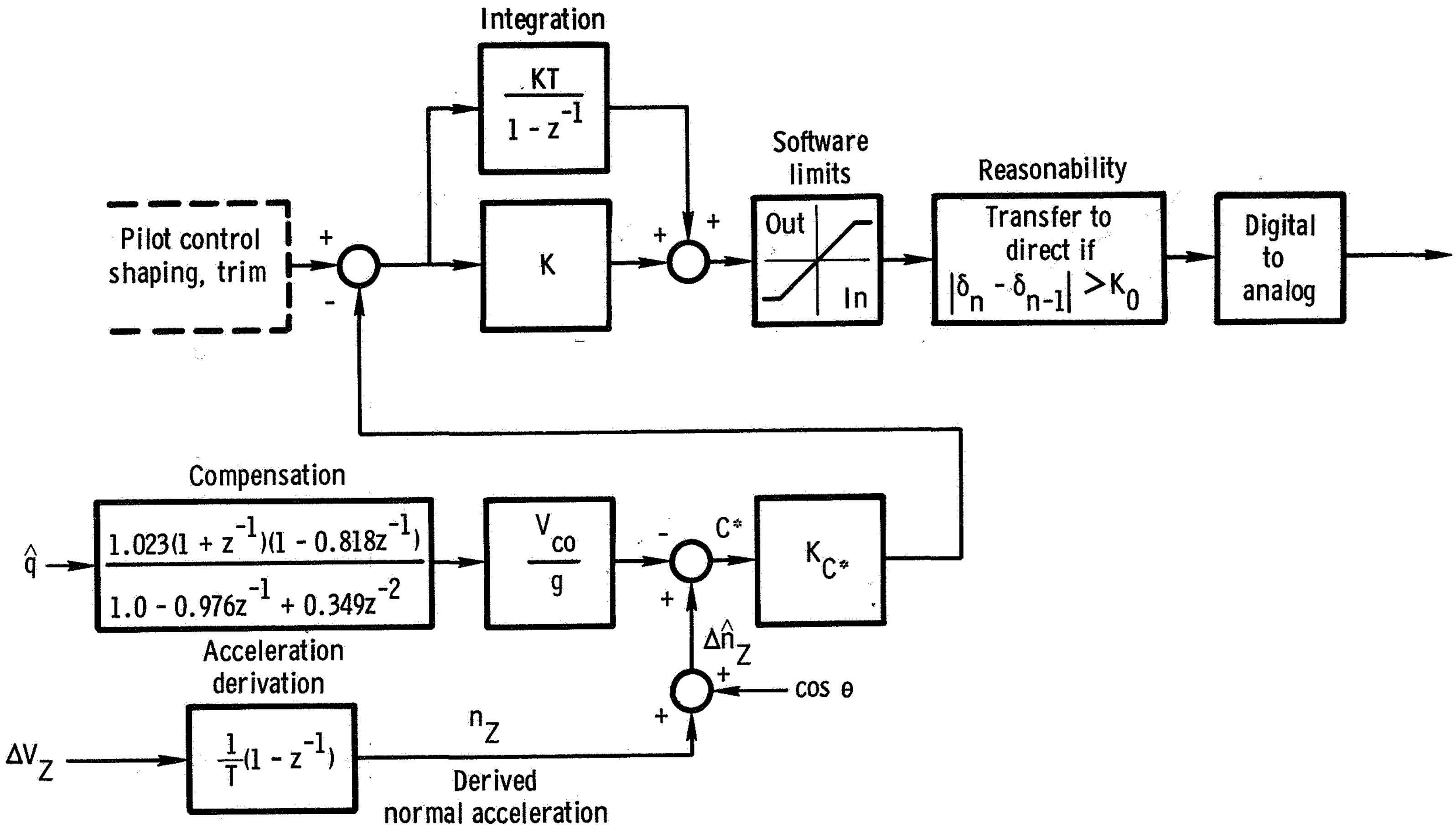
(a) Direct modes.

Figure 3. Digital flight control law diagram.



(b) SAS modes.

Figure 3. Continued.



(c) Pitch CAS mode.

Figure 3. Concluded.

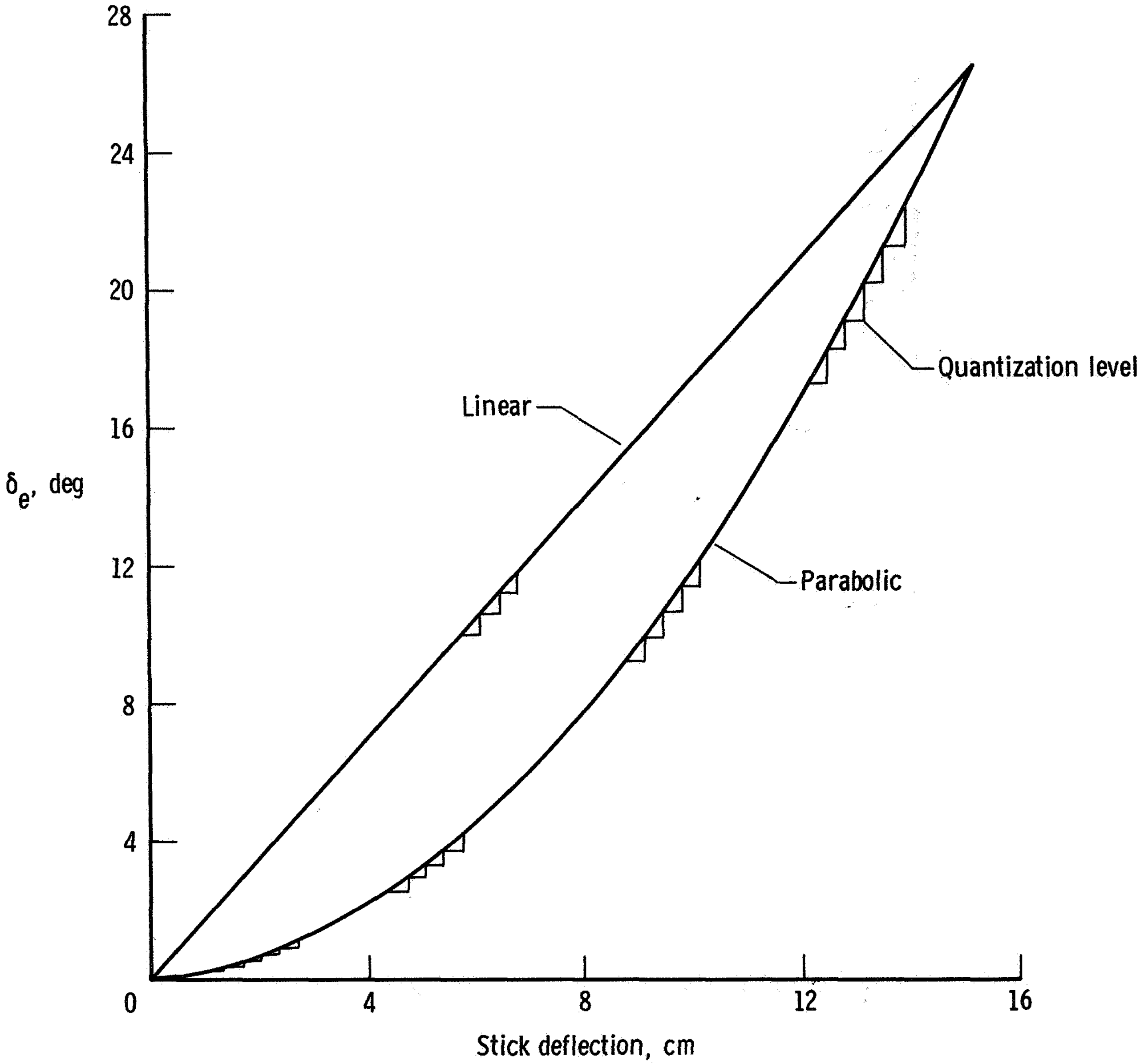


Figure 4. Pitch gearing comparison.

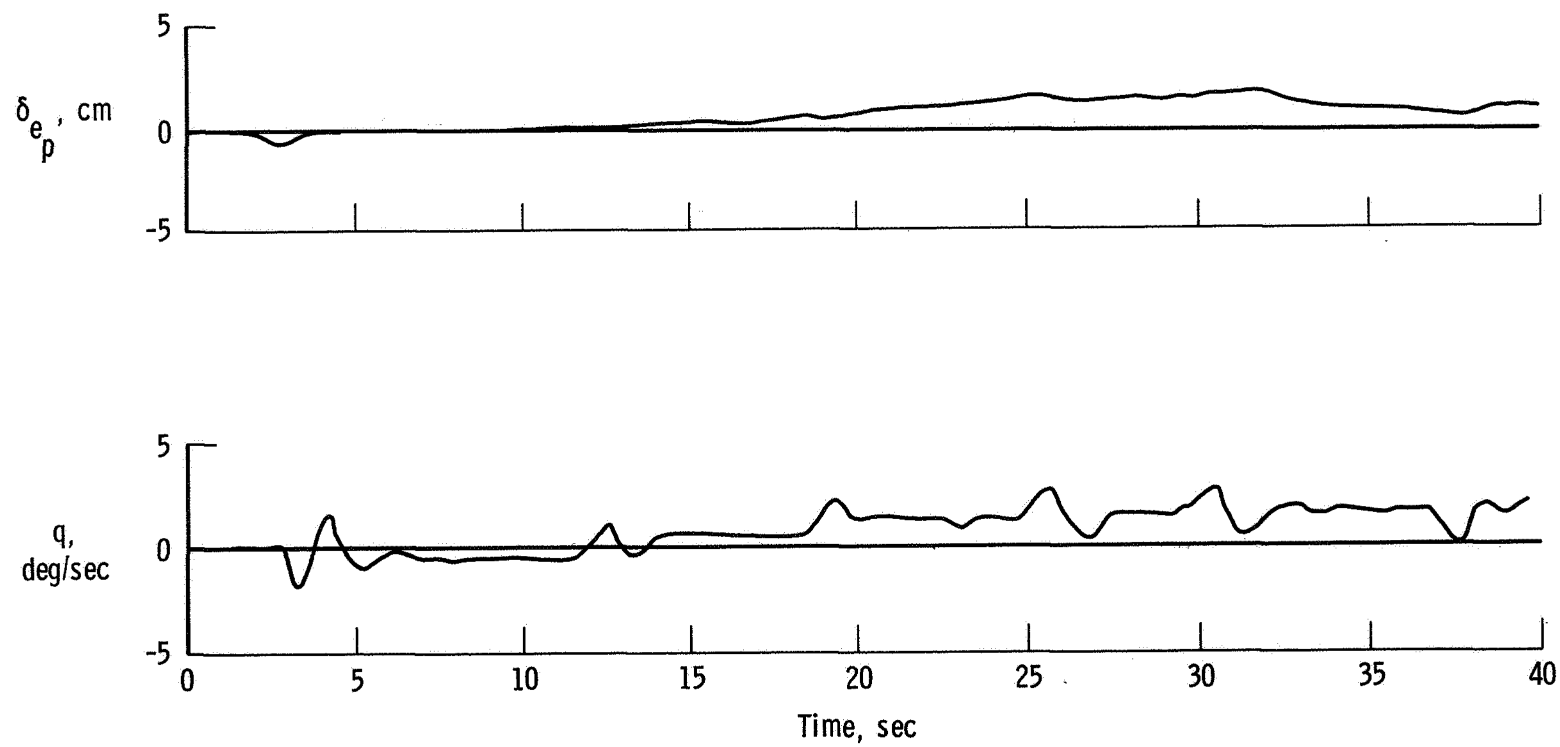


Figure 5. Pitch quantization with linear gearing. 365 KIAS; 6100 m.

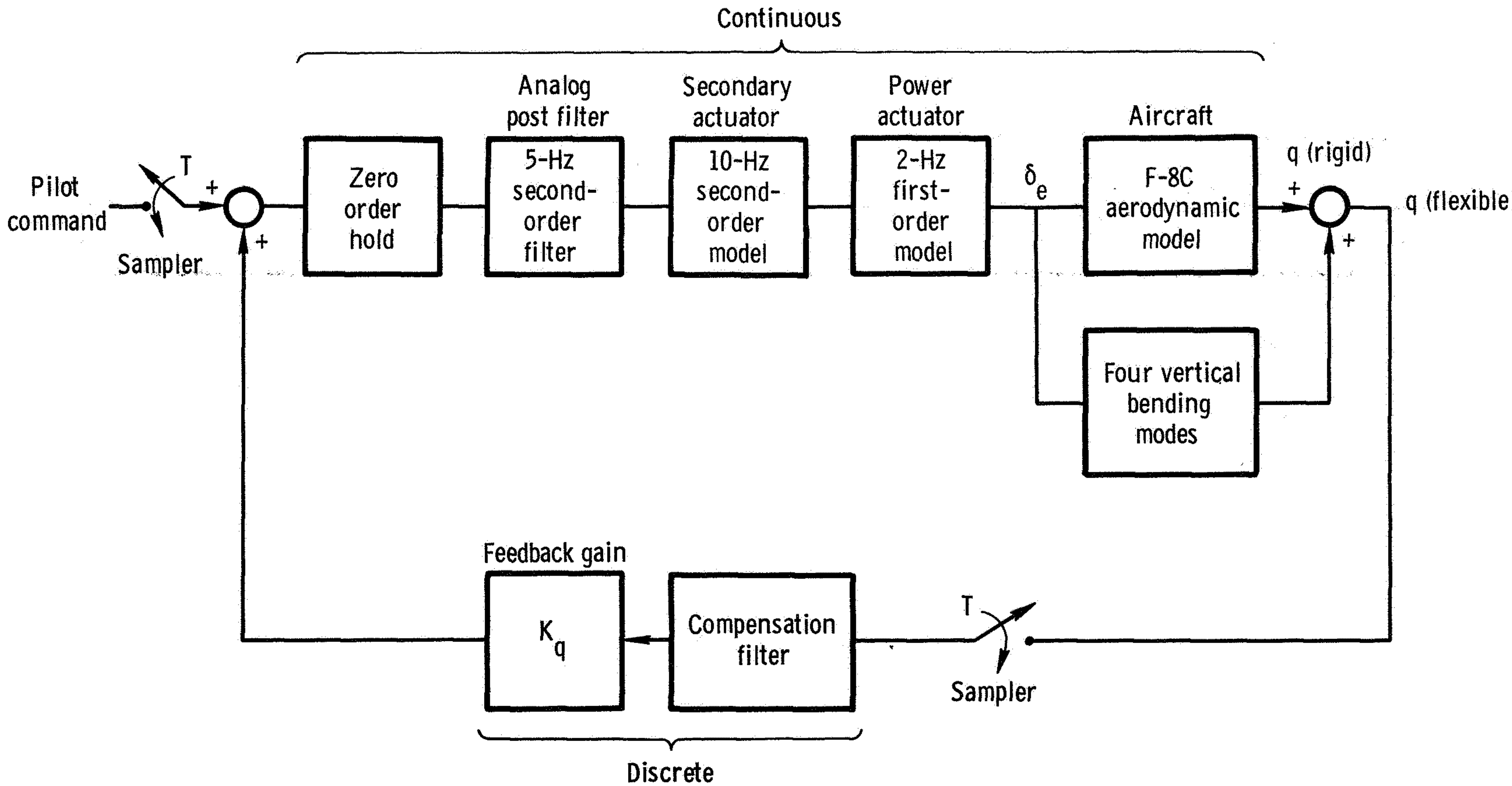
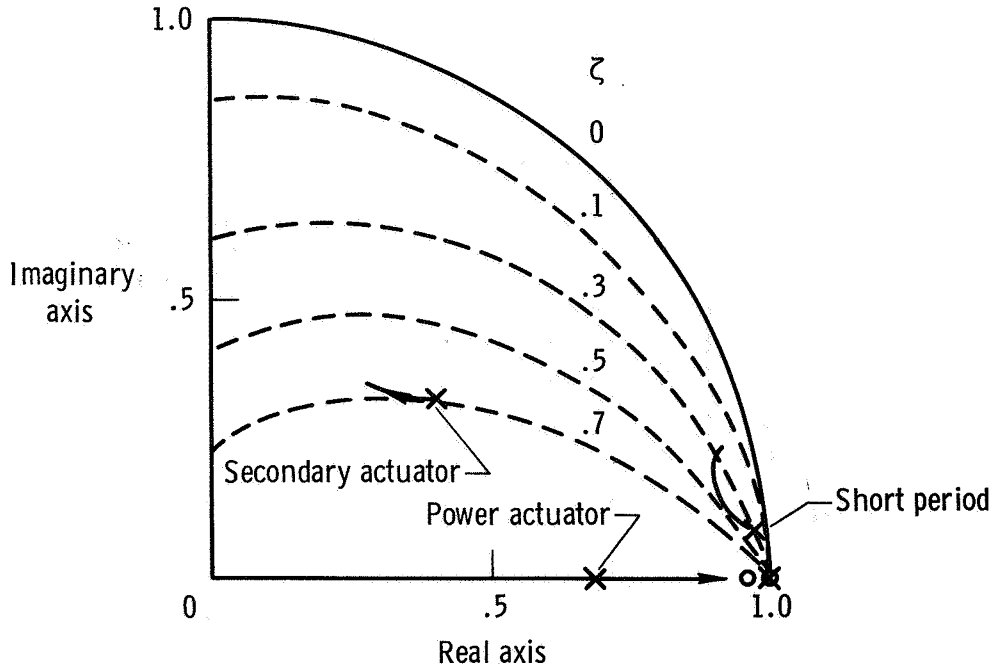
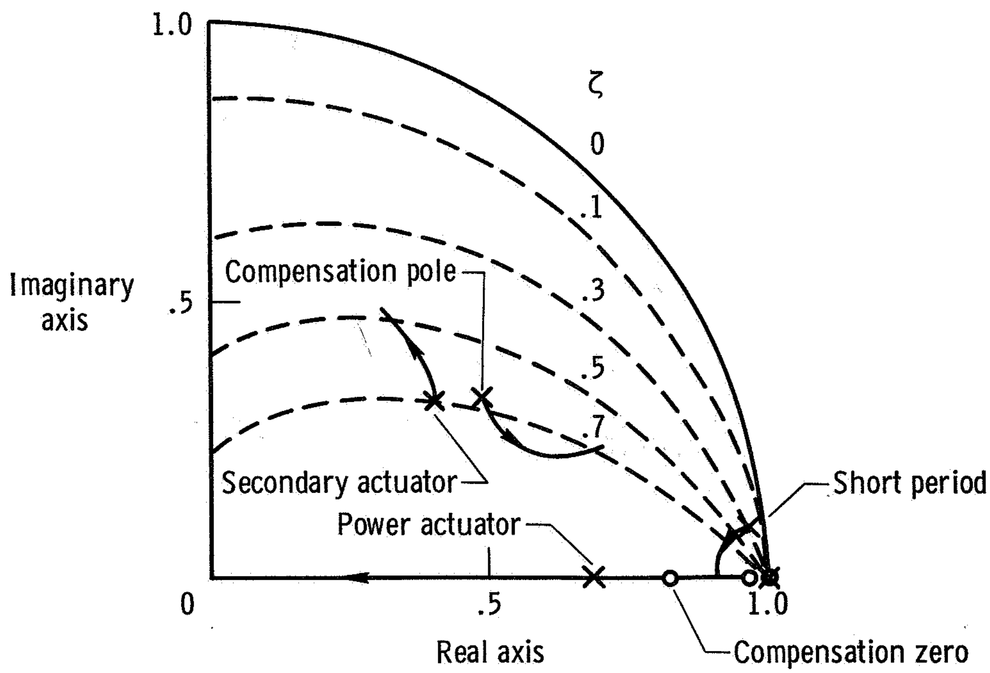


Figure 6. Linear system model for pitch SAS analysis.



(a) Uncompensated system locus.



(b) Locus with lead-lag compensation.

Figure 7. Sampled-data system design.

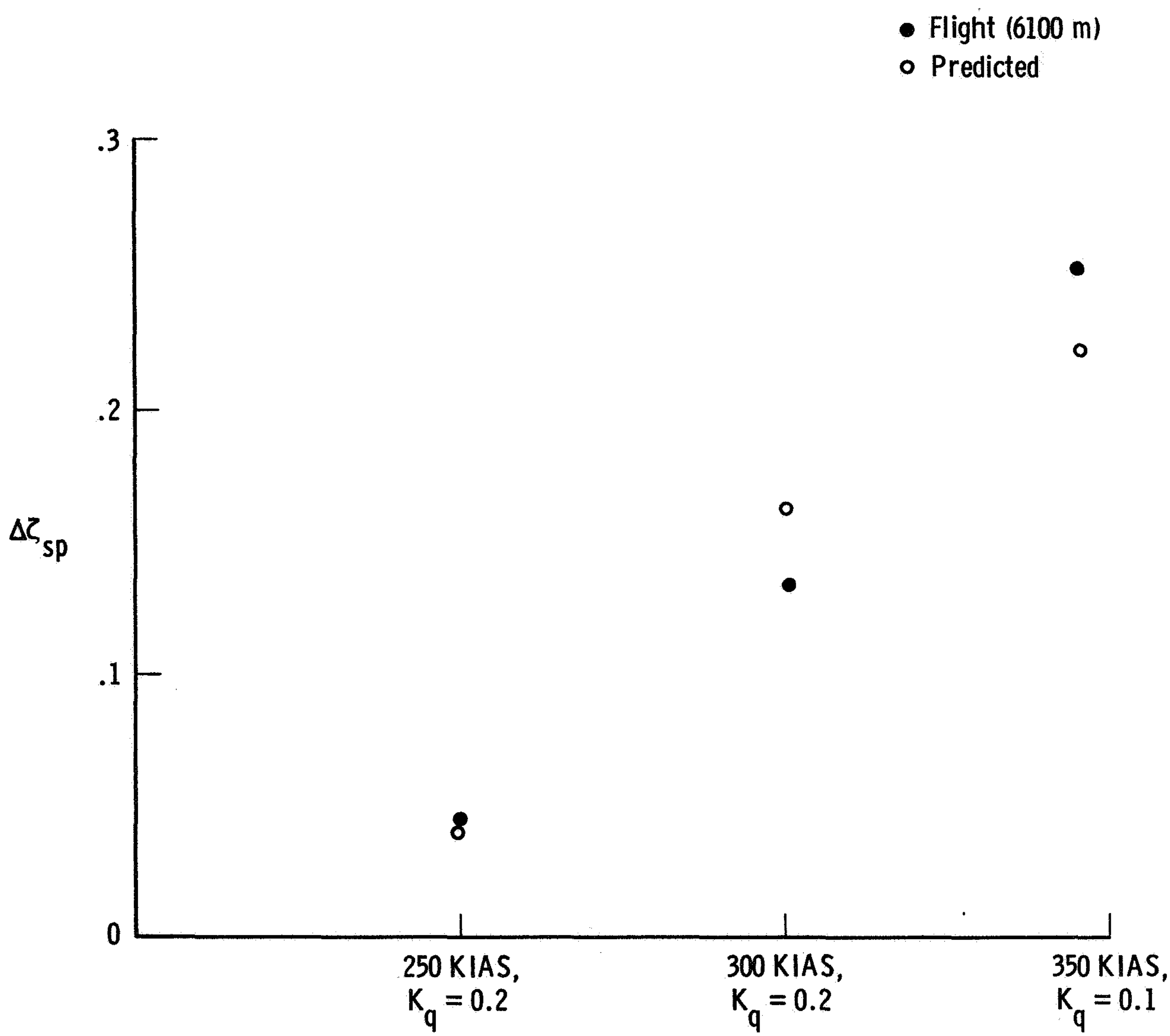


Figure 8. Improvement in short period damping ratio for lead-lag filter.

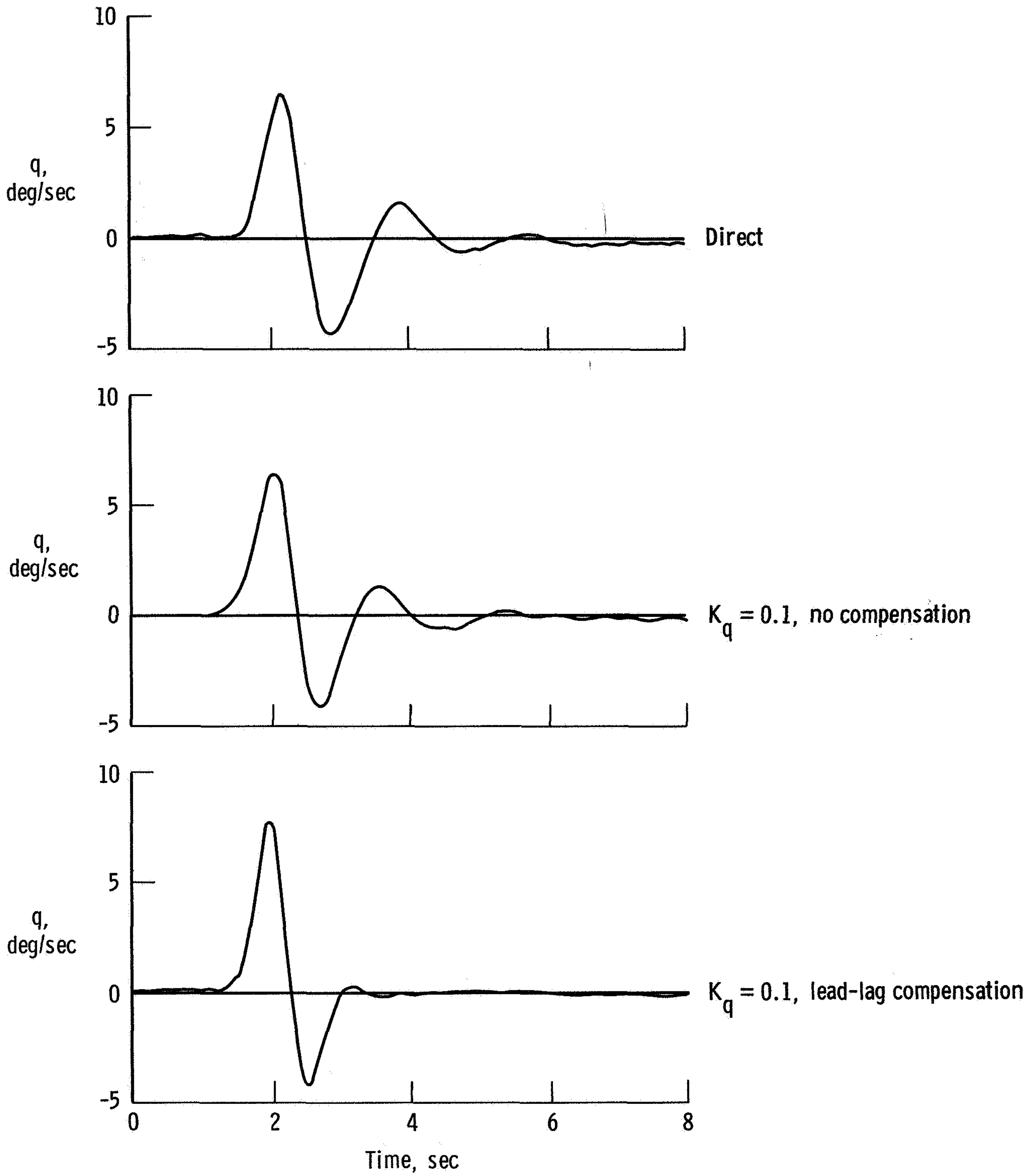
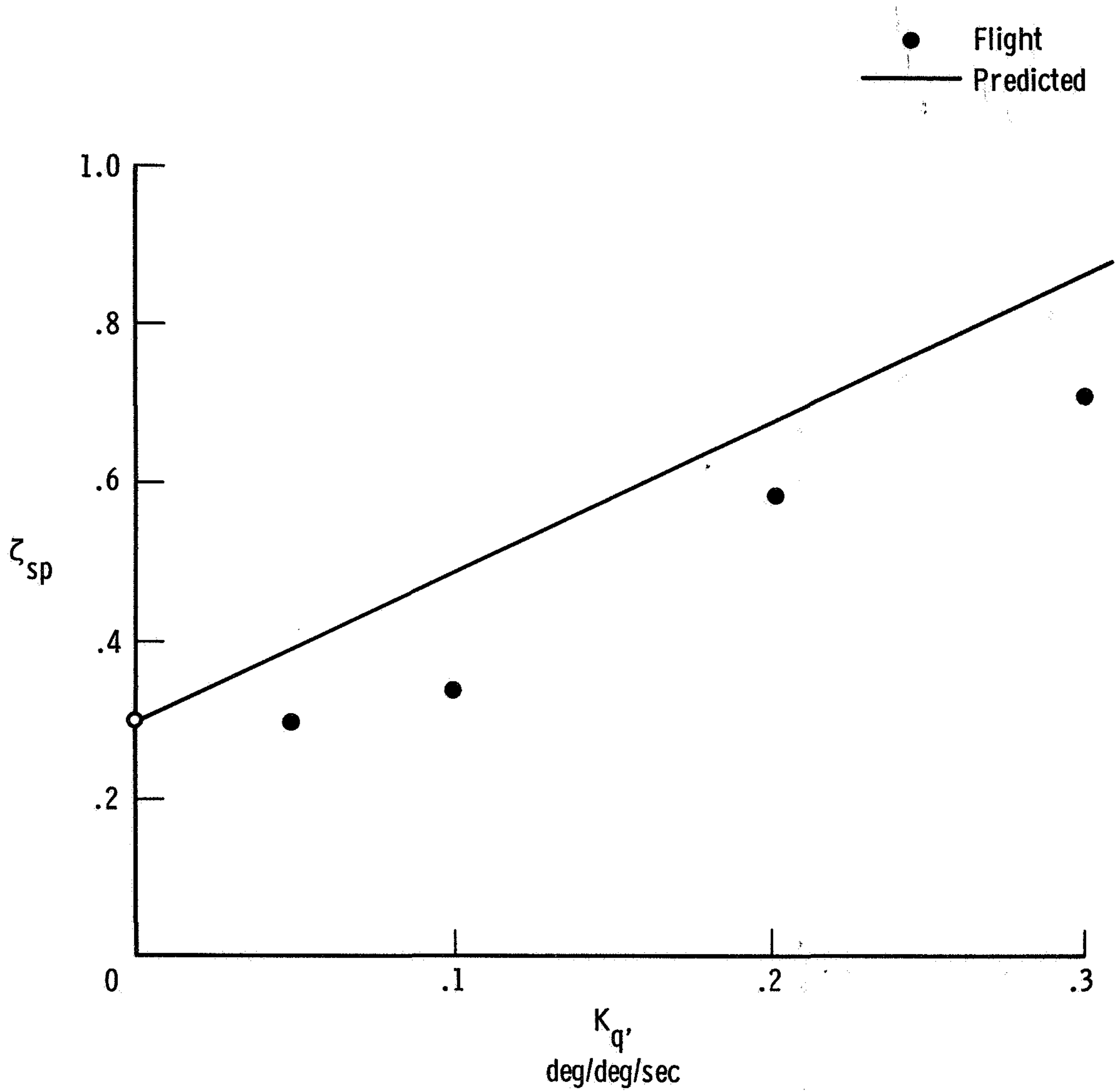
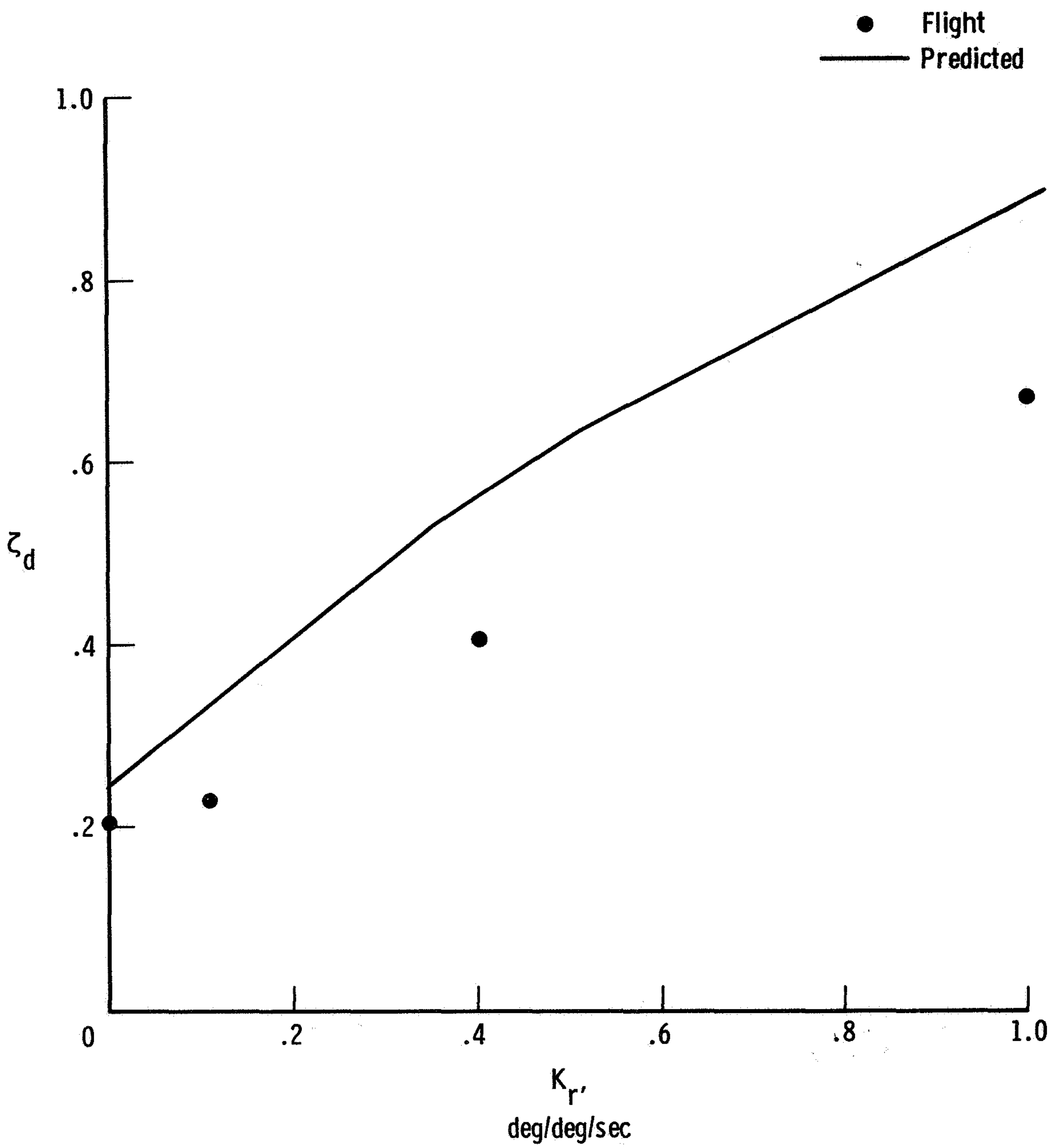


Figure 9. Effect of digital stability augmentation. Pitch SAS; 350 KIAS; 6100 m.



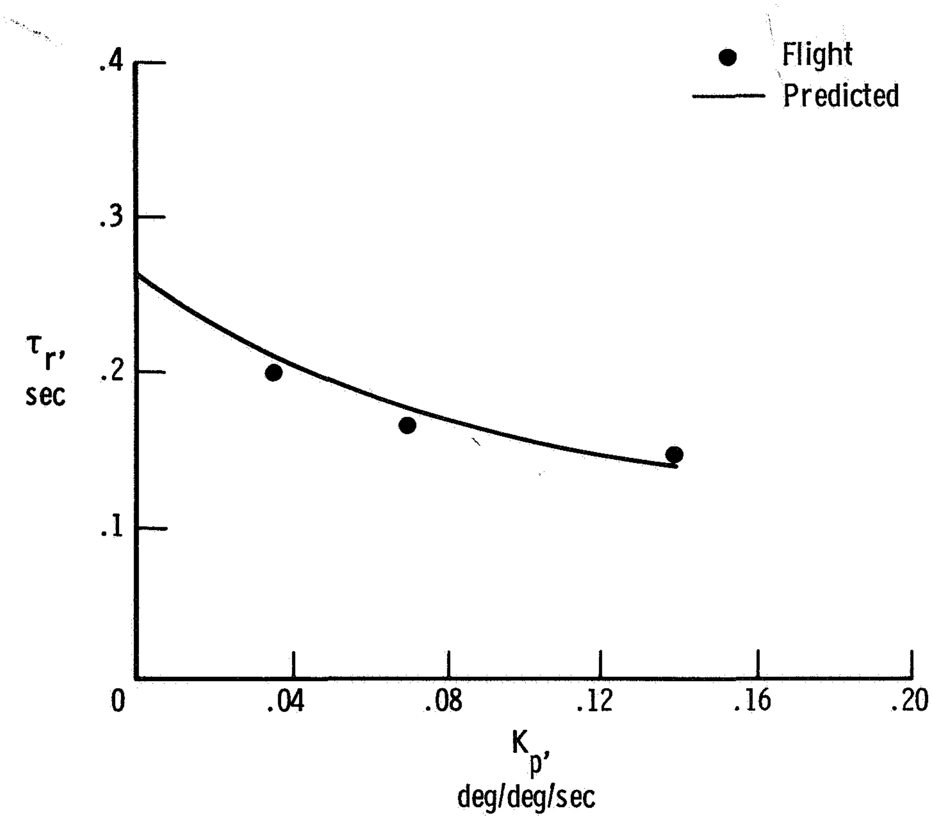
(a) Pitch SAS, 300 KIAS, 6100 m, 10,890 kg.

Figure 10. Comparison of in-flight and predicted digital SAS performance.



(b) Improvement in Dutch-roll damping for increase in yaw rate gain. 250 KIAS, 6100 m.

Figure 10. Continued.



(c) Improvement in roll mode time constant for increase in roll damper gain. 250 KIAS.

Figure 10. Concluded.

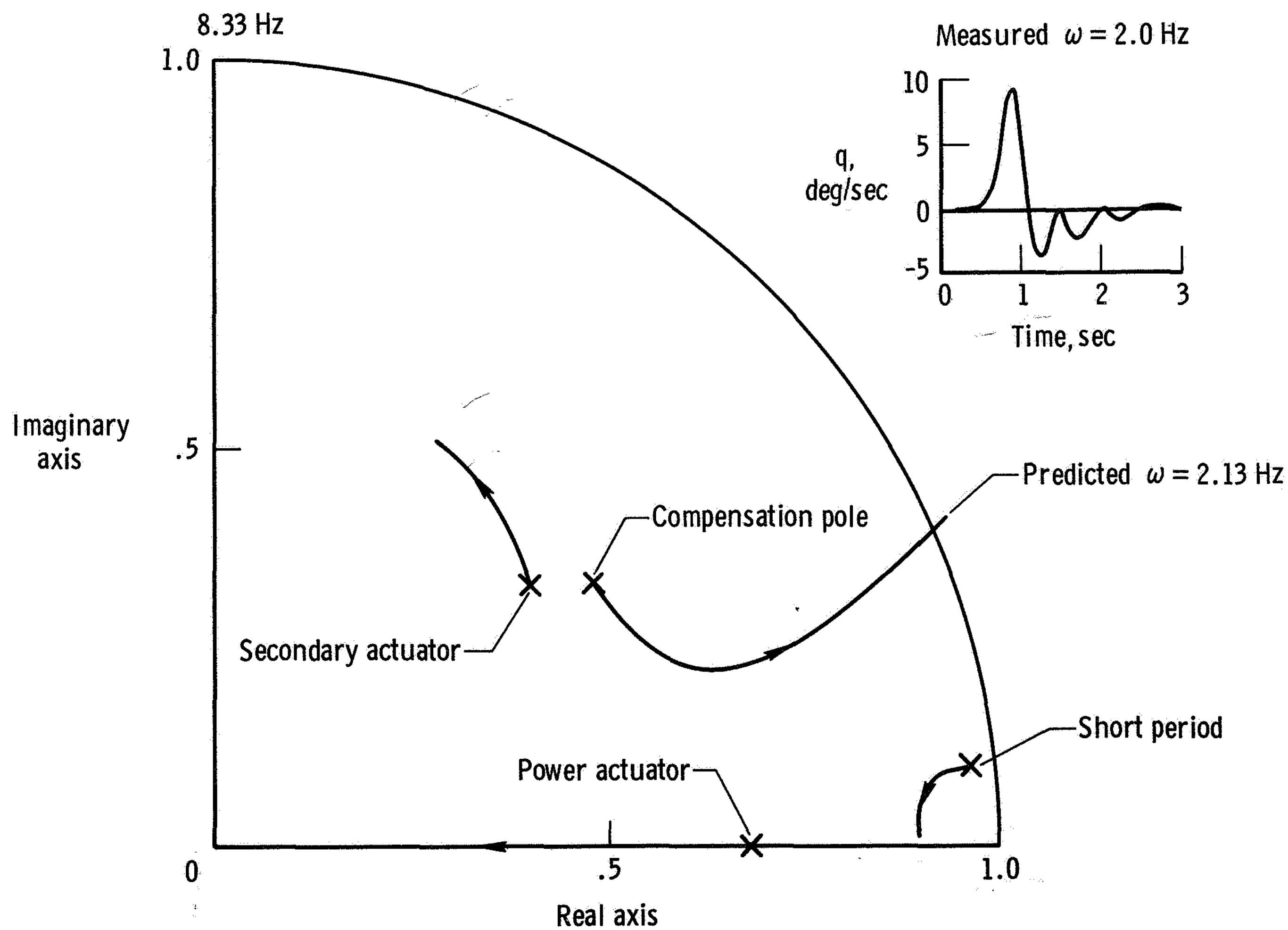
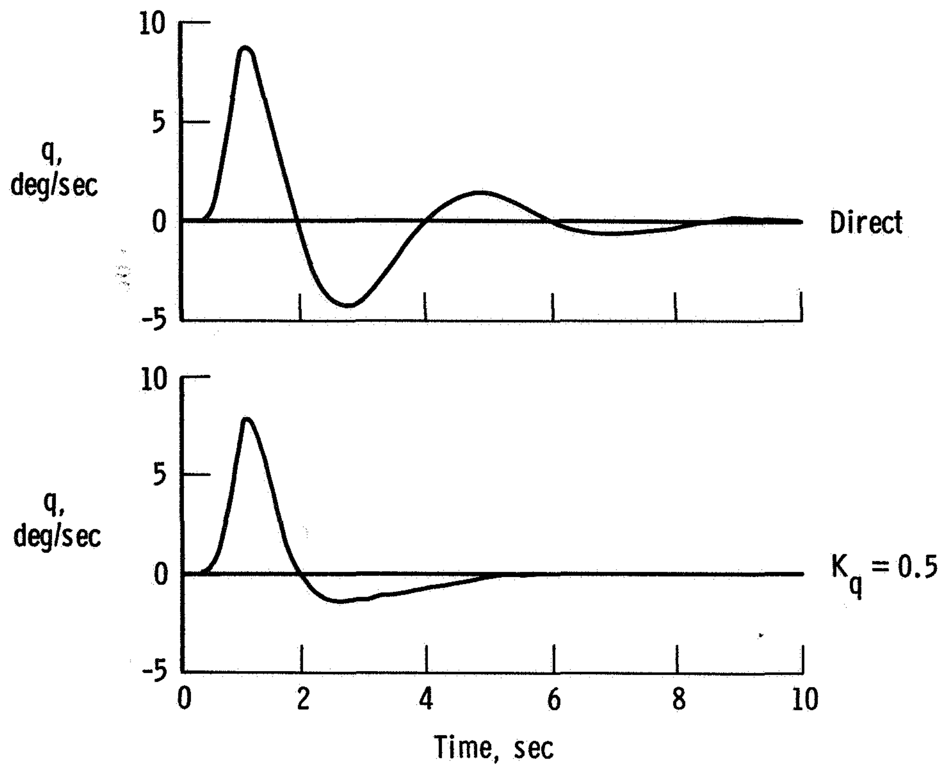
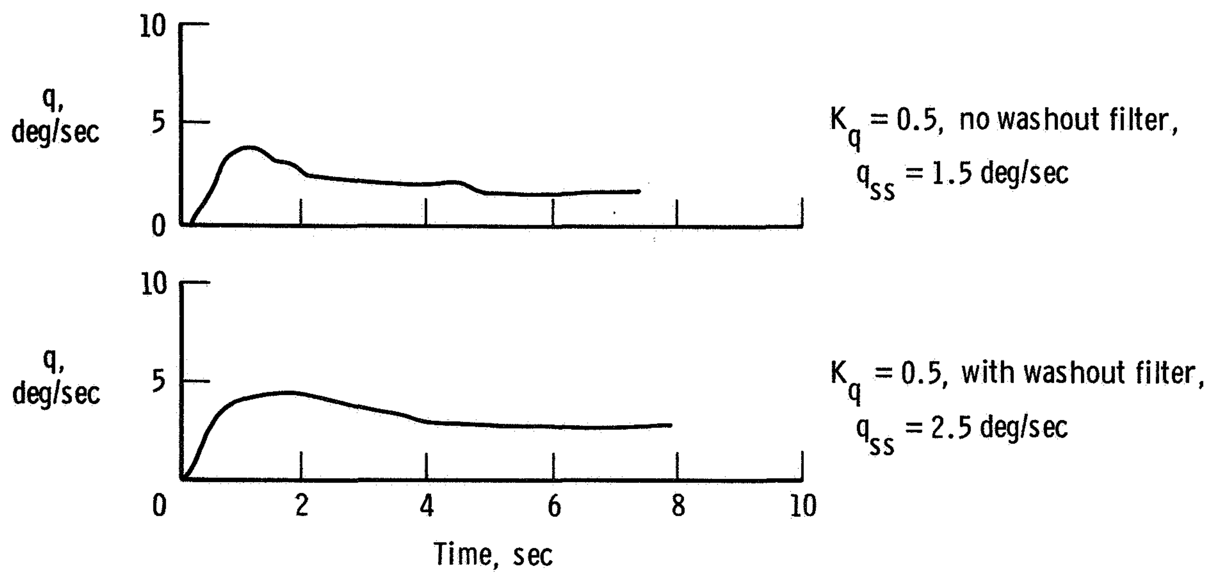


Figure 11. Prediction of system instability at high gain. 350 KIAS; 6100 m.

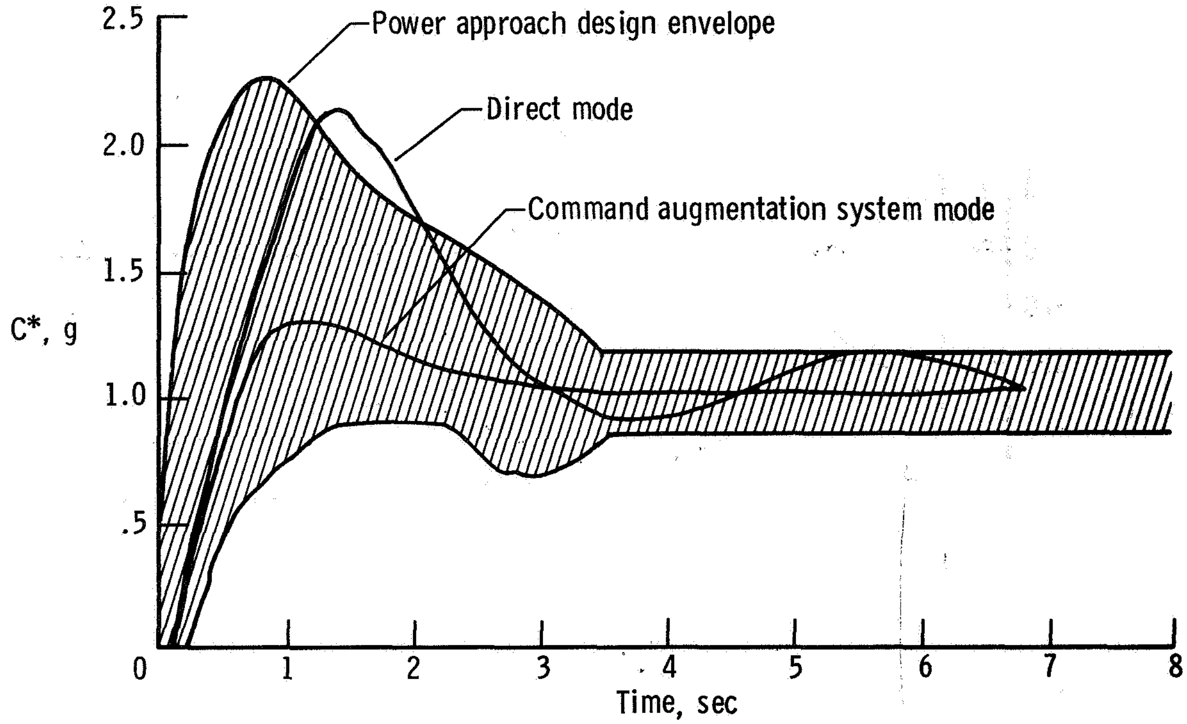


(a) Effect of SAS on damping ratio. Pulse pilot input.

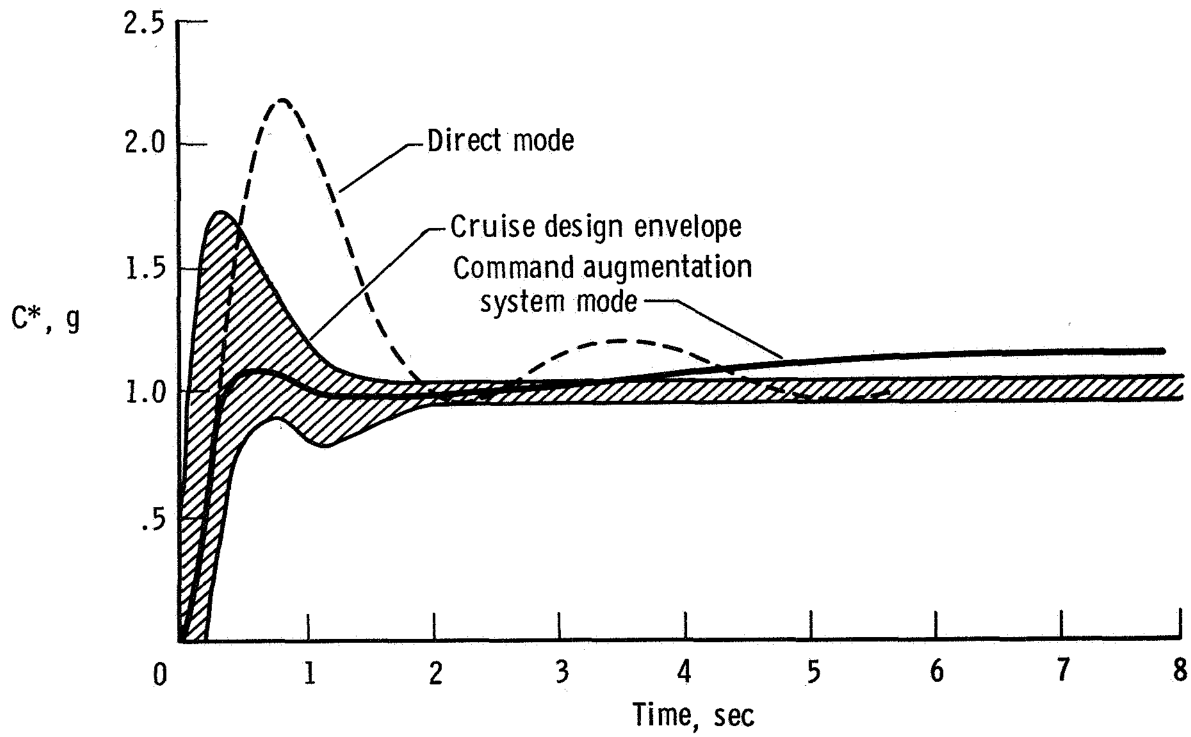


(b) Effect of washout filter on steady state response. Step pilot input.

Figure 12. Pitch SAS operation at low speed. 180 KIAS; wing up.

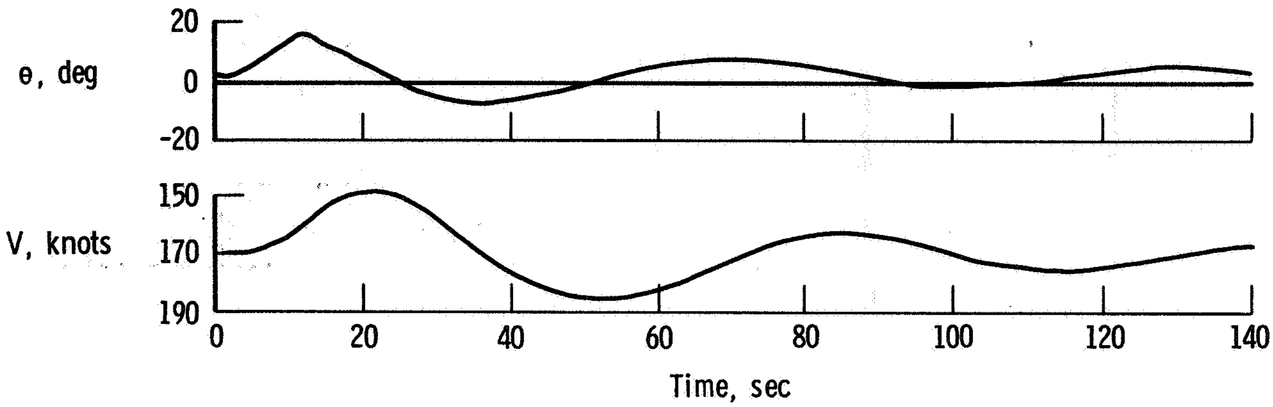


(a) 180 KIAS, 4570 m, $K_{C*} = 2.0$ deg/g.

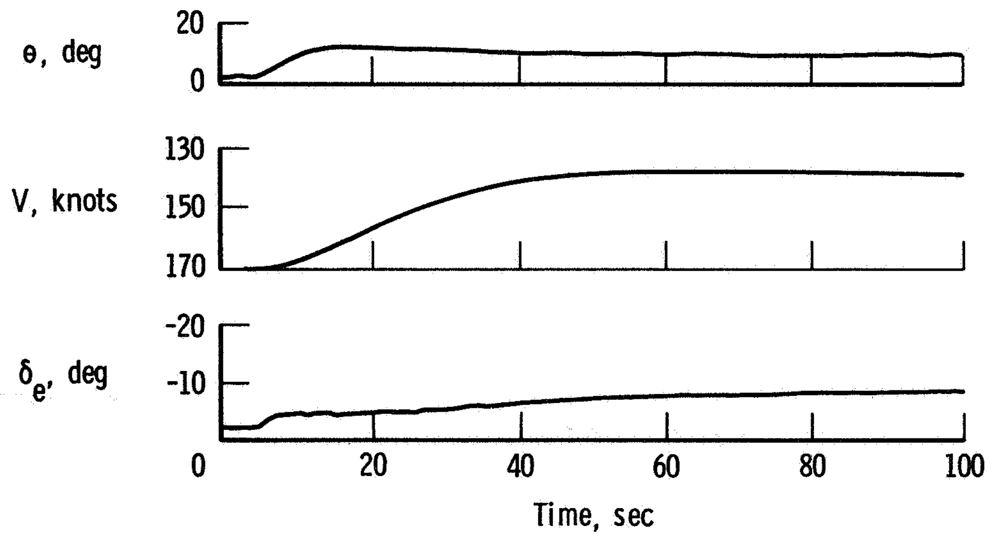


(b) 250 KIAS, 4570 m, $K_{C*} = 1.5$ deg/g.

Figure 13. C* response of F-8 DFBW aircraft.



(a) Direct mode, hands off.



(b) CAS mode, hands off, $K_{C*} = 2.0$ deg/g.

Figure 14. F-8 DFBW phugoid response.

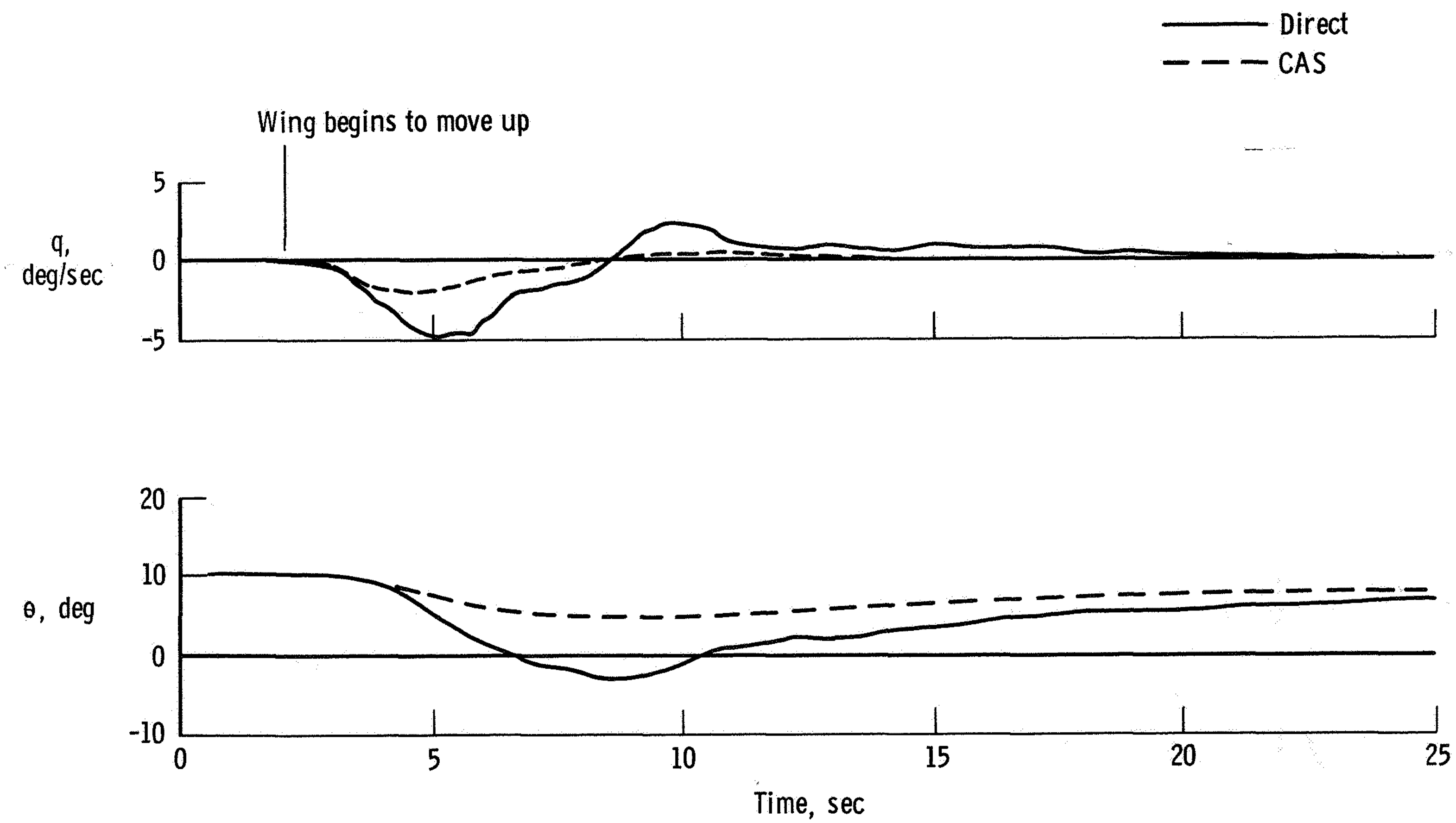
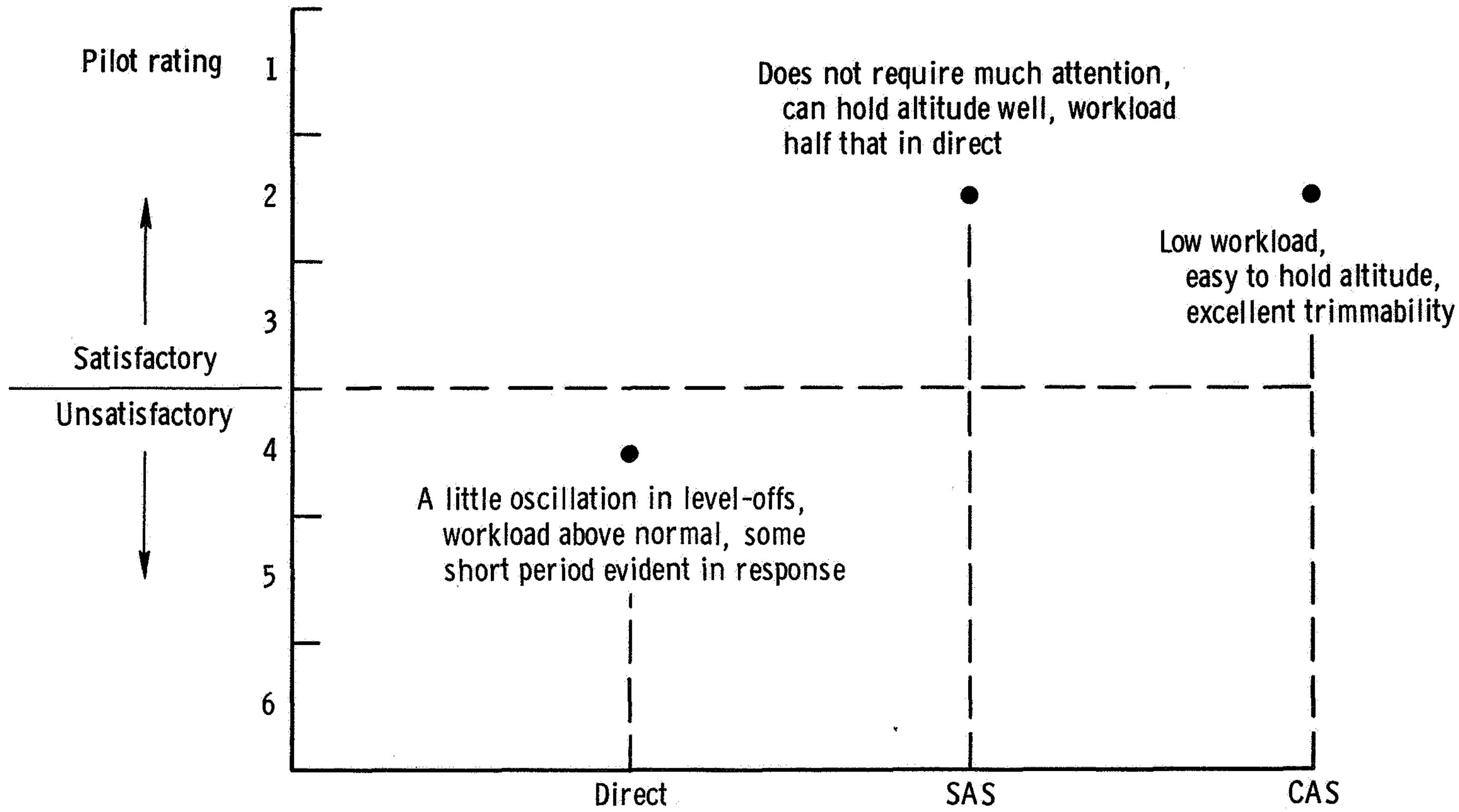
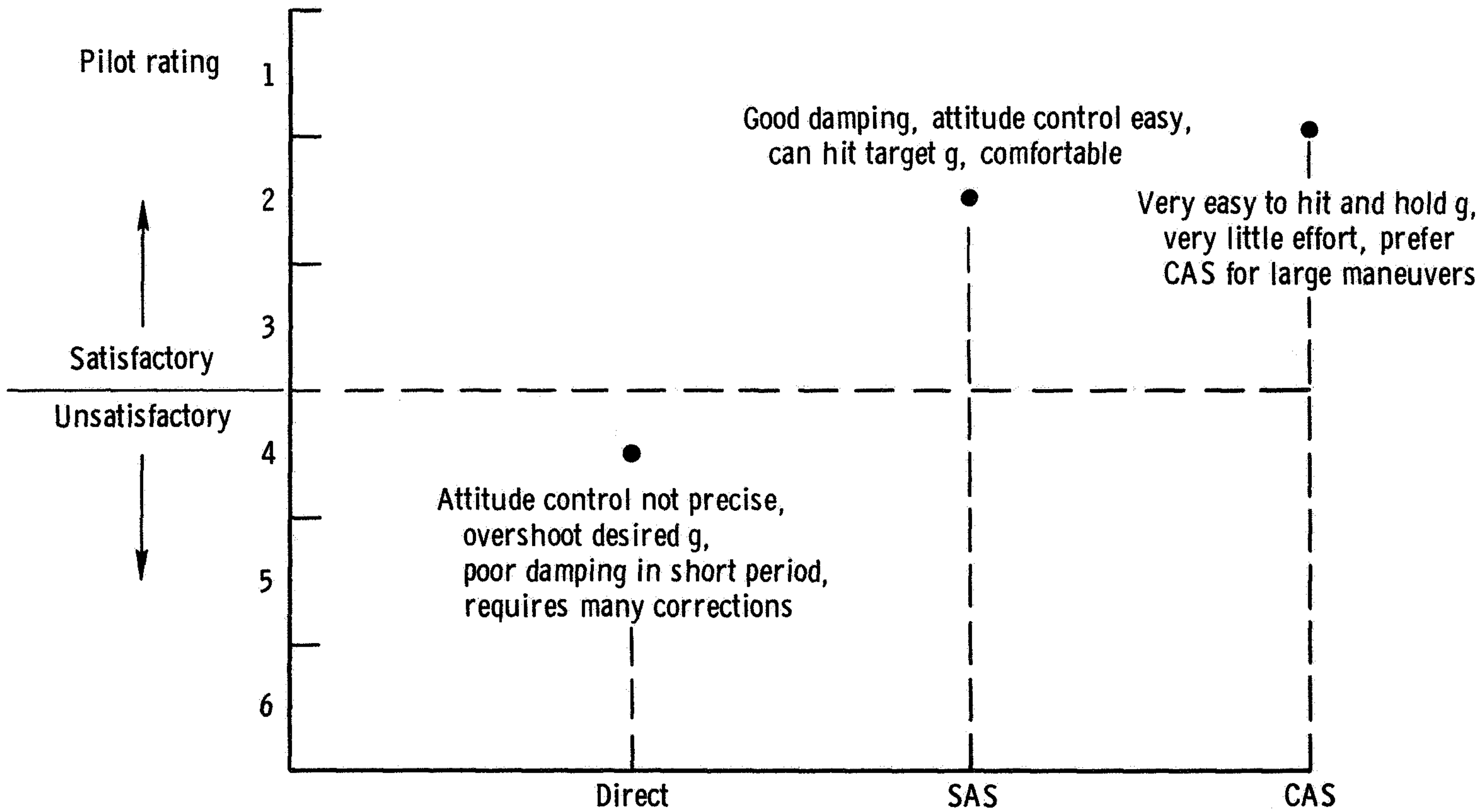


Figure 15. Effectiveness of CAS mode in reducing transient due to down-to-up wing transition.



(a) Instrument maneuvers. 250 KIAS, 4570 m.

Figure 16. Longitudinal handling qualities summary – pitch mode.



(b) Large maneuvers. 250 KIAS, 4570 m.

Figure 16. Concluded.

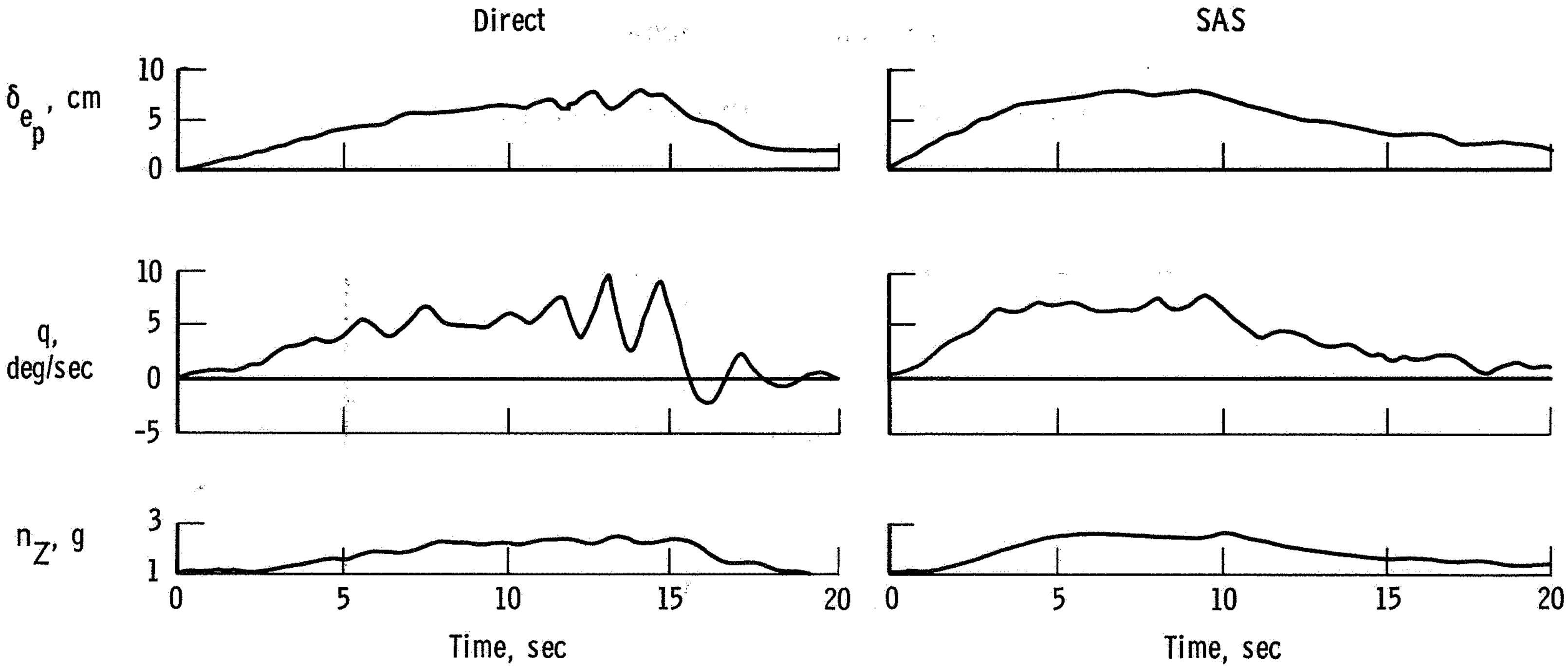
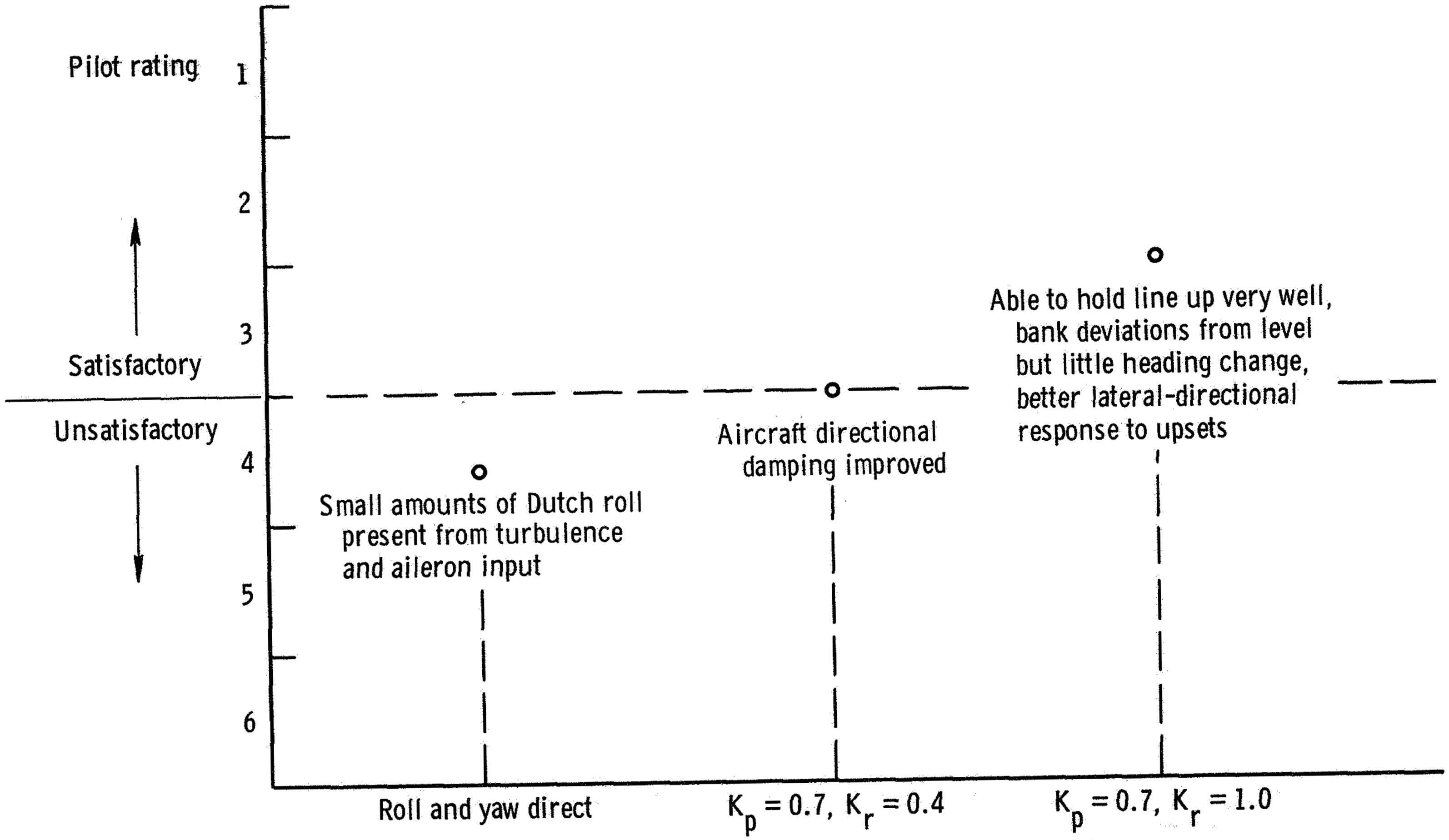
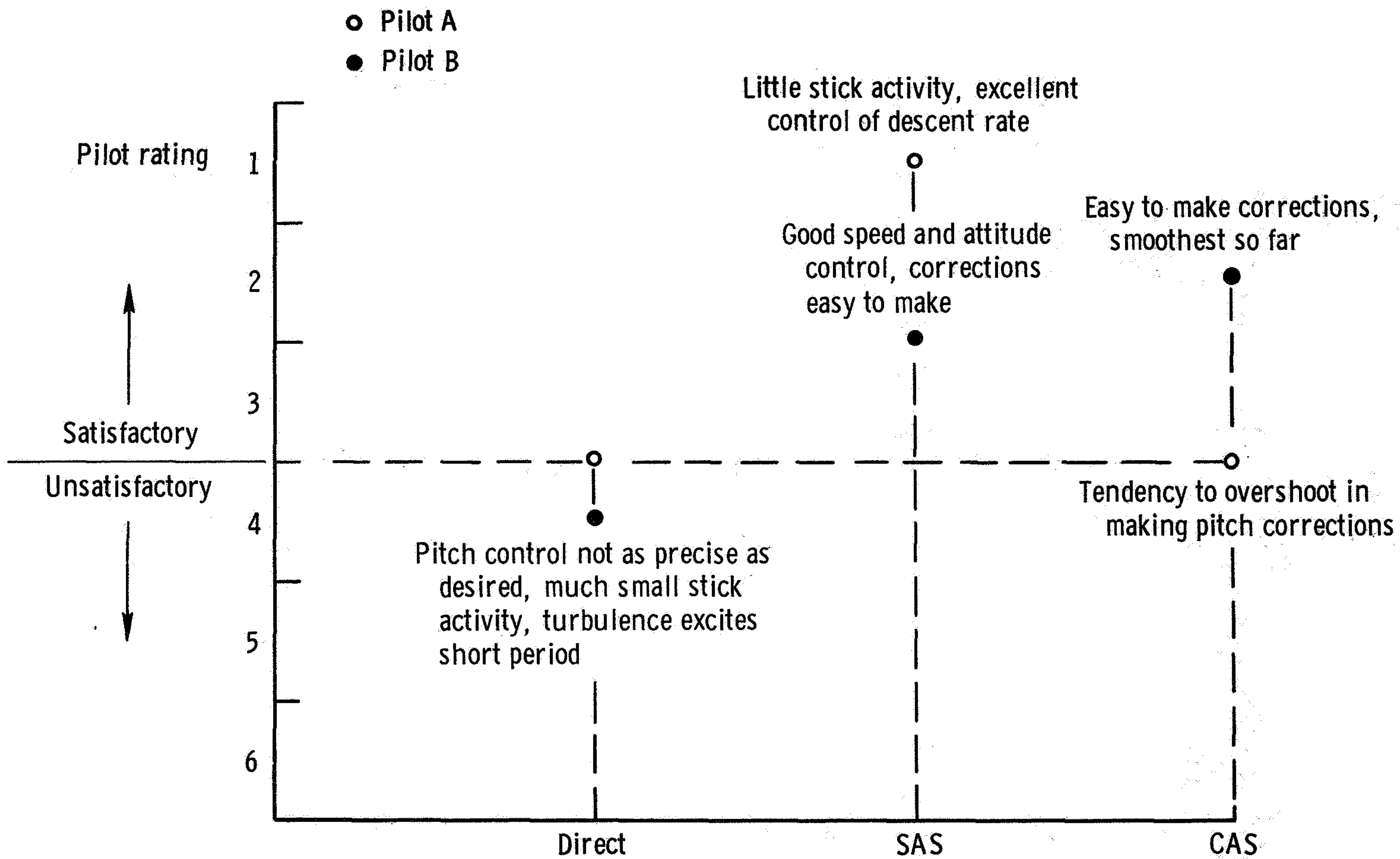


Figure 17. Effect of pitch SAS in wind-up turn. 300 KIAS; 6100 m.



(a) Lateral-directional.

Figure 18. Ground-controlled approach handling qualities summary. Light-to-moderate turbulence.



(b) Longitudinal, pitch mode.

Figure 18. Concluded.

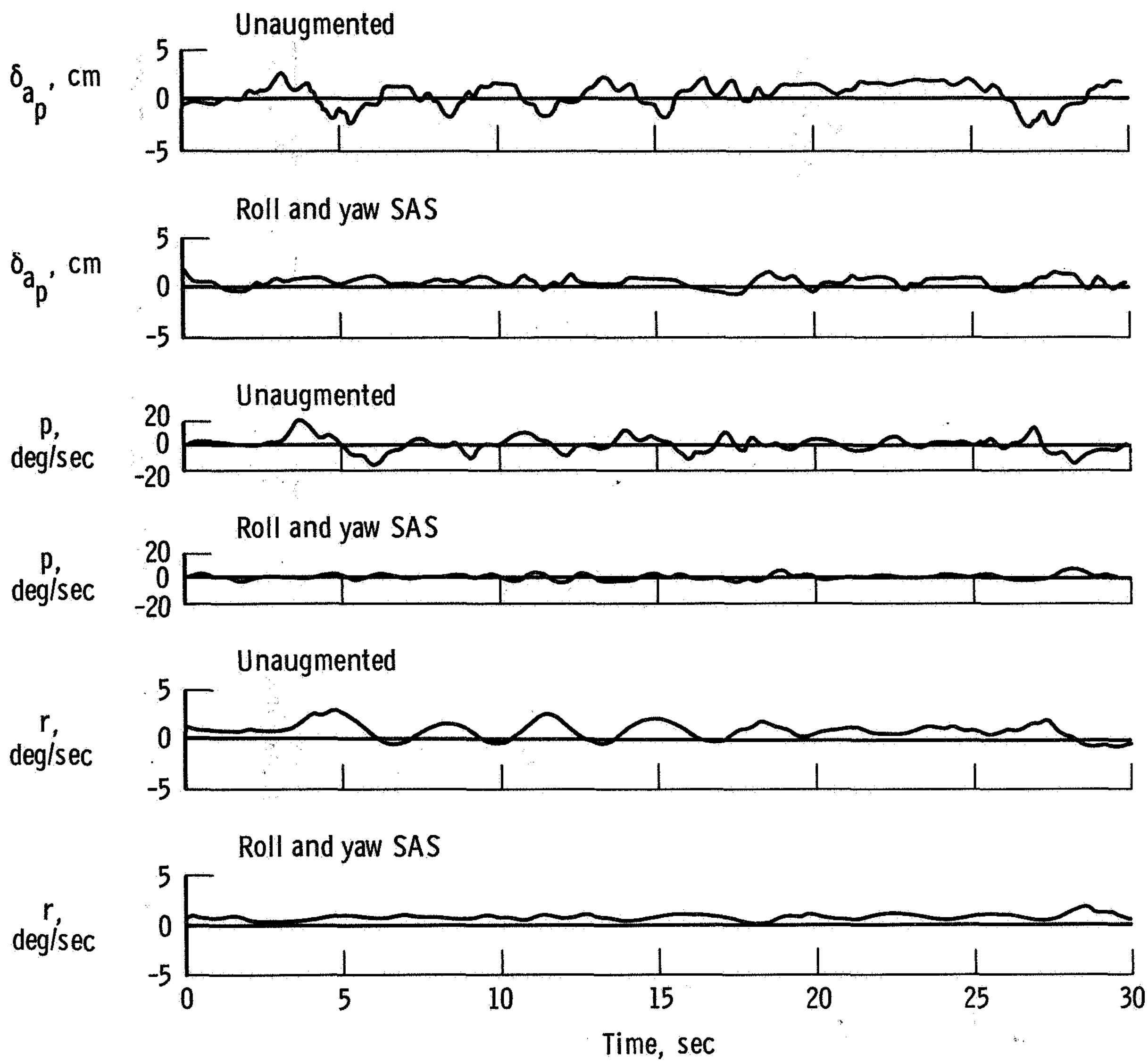


Figure 19. Improvement in lateral-directional flying qualities in gunsight tracking maneuver.

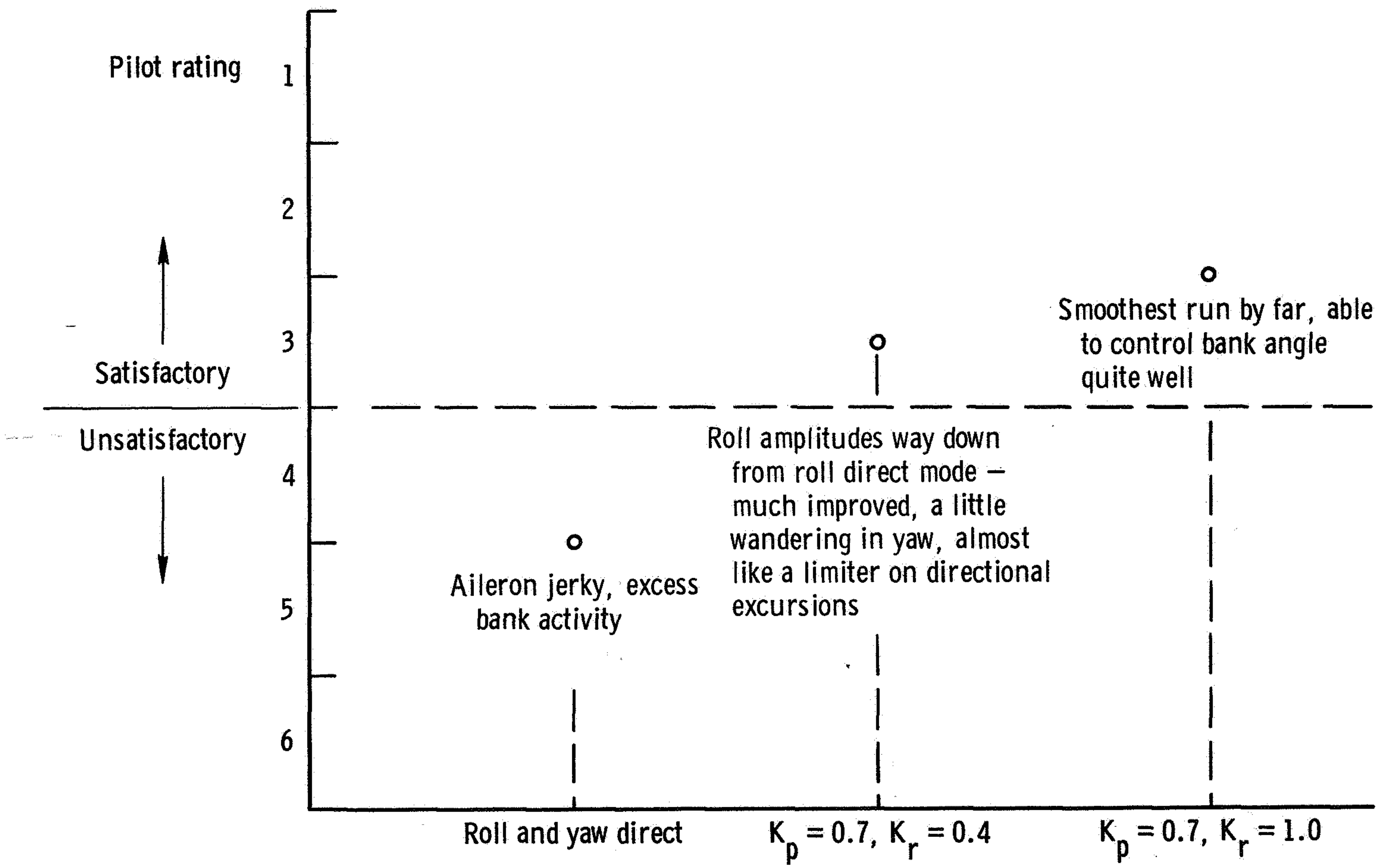


Figure 20. Pilot opinion of lateral-directional flying qualities in gunsight tracking maneuver.

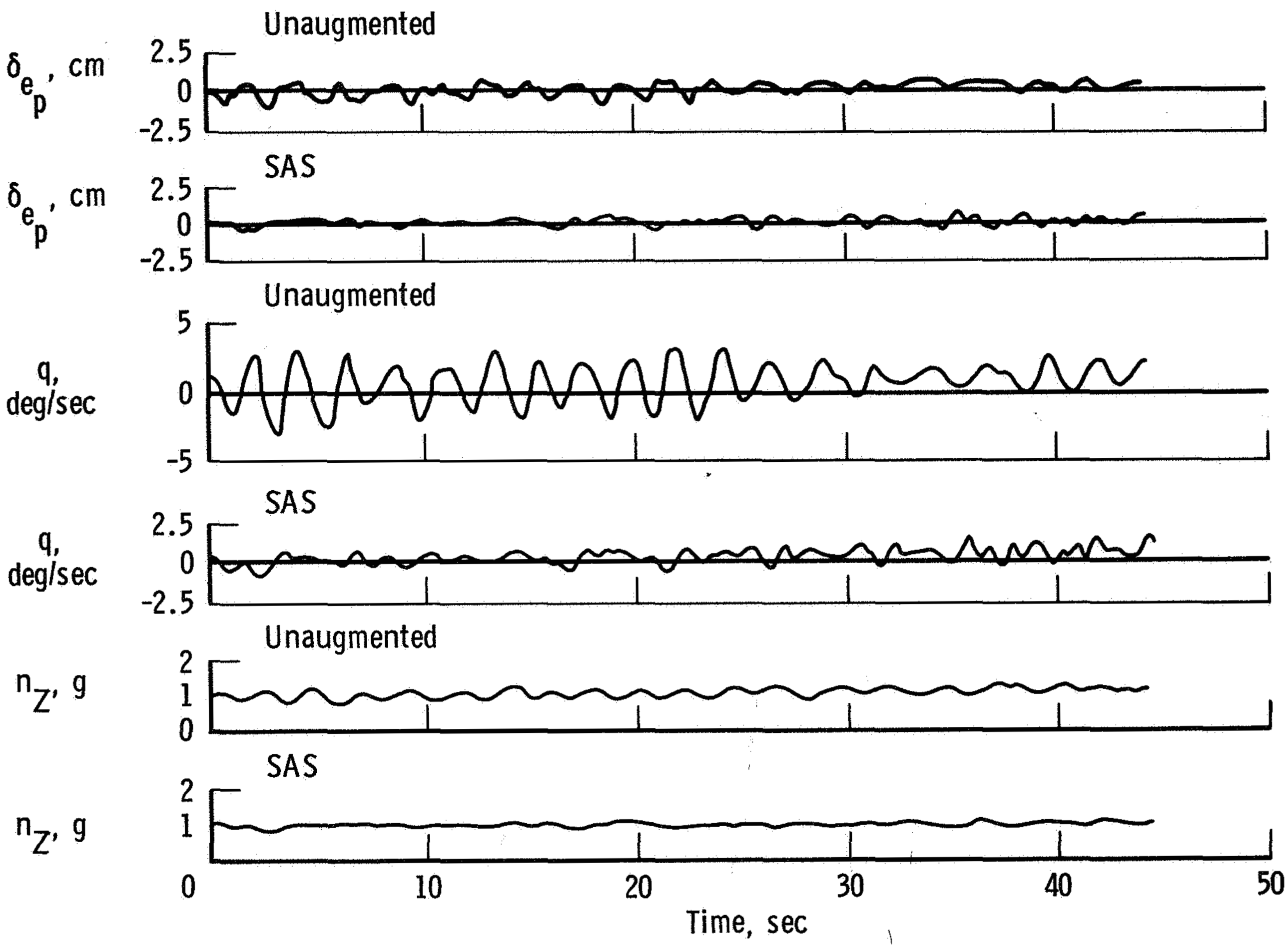
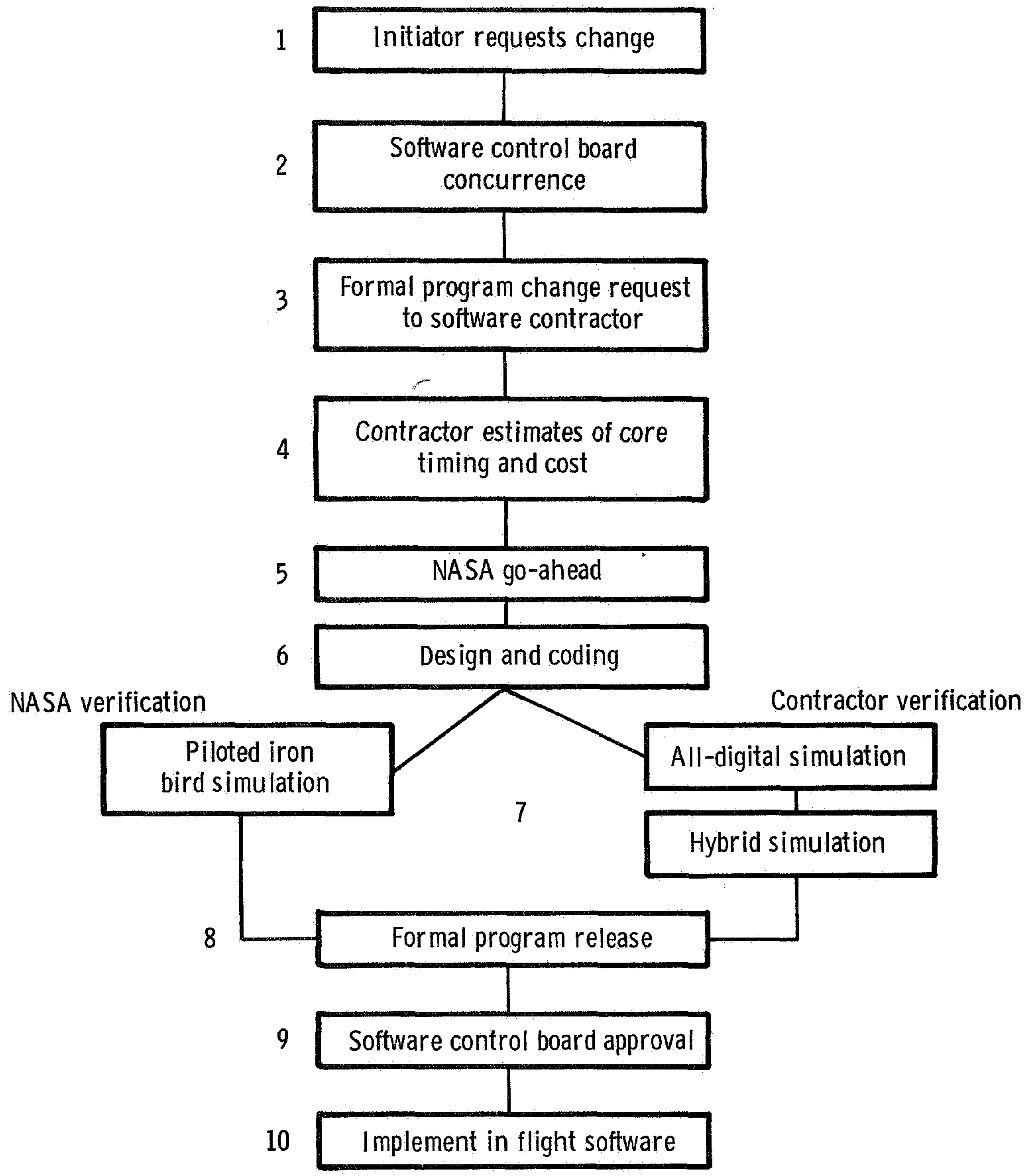
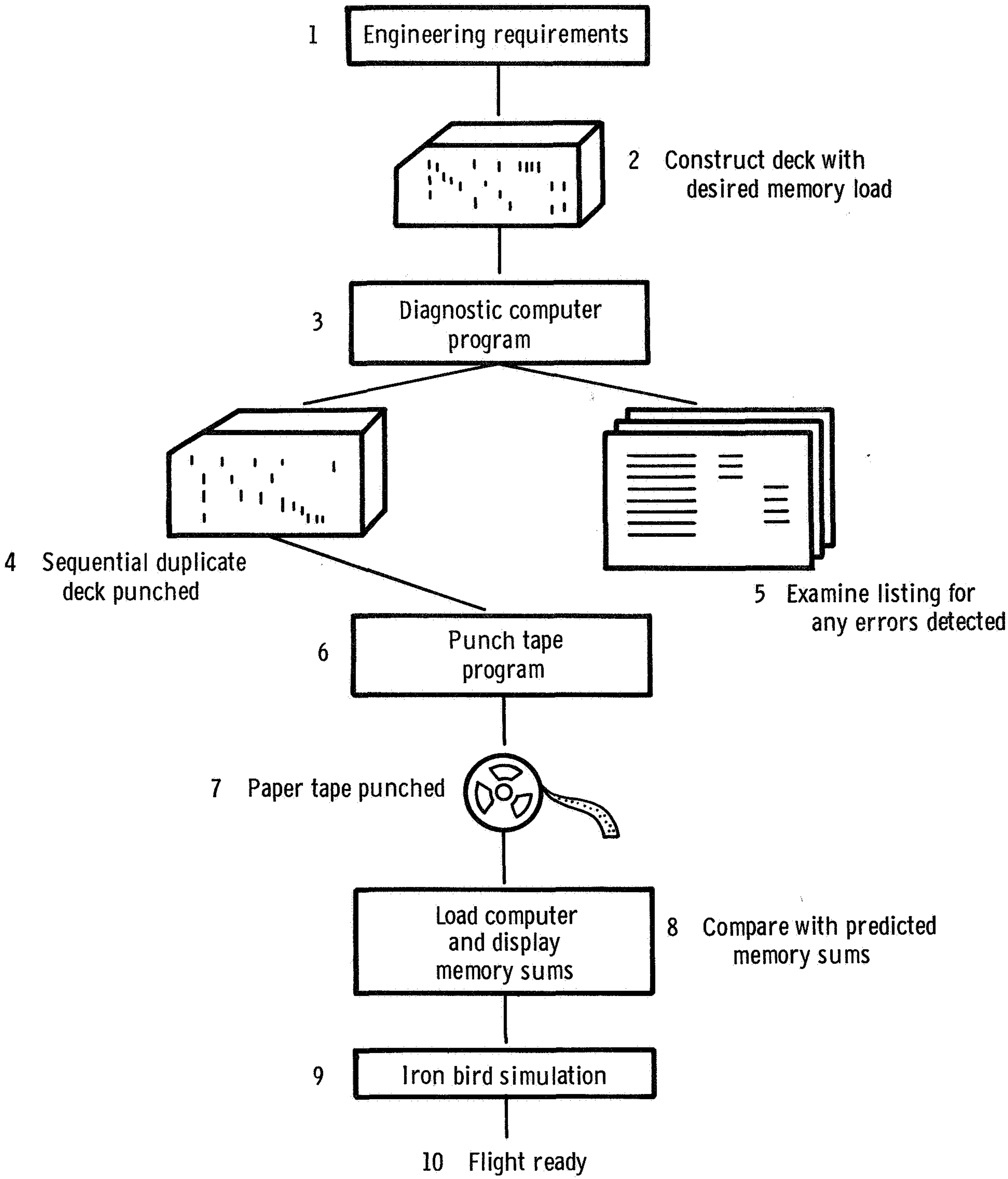


Figure 21. Pilot activity and airplane response in formation flight.



(a) Program change procedure.

Figure 22. Software management.



(b) Procedure for new erasable memory load.

Figure 22. Concluded.