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DESIGN AND DEVELOPMENT EXPERIENCE WITH A DIGITAL FLY-BY-WIRE CONTROL SYSTEM IN AN F-8C AIRPLANE

Dwain A. Deets
NASA Flight Research Center

SUMMARY

To assess the feasibility of a digital fly-by-wire system, the mechanical flight control system of an F-8C airplane was replaced with a digital primary system and an analog backup system. The Apollo computer was used as the heart of the primary system. This paper discusses the experience gained during the design and development of the system and relates it to active control systems that are anticipated for future civil transport applications.

INTRODUCTION

A major deterrent to the application of active controls to transport aircraft has been a lack of experience in designing highly reliable flight control augmentation systems and verifying them in flight. Digital fly-by-wire technology has the potential for providing the necessary reliability while still offering design flexibility. To assess the feasibility of a digital fly-by-wire system, the NASA Flight Research Center conducted a flight research program in which the mechanical flight control system of an F-8C airplane was replaced with a digital primary system and an electrical analog backup system.

This paper describes the fly-by-wire system and the design and development of the digital primary system. The system and the design procedures are assessed in light of similar applications being contemplated for future transport aircraft.

The paper was written in conjunction with references 1 to 3, which discuss the backup control system, software management, and results from the flight tests.

SYMBOLS

K	proportionality constant
KAZ	normal acceleration feedback gain to stabilizer, deg/g
KG	stick or rudder pedal gearing constant, deg/m
KP	roll rate feedback gain to ailerons, deg/deg/sec
KQ	pitch rate feedback gain to stabilizer, deg/deg/sec
KR	yaw rate feedback gain to rudder, deg/deg/sec
q_k	pitch rate at k^{th} sample, deg/sec
s	Laplace transform variable
T	sample period, sec
z	complex variable, e^{sT}
θ_k	pitch angle at k^{th} sample

TEST AIRPLANE

An F-8C airplane (fig. 1) was selected for use in flight testing a digital fly-by-wire system. Several characteristics of the airplane made it suitable for this test program. The handling qualities without control augmentation were acceptable for emergency operation, thus backup control could be provided through a relatively simple system. In addition, the airplane had enough space for the system's components, and the capacity of the hydraulic systems was adequate.

Some features of the F-8C airframe had an impact on the fly-by-wire system design. The variable-incidence wing moves up 8° for low-speed flight. This rotates the fuselage nose down relative to the free airstream, improving the pilot's visibility during the approach. Several functions within the flight control system are programed as a function of wing position. For example, the horizontal stabilizer is driven 5° leading edge up when the wing is raised.

The F-8C airplane does not have independent flap surfaces, so the ailerons are driven collectively to serve as flaps through a mechanical linkage independent of the primary control system. For this program the linkage was disconnected and the ailerons were driven to the drooped flap position through the fly-by-wire system.

The original F-8C flight control system had a direct linkage in pitch (no augmentation), and roll and yaw stability augmentation systems (SAS) (figs. 2(a) and 2(b)). The yaw SAS included an aileron-to-rudder interconnect with a gain programed as a function of stabilizer position.

DATA ACQUISITION SYSTEM

A standard NASA pulse code modulation (PCM) system was installed to record airplane motion, pilot input, and fly-by-wire system parameters external to the computer. Seventy-seven channels of 9-bit data were recorded on an onboard tape and telemetered to a ground station for real-time monitoring. Eight of the channels were digital words indicating the state of 57 discrete values from the fly-by-wire system. Although excellent for automated data reduction, the PCM system proved to be unsuitable for investigating the effects of analog-to-digital and digital-to-analog quantization. The resolution of the PCM system was on the same order as that of the digital control system quantization, which made it difficult to isolate the effects of the control system quantization.

Internal digital computer parameters were recorded on the onboard PCM tape recorder. One-hundred-word pairs were strobed out every 2 seconds and recorded for postflight analysis. The word lists were resident in the software onboard the airplane and could be reprogramed during the flight tests. This recording system was used primarily to determine the gross status of the computed parameters within the flight control system; the strobe rate was inadequate for tracing individual parameters each control computational cycle.

IRON BIRD SIMULATOR

An iron bird simulator played an important role in the development of the fly-by-wire system. The simulator consisted of another F-8C airplane, in which all the digital fly-by-wire flight control hardware was installed, tied in with a hybrid computer and appropriate interface equipment (fig. 3). The digital fly-by-wire hardware was flight qualified and served as spares for the flight vehicle. The F-8C aerodynamics and bending modes were modeled by using the digital portion of the hybrid computer for the aerodynamics and the analog portion for the bending modes.

FLY-BY-WIRE SYSTEM

The fly-by-wire system had a digital primary control system and an electrical analog backup control system. Components of the fly-by-wire system are shown in figure 4, and the location of the components in the F-8C airplane is illustrated in figure 5.

A simplex digital primary system and a triplex electrical analog backup system provided enough reliability that the mechanical system could be completely removed. The digital primary system consisted of a computer, inertial measurement unit, coupling data unit, and display and keyboard, all taken from the Apollo guidance and navigation system. Reference 4 provides details on this equipment relative to the Apollo application. The triplex backup control system consisted of only surface position command electronics. Specially designed electrohydraulic secondary actuators interfaced the primary and backup electronic commands with the conventional F-8C control surface power actuators.

Components of the fly-by-wire system were part of the primary or backup system and, in some instances, were shared between the two systems. Individual components are described in the following sections according to function.

Computational

The Apollo computer performed all flight control computations in the primary control system. Characteristics of this computer are summarized in the following tabulation:

Read-only memory	36,864 words
Scratch pad memory	2,048 words
Word length	14 bits plus sign and parity
Number system	Fixed point, ones complement
Memory cycle time	11.7 microseconds
Computation time —	
Add	23.4 microseconds
Multiply	46.8 microseconds
Divide	81.9 microseconds

Although slow by today's standards, the computer could perform all flight control functions within 30 milliseconds. Flight control laws for the F-8C airplane were programed for the computer's hardwired memory and could not be changed after the memory was manufactured. However, flexibility was achieved by placing feedback gains, logic flags, digital filter coefficients, and other gain variables in the computer's scratch pad memory; 105 of these variables associated with the flight control system could be changed.

Control laws for the backup system were mechanized in triplex control electronic boxes, which can be considered to be special-purpose analog computers. Each electronic box contained 67 operational amplifiers. Requirements for high reliability in the flight environment dictated the use of ruggedized packaging and hardwired circuits. Consequently, flexibility for changing control laws was limited to gain and nonlinear constant changes; even these changes required replacement of hardwired resistors and diodes. Although the control system is considered to be an analog system, more than half of each electronic box containing individual channels of the analog electronics was devoted to logic elements, such as comparators between the redundant channels.

Pilot Interface

Vehicle control and trim inputs.—The center stick was the primary means of control for both the primary and the backup systems. It was connected to the basic F-8C mechanical feel system. Disconnecting the mechanical links to the control surface caused undesirable looseness in the lateral stick, so a viscous damper was added. Stick position was sensed through two separate linear variable differential transformers (LVDT) in each axis. Each LVDT had triplex windings. Two of the windings from one LVDT in each axis were used by the primary system; the third winding was for instrumentation. The second LVDT provided triplex inputs to the backup control system.

A minimum displacement, two-axis side stick was used as an alternate controller input to the backup system. This side stick was not part of the original fly-by-wire system; it was used only as a means of evaluating a "force type" side stick controller in an actual aircraft environment (ref. 1).

The rudder pedals provided inputs to both the primary and the backup systems. Similar to the center stick, they were connected to the F-8C mechanical feel system and used LVDT's for sensing rudder pedal position.

Trim commands for primary pitch and roll were made through a two-axis beeper switch on the center stick. Backup trim inputs in all three axes were made through separate spring-loaded toggle switches on the pilot's left-hand console. These toggle switches were also used to provide primary yaw trim and primary pitch and roll trim inputs if the center stick trim switch failed. When the side stick was being used, trimming was accomplished through a beeper switch on the side stick.

Fly-by-wire functional control.—Figure 6 shows the mode and power panel, which was the pilot's means of communicating with the primary system. The pilot was able to choose between several different control system modes simply by depressing the appropriate button. Additionally, he was able to change system gains according to the logic loaded in the software before the flight. Several primary system failure status lights were located across the top of the panel, and power switches and power status lights were located across the bottom. Individual axes could be transferred to backup through the backup control system (BCS) switches. The pilot could also transfer all axes to backup simultaneously by using a "paddle switch" on the center stick.

A servo engage panel on the left-hand console permitted the pilot to selectively engage or disengage each channel of each servo actuator. The panel provided control over both the backup and the primary systems. The status of the actuation system and the backup electronics was displayed on this panel.

Motion Sensing and Interface

Another component from the Apollo guidance and navigation system was the inertial measurement unit. Although angular body rates and linear accelerations were not measured directly in the Apollo application, the substitution of aircraft gyros and accelerometers for the fly-by-wire tests would have required a major system modification and would have sacrificed the integrity of the total Apollo system. An alternate approach was adopted: Body rates and linear accelerations were computed from the gimbal angles and the digital incremental velocity vector information the inertial measurement unit provided to the Apollo computer for use in the primary system.

A coupling data unit provided the interface between the inertial measurement unit and the Apollo computer in the Apollo guidance system and provided the same function for the primary system in the F-8C application. The angular resolution was 0.011°; however, the rate resolution was of more importance and was nonlinear with gimbal angle rate. Body angular rate estimation was directly related to gimbal angle rate resolution. Gimbal rate resolution was ±0.183 deg/sec for rates less than 4.4 deg/sec, and ±2.74 deg/sec for rates between 4.4 deg/sec and 70 deg/sec. The error was manifested as a random noise band of 2.74 deg/sec peak to peak for rates greater than 4.4 deg/sec. Acceleration was sensed by using pulse integrating pendulous accelerometers. The quantization level for the normal acceleration signals was 0.2g.

The coupling data unit contained several digital-to-analog converter channels, which made it possible to send the necessary primary system commands to the control surfaces. To protect against undetected failures between the computer and the surface actuators, dual signals were generated in each axis, beginning with dualized commands to dual digital-to-analog converters in the coupling data unit in each axis. The control surface drive signals were quantized to ±384 levels, which is somewhat less than a full 9-bit word.

Control Surface Actuation

Similar actuation systems were used in each axis. Each actuation system had a secondary actuator and a power actuator. Separate sets of actuators were used for the left and right horizontal stabilizers and ailerons. A single set of actuators was used for the rudder.

The hydraulic power actuators from the basic F-8C airplane were used without modification. Electrohydraulic secondary actuators were installed to drive the metering valves of each of the five power actuators. The secondary actuators acted as three-chamber force summing devices when driven from the backup system. The primary system drove the secondary actuators through active monitor servo valves.

When the secondary actuators were driven through the primary system, they were stabilized through the active servo valve in the primary system electronics box. Analog 5-hertz low-pass filters were included in the primary system electronics. When driven through the backup servo valves, the secondary actuators were stabilized in the backup system electronics packages, one for each of the three backup channels.

Reference 1 describes the actuation systems in more detail.

Primary/Backup System Interface

A functional diagram of the fly-by-wire system is shown in figure 7. The Apollo computer received inputs from the pilot's stick together with aircraft motion information from the inertial measurement unit. Surface commands were computed according to the programed control laws.

The two drive signals for each surface represented commands to the secondary actuator position loop, which was closed with analog stabilization electronics outside the Apollo computer. As shown in figure 7, there was an active and a monitor servo path. If a failure occurred in either path, a hydraulic comparator would sense the differential pressure between the active and the monitor servo valve and transfer control to the backup control system. As long as the primary control system was operating normally, the backup control system would track the active channel by way of the synchronization network. Only the hydraulic pressure was bypassed at the secondary actuator, so that the backup system was ready to take over at any time. If a transfer to the backup system was requested, the bypass was removed and the synchronization network was disabled, resulting in immediate proportional control from the pilot's stick. In the backup mode, the active servo valve was blocked and the secondary actuator operated as a force summer for the three backup channels. The digital computer continued to operate, computing the control laws which gave the best estimate of what the backup system commanded. If a transfer to the primary control system was attempted, the transient was small as long as the computer was tracking the backup system. If the error was excessive between the primary control system and the backup control system, a cross-channel comparator prevented transfer to the primary control system.

Fault Detection

Although built-in fault detection was extremely important for both the primary and the backup systems, it was of particular importance in the primary system. Because the primary system was full authority as well as single channel, its responses could have been hazardous if failures were not handled properly. Therefore, it had to be established that no digital computer system hardware failure could cause a hardover or otherwise hazardous signal. Figure 8 shows the type of digital system failure detection used. The Apollo computer had an extensive and proved fault detection and reporting system which was built into the computer hardware (item 1 in the figure). This system, modified slightly for application to the F-8C airplane, was the most significant portion of the failure detection system. Some of the types of failures detected were:

Logic circuits –
- Parity failed
- Program entered loop and did not exit
- Program attempted to access unused read-only memory
- Program failed to check in occasionally

Analog circuits —
- Voltage went out of limits
- Oscillator failed
- Timing pulse generator failed

Each of the failures caused a restart, that is, a hardware-forced transfer out of the control law program to a software routine which performed several clearing and initialization steps in an attempt to correct the cause of the restart before allowing control law computations to continue. For some restart conditions, a signal was issued which caused a transfer to the backup control system.

The Apollo computer also monitored the performance of the inertial measurement unit (item 2, fig.8). Written into the software were decisions either to transfer the system to the backup control system for serious failures or to select the direct mode in the primary system for situations such as an inertial measurement unit accelerometer failure, which would affect only certain augmented modes.

Analysis of primary system failures showed the need for additional hardware failure detection circuitry (item 3, fig 8). The failure of certain channel outbits not monitored by the Apollo computer, in combination with normal pilot reactions, could have led to hazardous situations. These conditions first became apparent in piloted, closed-loop simulations using the iron bird simulator. The necessary hardware modifications were made and implemented in the system to circumvent these failure conditions or to cause a transfer to the backup control system when prevention was not possible.

Built-in test equipment for the backup system and primary electronics was provided in the pilot's side console. This self-test equipment could be activated only during preflight tests (ref. 1).

FLIGHT CONTROL SOFTWARE

Software flexibility made it possible to investigate a multimode F-8 digital flight control system using hardware that was designed for an entirely different purpose — guidance and navigation in space. The structure of the primary system control laws, which were implemented through software, and the associated logic functions are described.

Control Law Modes

Control in each axis was provided in the control laws. The simplest form in each axis was the direct mode, illustrated in figure 9. The control law structure and gain settings were selected to be as close as possible to those of the backup control system. The first level of augmentation was rate feedback in the pitch and roll axes. Figure 10 illustrates these SAS modes as they were during flight tests. In the yaw axis, an aileron-to-rudder interconnect was included in addition to the yaw rate feedback (fig. 11). The most advanced type of control law was a blended pitch rate and normal acceleration command augmentation mode, illustrated in figure 12.

The sampled-data aspect of the digital fly-by-wire system significantly affected the implementation of the software. The pitch SAS mode, illustrated in figure 13, was representative of the other augmented modes. A multirate sampling system was used, with a major cycle sample time, T, of 30 milliseconds and a minor cycle sample time, 3T, of 90 milliseconds. Gimbal angles were transformed to body angles by using sines and cosines of gimbal angles updated once every minor cycle. A second-order rate estimator operating on these gimbal angles provided body rates, which were then filtered. General-purpose digital filters were programed so that different characteristics could be selected independently for each feedback parameter. Proper selection of the difference equation coefficients could provide a wide range of filter characteristics. A first-order prefilter, a dead band, and a parabolic nonlinear gradient were available to shape the pilot's input.

Logic Functions

An important capability made possible through software was the integration of logic statements in the control law code. Logic statements, even though complex, were easily written into the software. Had the system been analog rather than digital, special-purpose hardware would have been necessary to perform the same logic functions. One mode logic function associated with the yaw axis is discussed in reference 5.

Another type of logic function was the software reasonability test which was applied to each surface command before it was sent to the digital-to-analog converter. If the new command differed from the previous command by more than a predetermined amount, the affected axis would have transferred to the direct mode. This down mode philosophy was based on the assumption that a reasonability limit would be exceeded because of generic failures in the augmentation control laws rather than because of a hardware failure which would have affected the direct mode as well. It was assumed that a hardware failure would have been detected by the built-in Apollo computer fault detection logic.

Trim inputs were also tested for reasonability before the trim value was updated. If a combination of primary trim commands was sensed that corresponded to an impossible situation for an unfailed system, a failure was assumed, the primary trim was deactivated, and an auxiliary trim system was activated. A test for runaway trim was included which disabled trim updates if the trim command persisted for more than 3 seconds.

DIGITAL FLY-BY-WIRE DESIGN

Design Ground Rules

Several ground rules were established in order to meet the objectives of the program. First, the airplane was to fly from the first flight without mechanical reversion capability. This forced the designers to take the care necessary to establish as much confidence in the system, including the software, as would be required for future active control applications. Second, the primary digital system was to utilize the hardware from the Apollo guidance and navigation system. The system was to remain intact; only software changes were allowed. The primary reason for this requirement was to retain the system's high degree of built-in integrity and reliability. A third basic design ground rule established early in the program was to make the pilot's interface with the computer as simple as possible. As a result, the pilot was given control over flight control functions rather than a direct communication with the computer. All functional changes (for example, a mode change) desired by the pilot were to be made through single switch actions. This allowed the pilot to perform functional changes rapidly and eliminated the possibility of incorrect entry or improper addressing which could have had dire consequences close to the ground or at high dynamic pressure. The last ground rule was to provide handling qualities that would be judged satisfactory by the pilots. A criterion based on C^* response to a step pitch stick command (ref. 6) was used as a guide during the design of the longitudinal control system. Military Specification MIL-F-8785B, level 1, was used during the lateral-directional control system design.

These ground rules had further implications. For example, the interface equipment associated with the Apollo hardware established limits on the flight envelope for satisfactory operation. The analog-to-digital converter used for pilot stick inputs had only 45 usable discrete levels between zero and full stick. In the pitch axis each discrete level resulted in a specific level of aircraft normal acceleration, depending on stick gearing and dynamic pressure. In this instance, the acceleration increment became objectionable to the pilot within the basic flight envelope. Thus a new flight envelope limit was established at the dynamic pressure at which the stick quantization effect was not objectionable. Because of the design characteristics of the inertial measurement unit, some additional restrictions were placed on the airplane's maneuverability. These included a roll angular rate limit of 70 deg/sec and a pitch attitude limit of 70°.

Design Synthesis and Analysis

The closed-loop primary system was synthesized and analyzed by using two methods. The first was an analog sample and hold simulation which was useful in the learning process in that it pointed out the more general aspects of the digital control problem. For example, the acceptable range of sample rate, 25 to 50 samples per second, was defined. The effect of the folding phenomenon of sampled-data systems on the structural mode frequencies and the influence of common nonlinearities were also studied in this design phase. The second method used a digital synthesis program which provided linear analysis as a cross check and a background for the sample and hold simulation. Basic control laws, compensation, and logic were established by using these two methods. A specification for the control law software was then formulated, thus providing the basis for coding the flight software.

Design Verification and Refinement

With the delivery of portions of the flight hardware, including the Apollo computer and the coupling data unit, and early releases of flight software, design verification and refinement was started. A six-degree-of-freedom digital aerodynamic model of the F-8C airplane was used in conjunction with the flight hardware to form a partial hardware hybrid simulation.

The first two analysis methods did not consider any pilot interface, but the partial hardware hybrid simulation included a lunar module hand controller with which the F-8C model could be crudely flown. Coarse input quantization, a problem of importance later, did not become evident in this simulation because it was completely masked by the characteristics of the hand controller. This is one of the major disadvantages of any simulation which does not include major hardware elements.

Another important tool in the design verification and refinement was the batch process all-digital simulation. This simulation included the software being verified, an Apollo computer emulator, and a program representing the F-8C aerodynamics, all run on a large host computer. Powerful plotting routines made the internal computer parameters visible during each run. All the control system parameters were examined for reasonableness, particularly as they responded to mode and gain changes. One of the most useful plots was duty cycle versus run time. A typical variation of duty cycle during a maneuver is shown in figure 14 for a roll step. Because of some additional code for computation of stick nonlinearities when the stick was displaced from zero, additional computational time was required during this maneuver. This was reflected in an increase in the duty cycle, as shown in the figure. Other contributions to duty cycle were the interrupts from the motion sensors. The increase in roll rate produced a loss in available computation time roughly proportional to the roll rate. This loss of computation time effectively increased the duty cycle.

The last step in the design verification used the iron bird simulator. One problem — the coarse quantization of the pilot's stick inputs — was uncovered immediately. The problem became obvious once the hand controller was replaced with the actual center stick. The staircase shape of the computer output commands produced sharp responses at the secondary actuators which were unacceptable from the standpoint of mechanical motion and structural element excitation. Low-pass filtering of the computer output was undesirable because of its adverse effect on closed-loop performance. This suggested the use of a digital pilot prefilter that had not been anticipated in the control law specifications. The flight software had already been substantially verified, but fortunately the read-only memory had not yet been manufactured. The prefilter was quickly programed in software and the code was reverified. Consequently, there was essentially no effect on the overall schedule. This points out one of the significant advantages of a digital flight control system: Necessary changes can be made late in the design without affecting hardware procurement, packaging, or requalification. Although additional software verification will be required, it will not have the adverse effect on program schedules that is typical of a hardware redesign of an analog system.

Looking back on the various design and analysis tools, it is apparent that they complemented one another. Confidence in the system grew each time an independent simulation or analysis gave results comparable to those obtained previously. The importance of having the pilot in the loop with as much actual hardware as possible was demonstrated vividly. In terms of time spent on verifying the various aspects of the design, the largest proportion was devoted to systematically verifying each logic function and mode transfer and the effects of failures. Another time-consuming aspect was the refinement of stick gradients and nonlinearities near zero stick. A much smaller proportion of time was spent on closed-loop augmentation characteristics, probably because of the good agreement generally found between the results from sampled-data analysis methods and simulation results.

RELATIONSHIP TO FUTURE APPLICATIONS

The configurations of future fly-by-wire systems will probably be strongly dependent on the specific missions for which they are designed. As such, each system will be unique in some respects, but will have a large degree of commonality with other fly-by-wire systems. The F-8 fly-by-wire system was unique in that it consisted of a simplex digital primary system, a triplex analog backup system, and no mechanical reversion capability. However, in this unique system were several features that will be relevant to the systems that will be required to achieve the advantages that active control offers. These features were, basically, dissimilar redundancy, single string software, and the experience associated with the digital system design.

Dissimilar Redundancy

The F-8 fly-by-wire system experience with two dissimilar systems provides information applicable to future systems which are likely to have dissimilar redundancy. Most of the problems were concerned with the synchronization of the two systems. Transfers from one system to another were handled differently, but the goal was to minimize transients caused by the transfer. In each instance, the system in control was tracked by the other system so that transients would be minimized. However, the primary system tracked the backup system by estimating the surface command of the backup system based on the pilot's control commands and trim inputs only. In transfers from the primary system to the backup system, the backup system tracked the output of the primary system. Although this eliminated the need to reconstruct the primary system signal propagation in the backup system, it did open the possibility for unusual initialization conditions when the transfer occurred during an abrupt maneuver. Another factor was that a transfer from the primary system to the backup system could have been initiated automatically as a result of a failure, thus the failure analysis had to consider all possible failures that could have resulted in a transfer. The timing of this transfer was critical in some instances when it could have coupled with the pilot's normal response to cause unacceptable conditions.

Some aspects of the dissimilar redundant system gave insight into redundancy management problems which may be expected in the future. The backup system mechanized the trim function using a digital integrator to reduce drift. Because of differences in the sampling mechanisms between the two systems, large errors between the two trim signals were noted after extended flight with the backup system in control in which numerous trim inputs were made. Figure 15 illustrates the two sampling mechanisms. The primary system sampled trim commands every 90 milliseconds. If trim was being commanded at the time of the sample, the trim value was updated in the software. The backup control system did not update its digital trim integrator until a capacitor was charged up to a prescribed threshold. Although the capacitor began charging the instant the trim button was pushed, approximately 175 milliseconds were required before the first update of trim. As a result of these two sampling mechanisms, trim inputs of less than 175 milliseconds, but greater than 90 milliseconds, caused the primary system, but not the backup system, to update trim. To correct the problem, there would have had to be either some exchange of actual trim value information between the two systems or some form of verification that one system received the trim command before the other system updated the trim value. Each of these possible solutions would have required additional connections between the two systems, which would have been undesirable because they would have created new failure possibilities. For this particular research application, a procedural change in conjunction with close monitoring of telemetered data in the control room made modification of the system unnecessary.

Single String Software

Because a simplex digital system can have only a single program in control at one time, it can be described as a system with single string software. However, redundant digital systems with the same program in each computer also effectively have single string software. The experiences with the F-8 digital system software are closely related, then, to the multichannel digital systems expected in future civil transports. Generic software failures would have equivalent effects on any system with single string software, regardless of the system's redundancy. The software controls described in reference 2 suggest that careful verification will always be necessary, but that the confidence necessary for man-rating the software can be established.

Another factor that emphasized the importance of man-rated software was that the single string software had full-authority control over the control surfaces; thus it was obviously flight critical. Digital systems will be called on to perform more and more flight-critical functions and, on the basis of our experience, can be depended on to perform with high integrity.

Removal of all mechanical reversion capability before the first flight had a significant effect on the entire design and verification process. It forced an approach that would establish complete confidence in the system on the basis of simulation alone. If the alternate approach had been taken, that of retaining a mechanical link, the most probable flight-test procedure would have been to fly to a safe altitude using the mechanical system and then engage the fly-by-wire system. After confidence was gained at altitude, the more critical flight safety functions, such as takeoff and landing, would have been encompassed gradually.

Design Experience

On the basis of the F-8 digital fly-by-wire design experience, several recommendations can be made regarding the design of digital control systems for future civil transports. Many of these recommendations correspond simply to good design practice. Analyzing closed-loop performance using standard sampled-data analysis techniques such as z-plane root locus can be relied on to give good agreement with more complete simulations. Several forms of simulation and analysis should be used to build confidence in the system before the first flight. A simulation that includes as much actual hardware as possible is important in correctly assessing system performance. The interface with the pilot is particularly important.

CONCLUDING REMARKS

The feasibility of a digital fly-by-wire system was assessed by replacing the mechanical flight control system of an F-8C airplane with a digital primary and an analog backup fly-by-wire system. The design and verification procedures which will be necessary if flight-critical active control is to be used in future aircraft were established and successfully applied as part of the flight program. Careful application of standard sampled-data design methods and systematic verification of control system hardware and software using complete simulations resulted in a digital fly-by-wire system with extremely high integrity. The successful use of single string software in a full-authority flight control system demonstrated the high level of confidence which can be placed in digital flight control.

The experience with the F-8 digital fly-by-wire system pointed up several factors that will be important in the successful design of future full-time, flight-critical digital control systems:

(1) Batch process all-digital simulation was extremely helpful in tracing internal computer variables and in providing visibility to system response during mode changes.

(2) A complete piloted simulation with actual flight control system hardware provided important results relative to the pilot/stick interface that had not been obtained in earlier simulations which did not include the actual control stick.

(3) The largest portion of the design and verification effort was devoted to logic functions, such as mode transfers, and the effects of failures.

(4) Software changes made late in the design to correct hardware-related problems had a negligible effect on the program schedule.

A major aspect of the F-8 digital fly-by-wire system which will have application to future systems was its dissimilar redundancy. Failure isolation between the primary and the backup systems was achieved as desired, although some problems were encountered with intersystem synchronization.

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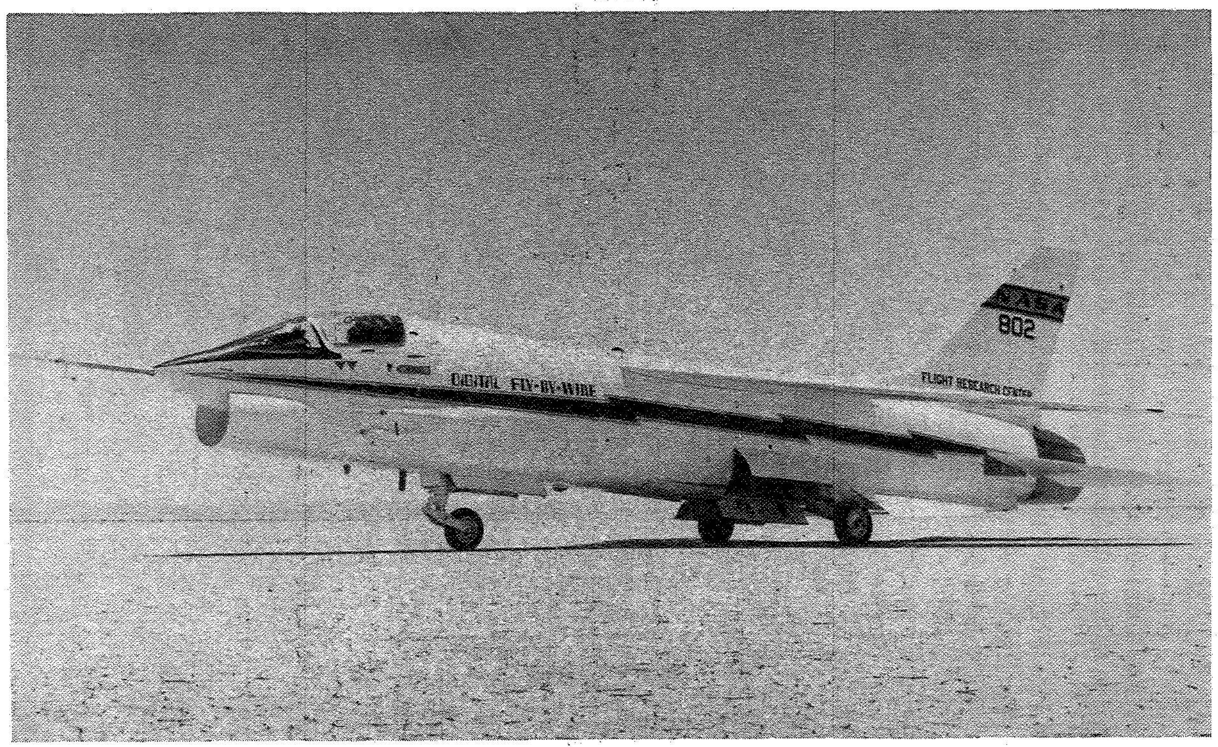

Figure 1. F-8 digital fly-by-wire airplane.

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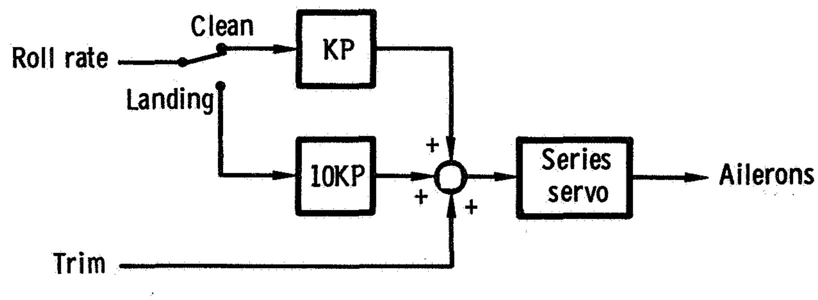
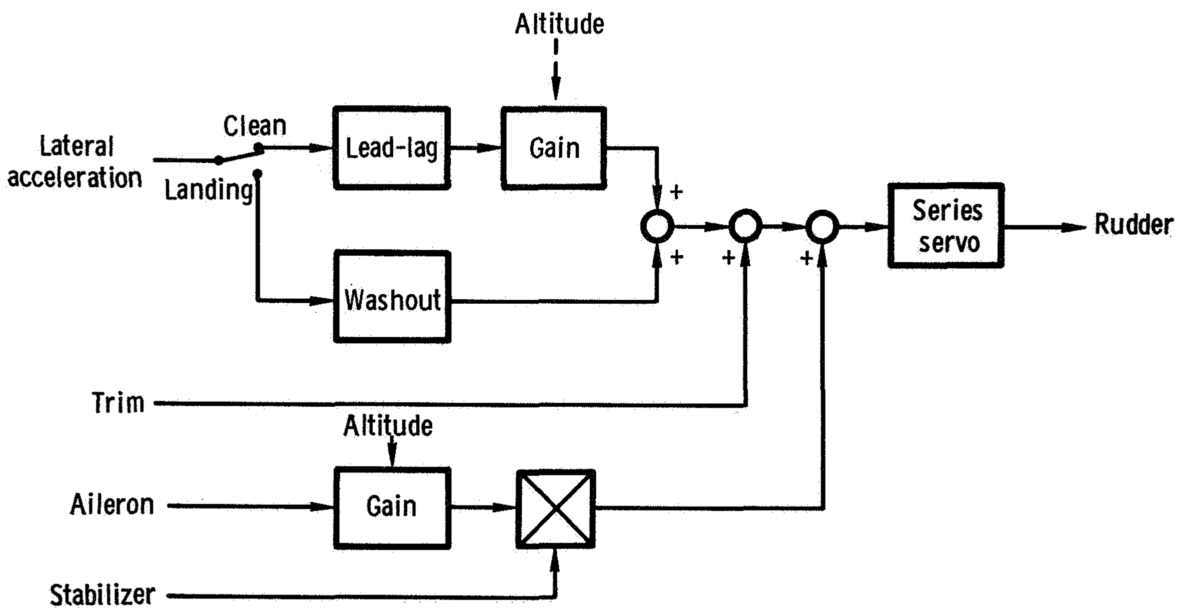


(a) Roll-axis damper.



(b) Yaw-axis damper.

Figure 2. Standard F-8C roll and yaw stability augmentation systems.

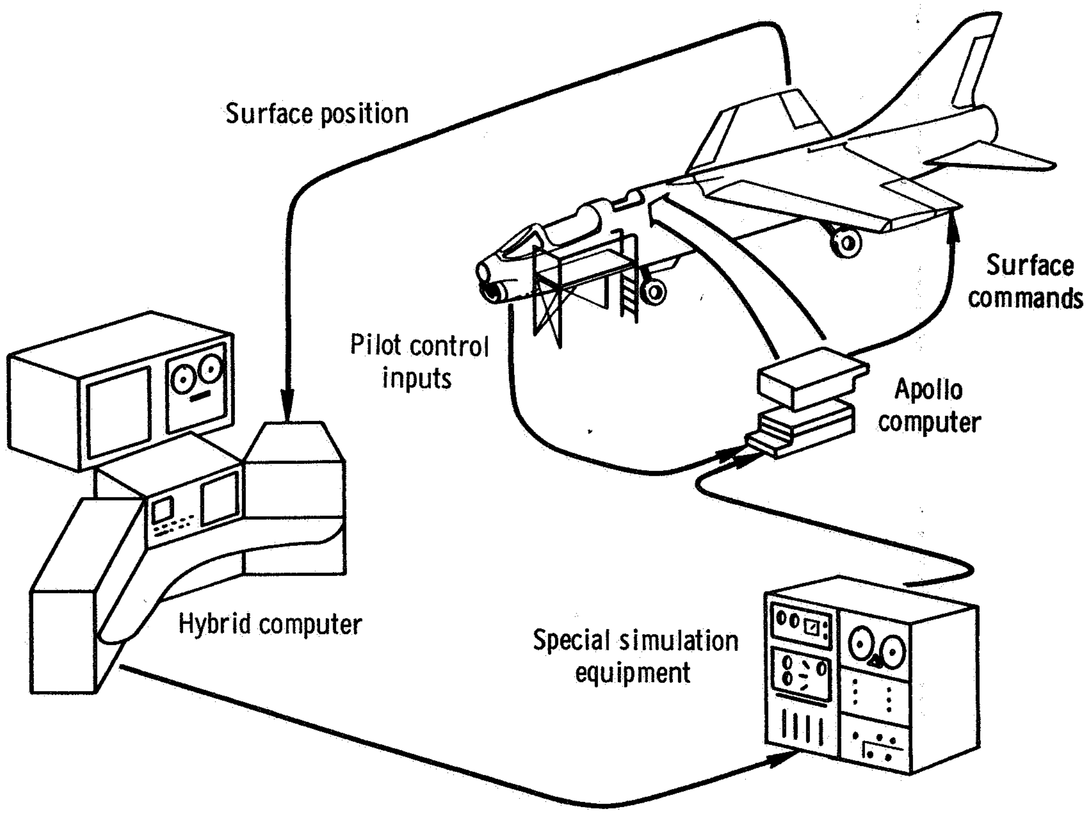


Figure 3. F-8C iron bird simulator.

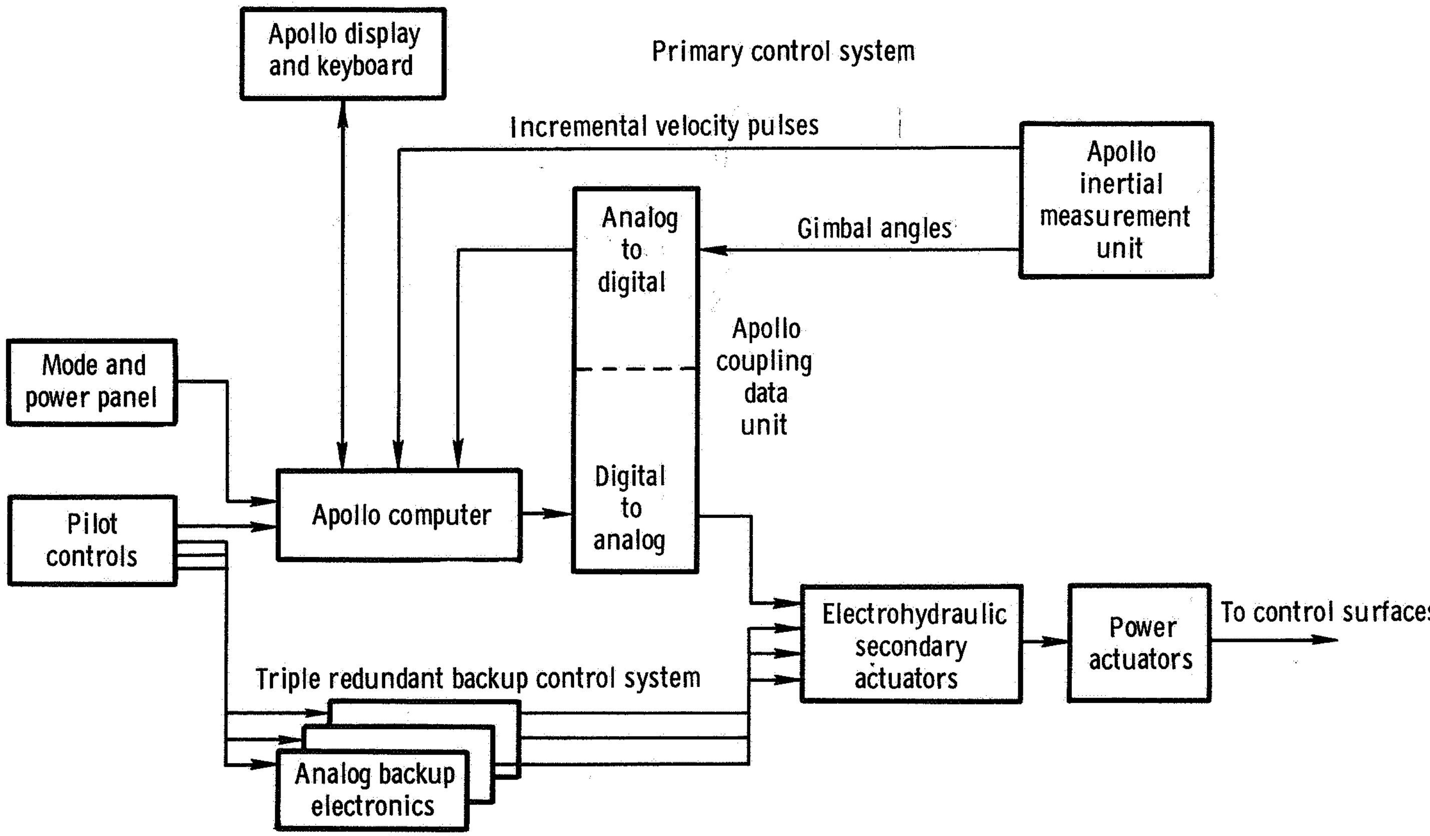


Figure 4. F-8 digital fly-by-wire control system components.

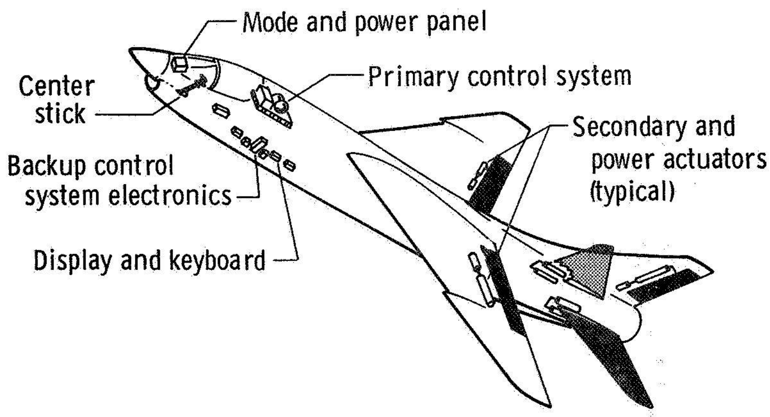


Figure 5. Location of fly-by-wire control system in F-8C airplane.

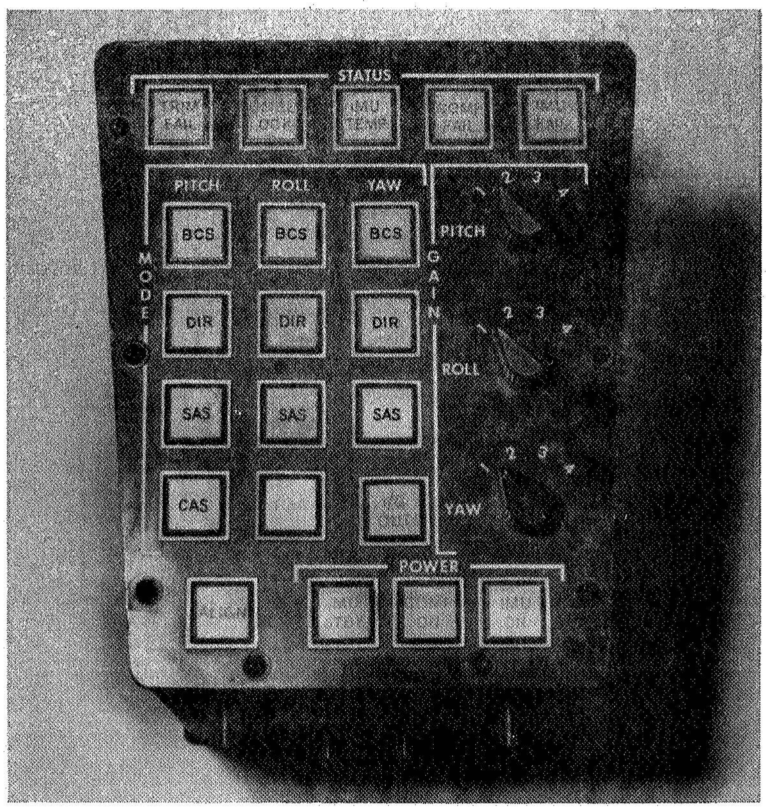


Figure 6. Mode and power panel. E-23983

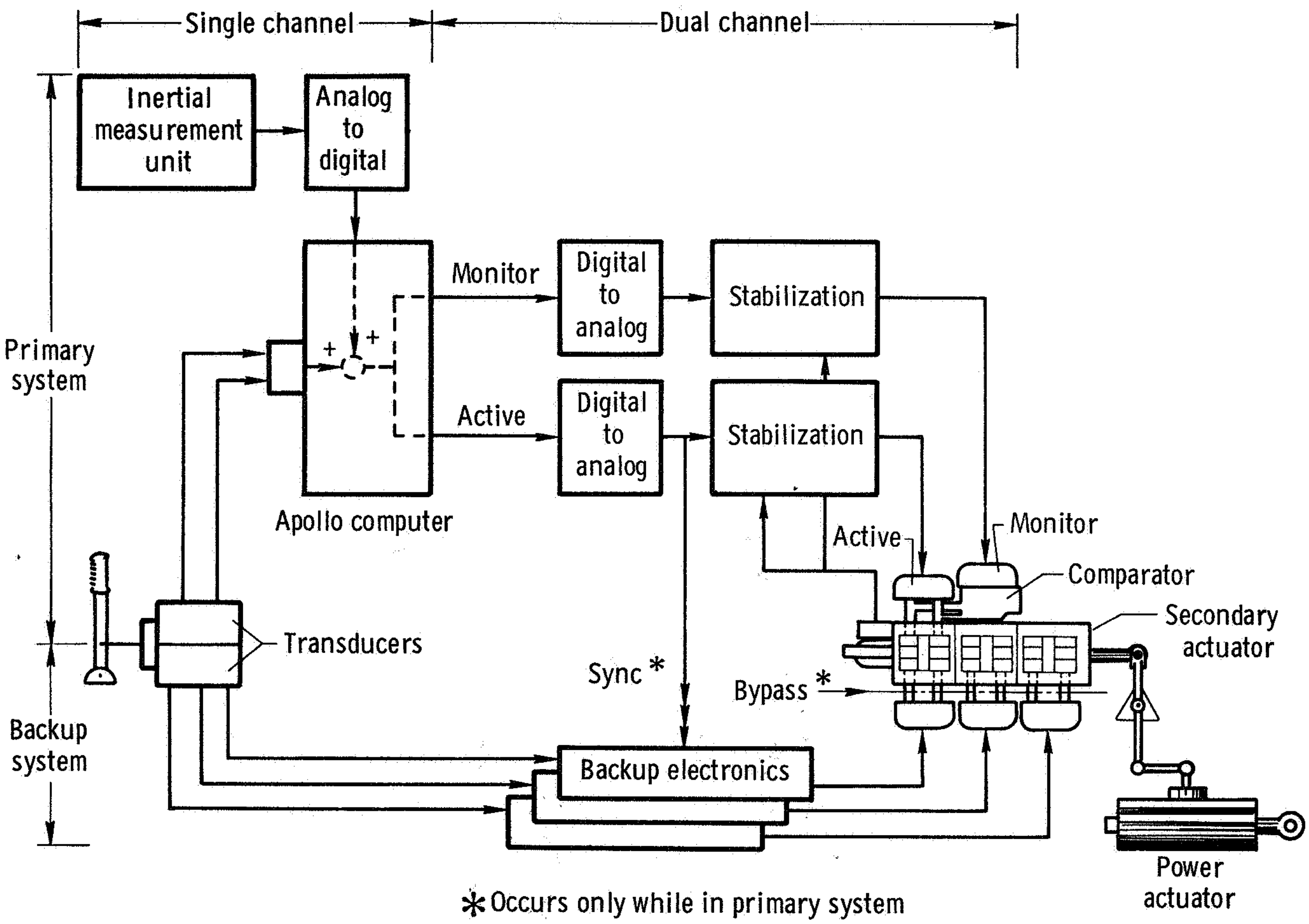


Figure 7. F-8 digital fly-by-wire system mechanization.

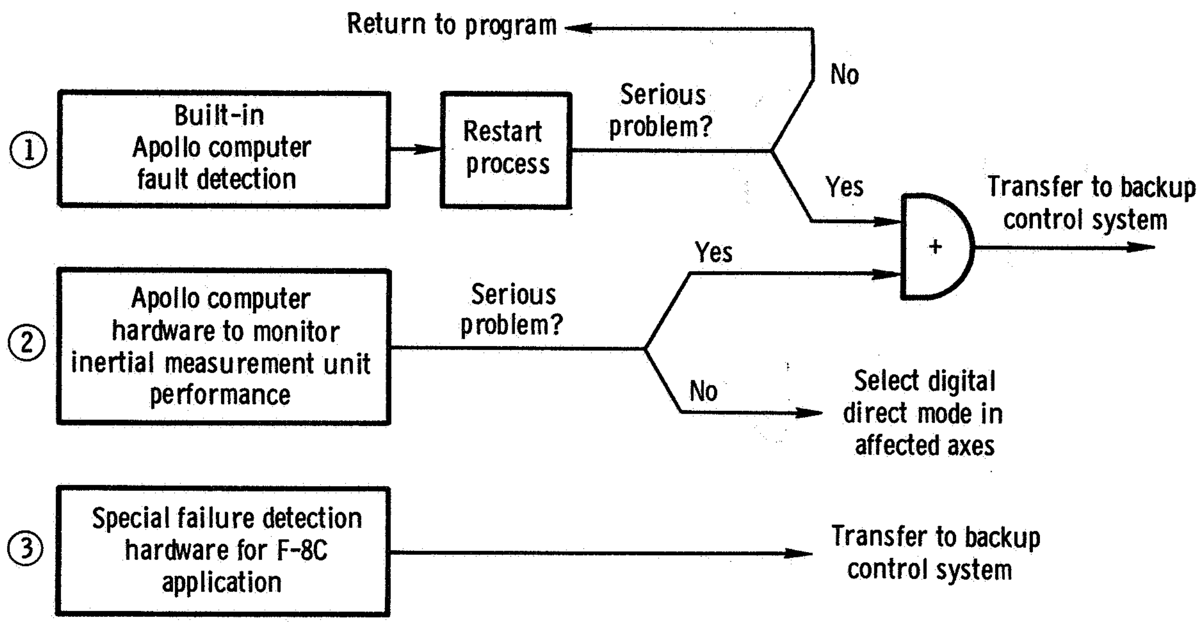


Figure 8. Digital system failure detection and reporting system.

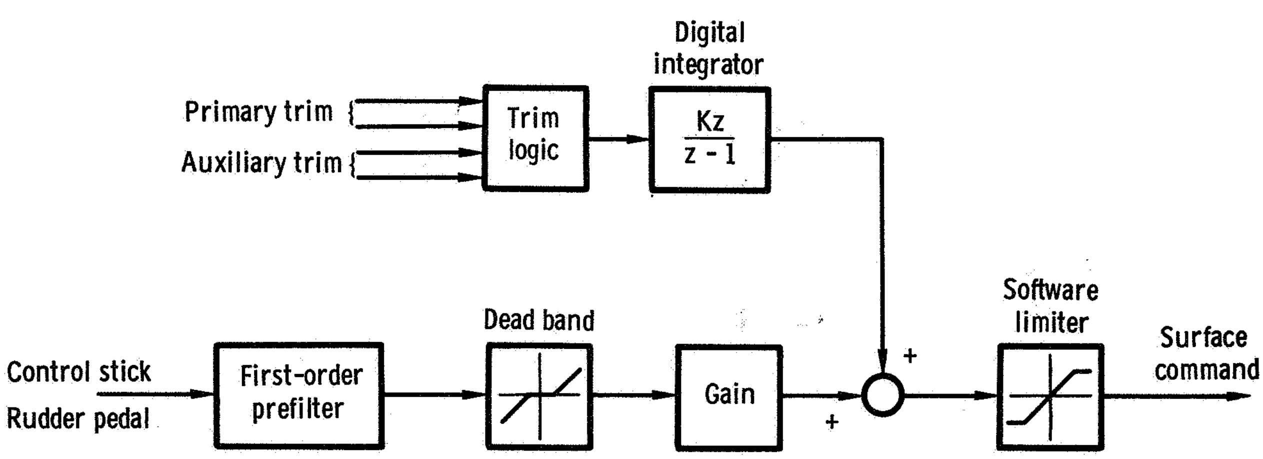


Figure 9. Direct modes for pitch, roll, and yaw axes.

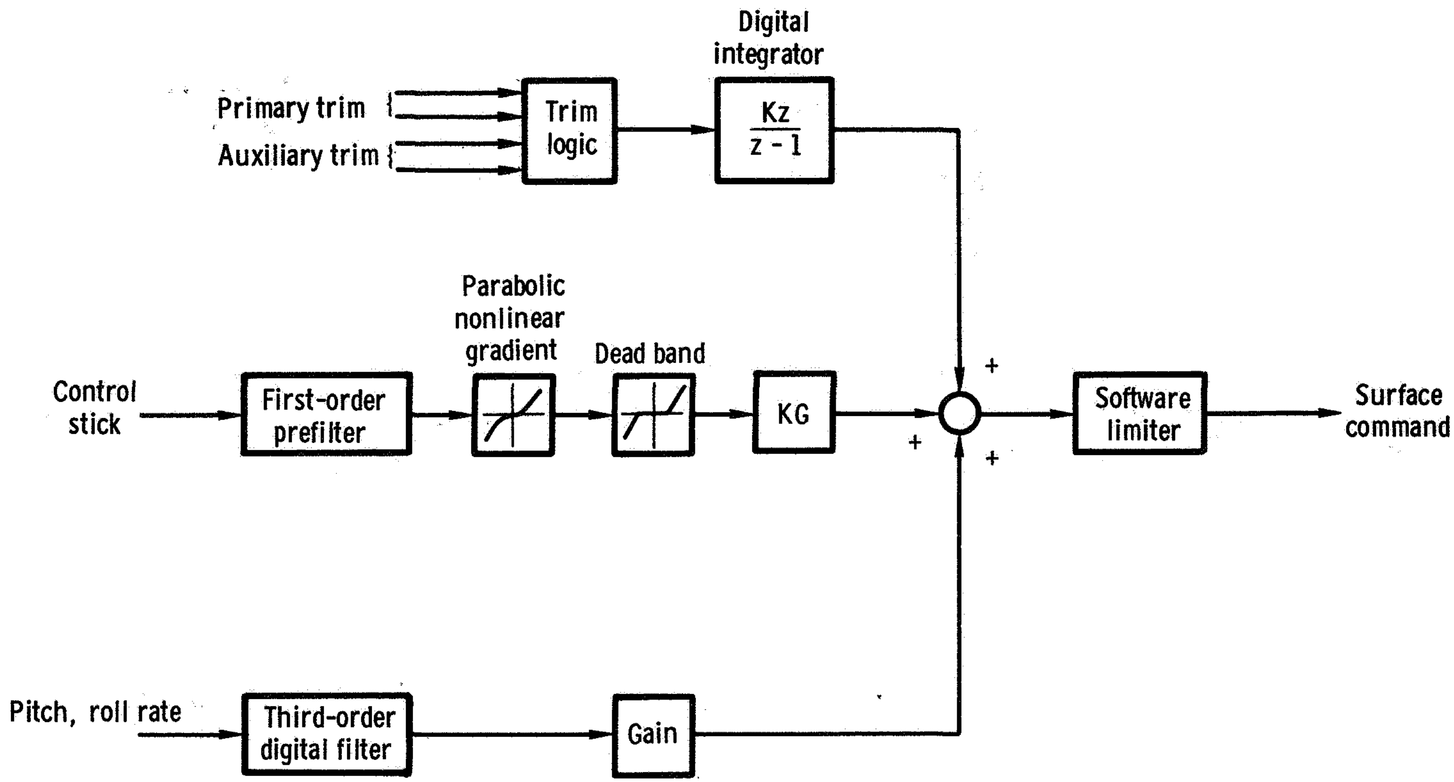


Figure 10. Pitch and roll stability augmentation systems.

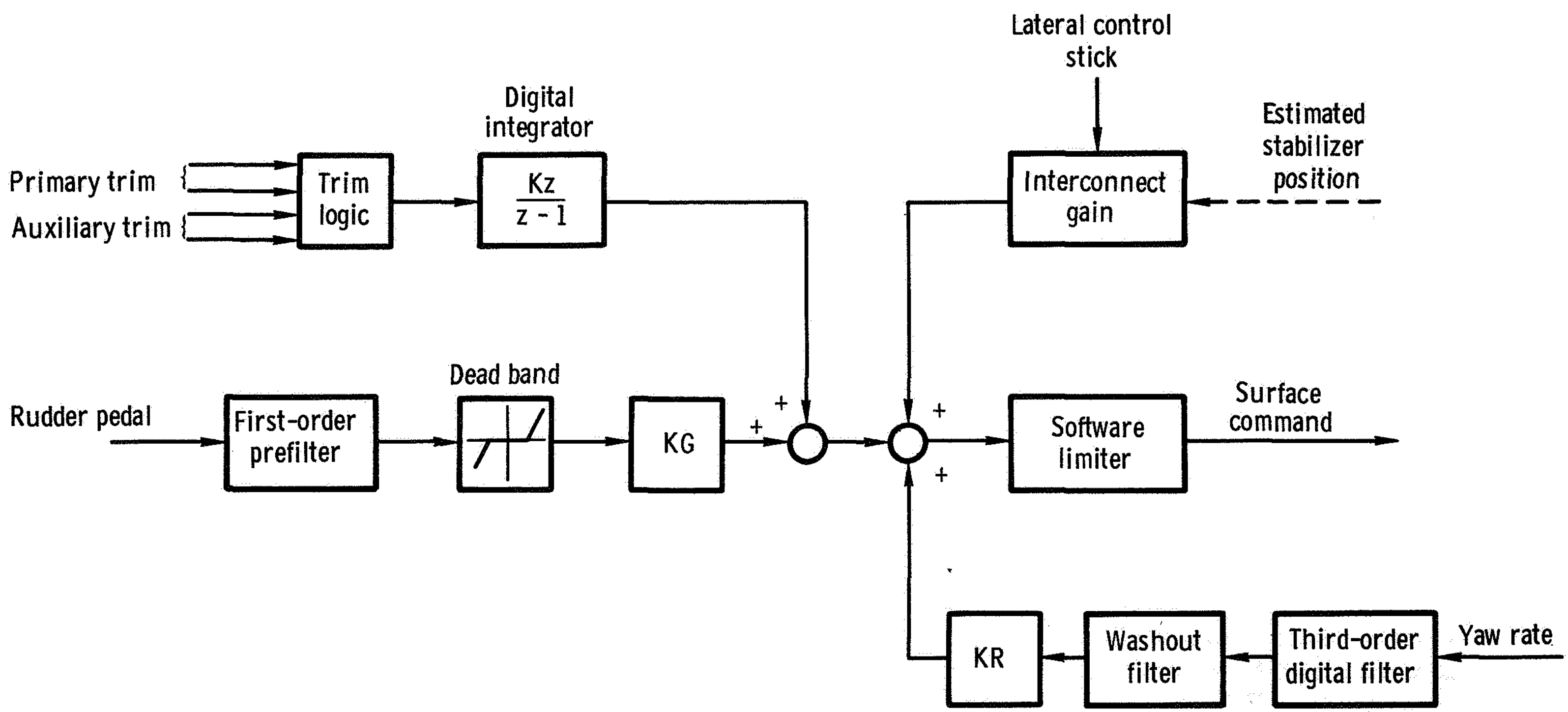


Figure 11. Yaw stability augmentation system.

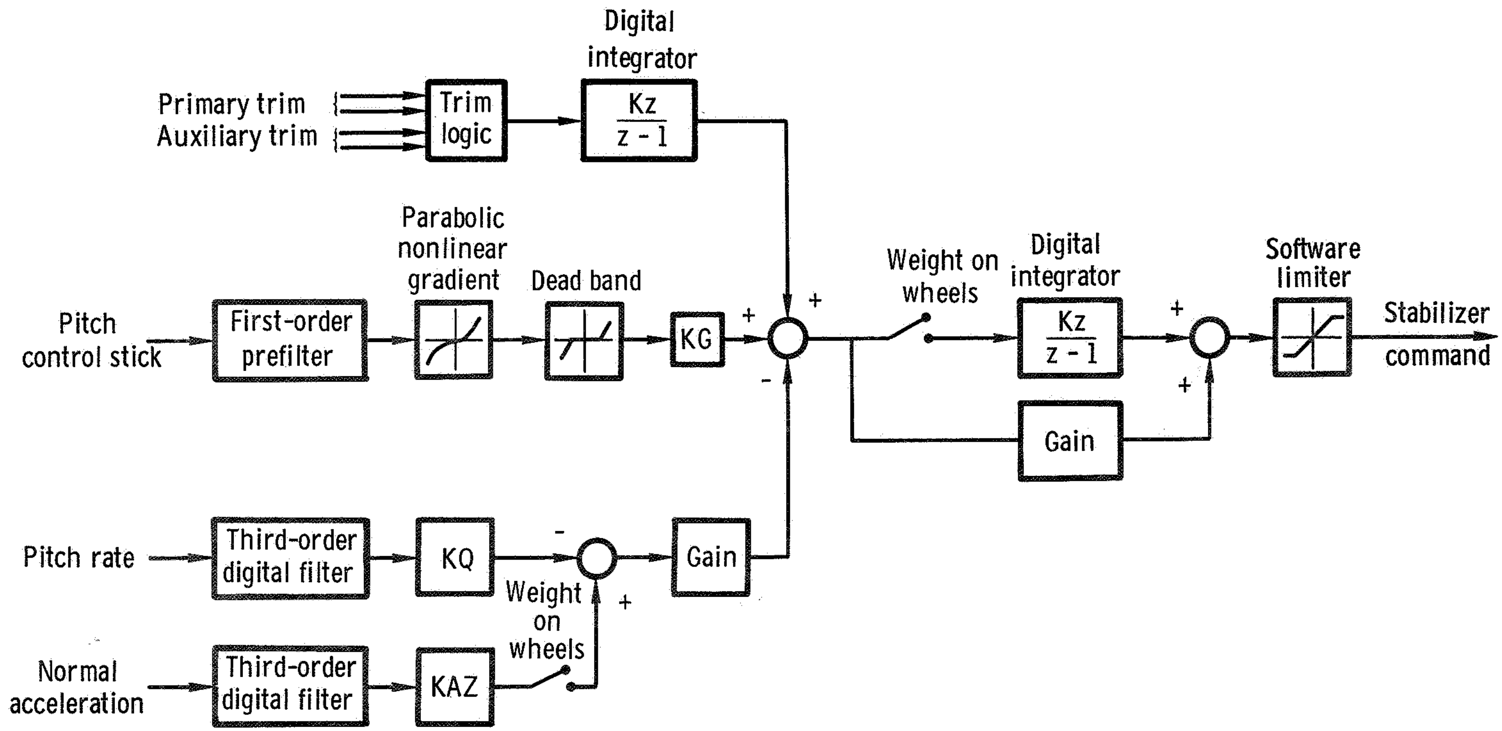


Figure 12. Pitch command augmentation system.

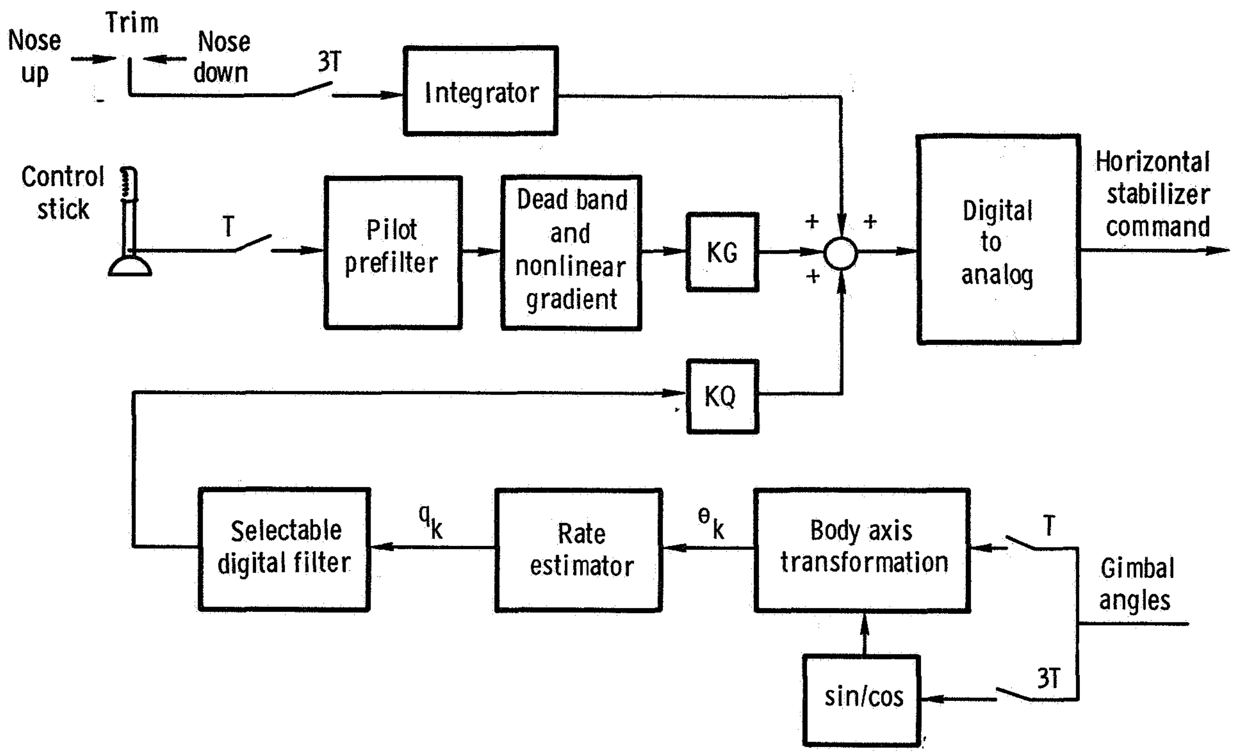


Figure 13. Pitch stability augmentation system mode as a sampled-data system.

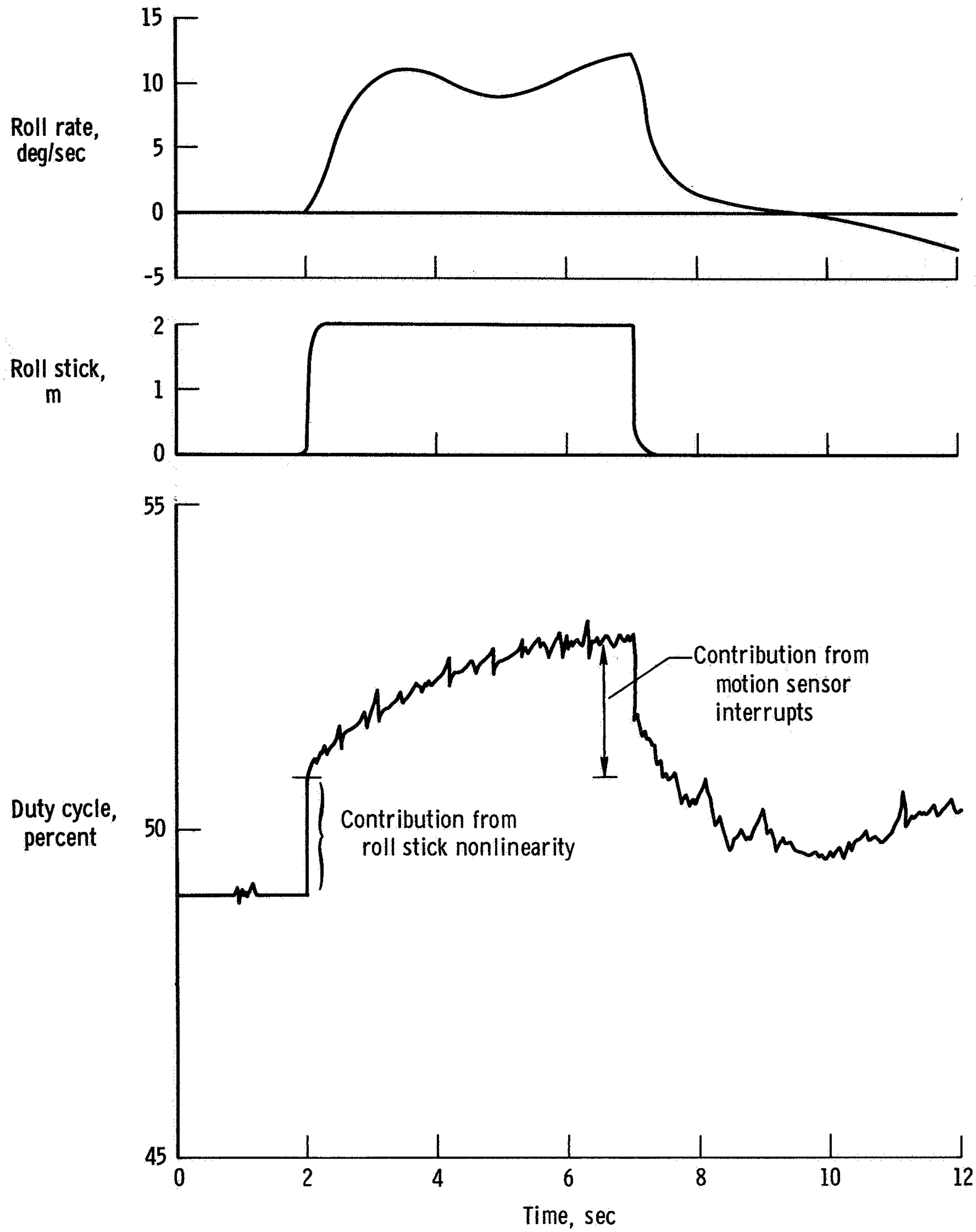


Figure 14. Duty cycle variation during roll step maneuver.

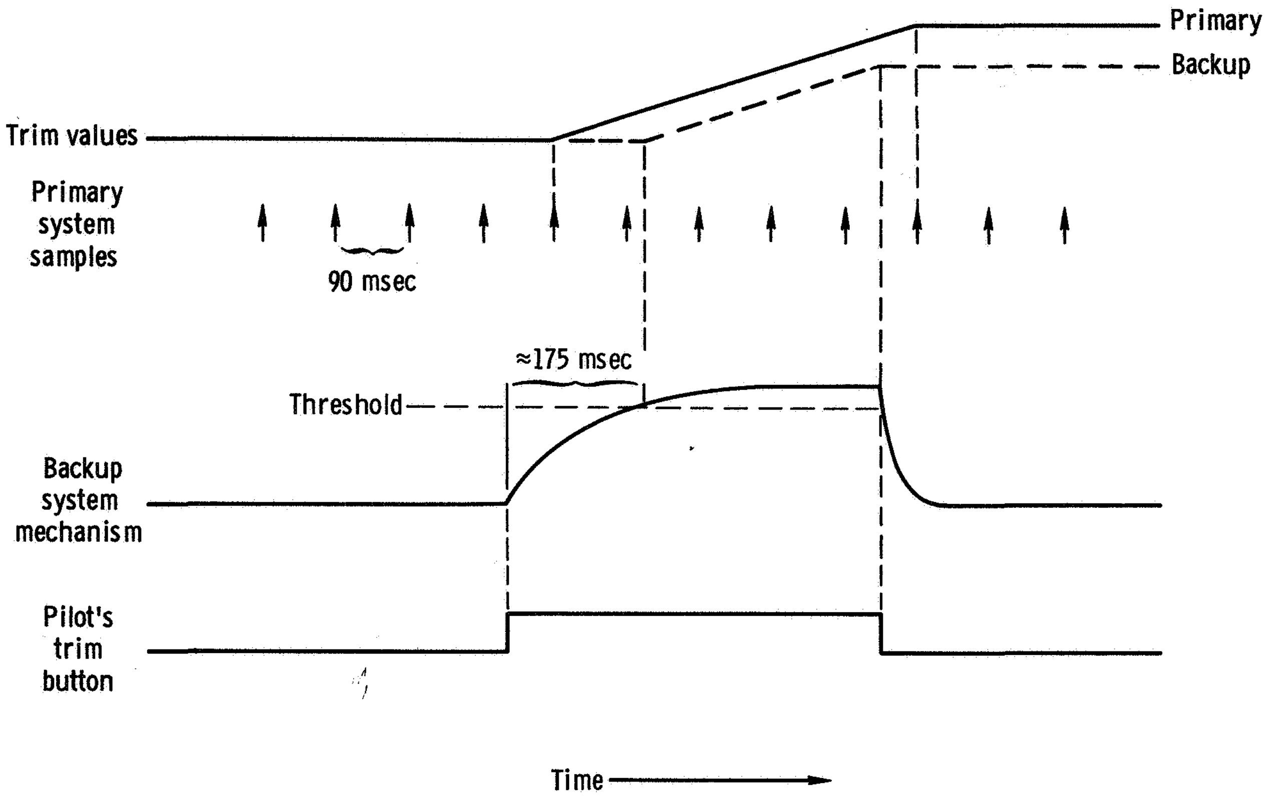


Figure 15. Trim sampling mechanisms of the primary and backup systems.