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JULY-AUG. 1966 J. AIRCRAFT VOL. 3, NO. 4

Evolution of the Honeywell First-Generation Adaptive Autopilot and Its Applications to F-94, F-101, X-15, and X-20 Vehicles

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A review of the considerations that influenced the basic analytical configuration of the Honeywell adaptive autopilot is presented. The control concept uses a wide bandwidth approach with the control gain varied in accordance with information derived within the control system. Desirable performance characteristics have been obtained from the inherent ability of a high-gain feedback control system to accommodate extreme variations in aerodynamic parameters. The gain-changing technique in its early form made use of a bi-stable element. Refinements to the gain-changing technique resulted from practical considerations, such as the influence of pilot inputs, effects of gust disturbances, and the unique performance requirements of various vehicles. The functional requirements, system description, and achieved performance are presented for the F-94, F-101, X-15, and X-20 applications.

Introduction

Although the concept of self-adaptive control systems is not new, most of the serious development effort to design such systems has been expended in the past 10 years. The growing complexity of aircraft control systems, the development of more advanced aircraft, and the advent of winged aerospace vehicles such as the X-15 and X-20 provide the stimulus to spur the effort on. A significant portion of this work was sponsored by the U. S. Air Force Flight Dynamics Control Laboratory of the Research and Technology Division (formerly WADC). At least eight different conceptual schemes have been studied at length by various agencies and investigators. These include systems based upon high-gain, model-following techniques, model reference with error-minimizing schemes, and perturbation response for signal input evaluation. Many of these systems have been developed to the point of proving feasibility in flight tests.

Honeywell has pursued development of the high-gain, model-following concept to a greater extent than any of the other approaches. This basic concept has been developed and tested in a number of vehicles. These include the F-94C, F-101A, X-15, and extensive X-20 hardware simulator studies.‡ This paper describes the functional features, system mechanization, and performance achieved in each case.

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‡ Application has also been made to the control of highly elastic boosters, including design and fabrication of a flightworthy adaptive controller for test in a Scout vehicle.[1]

Basic Concepts

The Honeywell adaptive autopilot inner loop fundamentally consists of a tight feedback system controlling a selected aircraft variable. This controlled variable is usually selected to facilitate the use of control-stick steering when the autopilot is used as a damper and to facilitate the use of outer loops such as attitude and altitude hold. For this discussion, pitch rate will be considered as the controlled variable, as shown in Fig. 1a.

Comparison of the system in Fig. 1a with conventional pitch rate systems shows few basic differences. There is an important distinction, however, in the response specification. The response of the inner-loop output $\dot{\theta}$ to the input $\dot{\theta}_m$ should be at least three times faster than the desired system response to pilot input commands. Fast response of the inner loop is achieved by using series lead compensation along with high forward-loop gains. Uniformity of the response to input commands is achieved by using an input-shaping model.

Command Response

The desired response to pilot commands for the Honeywell adaptive system is established by a model that shapes all commands to the inner loop. The form of the model and its characteristics are selected on the basis of pilot preference and outer-loop response criteria.

After the model is selected the response requirement for the inner loop may be established. It has been determined experimentally that if the bandwidth of the inner loop exceeds the model bandwidth by a factor of three or more, the over-all response of $\dot{\theta}/\dot{\theta}_c$ essentially will be that of the model. It is evident that once this requirement is achieved, closure of an outer loop such as attitude hold becomes quite convenient. The outer-loop controlled variable need only be related to the inner-loop controlled variable by a relationship that does not depend on flight condition. Thus an attitude outer loop would dictate an attitude-rate inner loop, and an altitude outer loop would dictate a normal-acceleration inner loop.

Relay Controller

The adaptive controller may consist of a bi-stable element such as the relay shown in Fig. 1b. It is evident that this system will display a continuous limit cycle (except for unusually large transient errors) at the frequency where the loop phase lag is 180°. This limit cycle inherently establishes an effective relay gain at the critical value for each flight condition, thereby assuring maximum control tightness for a limited range of system errors. If the relay were to be replaced by a critical value of linear gain, the linear system would provide identical operation except for an oscillation or ringing occurring at the limit cycle frequency. Acceptable linear system operation would require a gain reduction below critical of sufficient magnitude to reduce these oscillations to an acceptable level.

The acceptability of the simple relay system depends largely upon the attainable limit-cycle frequency. For a given relay output amplitude, the amplitude of the limit cycle measured in terms of either pitch rate or elevator position varies inversely with limit-cycle frequency. Thus, by achieving a sufficiently high limit-cycle frequency, obtectionable limit-cycle amplitudes may be avoided. Since the limit-cycle frequency is also an indication of system bandwidth, required response rates must also be considered.

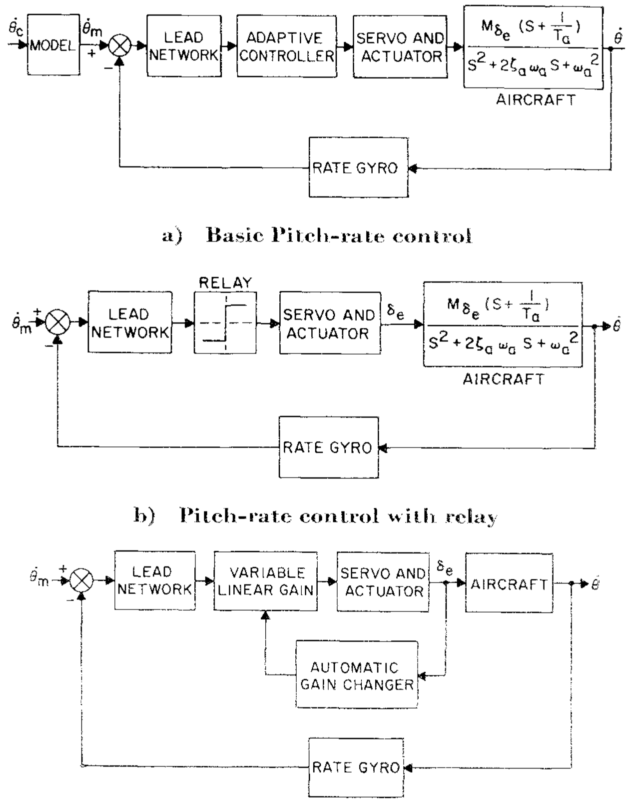
The ability of the system to ignore aircraft dynamics requires the closed-loop system bandwidth to be higher than the highest airframe natural frequency. Then the phase lag contribution of the aircraft at the limit-cycle frequency will be constant for all practical purposes. The effective relay gain will be an inverse function of surface effectiveness since, at these frequencies, the aircraft transfer function reduces to M_{δ_e}/S.

The required relay authority at a given flight condition will depend upon the maximum magnitudes of input commands and gust disturbances. A given input will require the largest relay authority at the lowest surface effectiveness condition. A limited amount of relay saturation (cessation of limit cycle) is permissible during a transient, but excessive saturation will result in an unacceptable loss in damping or decrease in bandwidth. Operation with continuous chatter throughout the transient will be the normal operation for the majority of the transients. Thus, system damping and command bandwidth generally can be analyzed on a linear basis by replacing the relay with a linear critical gain.

Inclusion in the system of practical component dynamics, especially those characteristic of large surface-actuators, will often prevent attainment of sufficiently high limit-cycle frequencies and acceptable limit-cycle amplitudes. Under these conditions, the simple relay will no longer be adequate. This system has been flight tested in an F-94C. A more detailed description is given in the discussion on applications.

Automatic Gain Changer

The basic adaptive inner loop with an automatic gain changer is shown in Fig. 1c. This system keeps the linear forward-loop gain at its critical value by maintaining a fixed amplitude of control-surface limit cycle. In this respect, operation is identical to that provided by the relay. However, the system is not limited in control authority as is the relay. The control-surface limit-cycle amplitude should be set as low as possible (determined by system threshold and hysteresis levels) without compromising control authority. The gain changer is designed to prevent unacceptable ringing at the limit-cycle frequency during transients. Except for



a) Basic Pitch-rate control

b) Pitch-rate control with relay

c) Pitch-rate control with automatic gain changer

Fig. 1 Types of adaptive pitch-rate control loops.

this nonlinear action, system operation is essentially linear and may be analyzed as such.

A specific mechanization of the automatic gain changer is shown in Fig. 2. The input signals to the automatic gain changer are the limit-cycle amplitude set point and servo position. The servo position signal is bandpassed at frequencies near the limit-cycle frequency. Full-wave rectification provides a d.c. signal proportional to the amplitude of the limit cycle as measured at the servo. If the limit-cycle amplitude is less than that called for by the set point, the system gain will be increased. Conversely, a limit-cycle amplitude greater than commanded by the set point will effect a gain decrease. The limit-cycle amplitude error voltage is filtered, limited both positively and negatively, and integrated to produce a d.c. bias voltage that varies the loop gain of the control system.

When operated with a control loop containing negligible nonlinearities around null, the automatic gain changer is capable of maintaining a uniform system limit cycle at the set amplitude. Using practical flight-control hardware, however, with finite nonlinearities around null, the gain-changing action is cyclic. This cyclic action results primarily from control linkage backlash, which permits the gain to be raised above the critical value without incurring oscillation. Since the effective gain of a backlash nonlinearity increases with the amplitude of input, the system oscillation, once started,

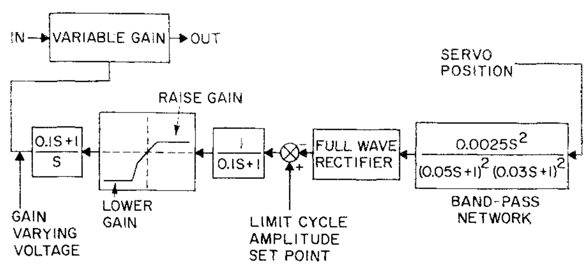


Fig. 2 Automatic gain changer.

diverges rapidly. Thus a rapid gain decrease is required from the automatic gain changer to prevent excessive limit-cycle amplitude. Once the oscillation is stopped from divergence by a gain decrease, a convergence to zero oscillation rapidly follows, supported by the effective gain decrease of the backlash. This natural process is encouraged and controlled by the nonsymmetrical limits placed on the error signal in the automatic gain changer. A low-gain increase limit keeps the gain-cycling period at an acceptable value (6 to 10 sec), whereas a high-gain decrease limit prevents excessive amplitude and duration of limit cycle. An increase in down gain is also provided to prevent excessive limit-cycle amplitude when the amplitude error reaches about one-half the set-point magnitude.

The set-point amplitude is not critical, but it must be high enough above existing noise levels to cause a sufficient gain increase rate, yet low enough to prevent excessive amplitude of oscillation. The intermittent oscillations are obviously preferable to a continuous oscillation when the effective gain set by the intermittent system is sufficiently near theoretical critical. The cyclic operation of the automatic gain changer has proved to be very desirable. Since a satisfactory analytical method of optimization has not been found for the automatic gain changer, all development has been done by means of analog computer simulation in conjunction with actual hardware, and by flight test experimentation in an F-101A.

Effects of Noise and Input Commands

Noise inputs to the system, like those experienced when an airplane exceeds its buffet boundary or flies through gusty air, may drive the system gain below satisfactory levels. Analog computer studies, however, indicate that random noise of reasonable magnitude could be tolerated. Adverse effects were noted only when the noise become periodic near the characteristic limit-cycle frequency.

Early F-101A flight tests verified analog computer results. Air turbulence caused only moderate reductions in system gain, resulting in a somewhat smoother flight while retaining precise response to command inputs. However, when flaps and dive brakes were extended and the landing gear was lowered, an aerodynamic buffet occurred with a characteristic frequency in the vicinity of the 4 cps limit cycle. As was the case in the analog computer studies, the gain of the system was forced down, especially for the smaller set-point settings. Still, the system gain remained sufficiently high to insure satisfactory performance. The system gain was driven below acceptable levels only when rapid pilot command input characteristic of the landing approach was added to the buffet characteristic.

One solution to this problem was the addition of a high pass and a limiter on the input to the gain computer in order to attenuate the pilot's inputs. A more sophisticated solution was developed in the X-20 program, as discussed later.

Aeroelastic Effects

The problem of adverse coupling of an automatic stabilization system with structural elastic modes has been aggravated by the high-gain, large-bandwidth system. This type of coupling was experienced in the early flight tests of the F-101A with the limit-cycle adaptive system. The first fuselage bending mode of the F-101A has a frequency of approximately 8 cps with very low damping, characteristic of structural elastic modes. The rate and acceleration sensors of the control system were located for practical reasons where they sensed motions associated with the first fuselage bending mode. Coupled with the high gain of the adaptive system, this was sufficient to cause the system to oscillate at 8 cps. Oscillation was prevented from diverging by the rate limiting of the power actuator and the action of the gain changer where sufficient signal was passed through the bandpass at 8 cps to limit the amplitude of the oscillation. However, this oscillation prevented the system gain from reaching the level desired for precise control. Therefore, a notch filter was installed which attenuated the sensor signals in the vicinity of 8 cps. This eliminated the 8 cps oscillation and permitted the system gain to attain the desired level.

Applications

Using the concepts previously discussed, several adaptive systems were designed, fabricated, and flight tested. The objective in each case differed somewhat, and the resulting systems featured various degrees of sophistication. Original feasibility tests were conducted in an F-94C for a single-control axis. A three-axis adaptive system with automatic gain changing was developed for tests in an F-101A aircraft. The addition of various outer-loop modes and blending of reaction and aerodynamic control were demonstrated in the MH-96 system for the X-15 research vehicle. Fly-by-wire, thrust vector control, and improved gain changer performance were extensively tested in hardware simulator studies on the X-20 Dyna Soar. A more detailed discussion of the features and test results obtained with each of these configurations follows.

Single-Axis Feasibility Tests

The early studies of high-gain model-slaving control loops utilizing a bi-stable element were supported in 1957–1958 by the Wright Air Development Center (WADC) flight control laboratory.[2] The results of these early studies were promising enough to warrant design and fabrication of a single-axis feasibility model to be tested in an F-94C aircraft. The configuration tested employed a dithered, bi-stable element and a simple gain-changer mechanism. The objectives were to determine the controller effectiveness of the basic adaptive system, utilizing dither, and to perform a preliminary optimization of the gain changer.

A simplified block diagram of the control system is presented in Fig. 3. The effect of the dither signal was to linearize the effective gain of the bi-stable element. The gain changer simply controlled the authority of the bi-stable element as a function of system model-following error. The system was designed to allow the flight test engineer to introduce and vary the width of the relay dead spot, vary the amplitude of the relay dither signal, change both the sensitivity and authority of the gain changer, select any of 11 filter combinations, select the magnitude of pitch-rate or attitude commands, permit pilot-induced pitch commands, and permit pilot to choose electric stick rate or attitude commands.

A series of flights was made to determine the effectiveness of the basic system (gain changer inoperative). Pitch-rate command and simulated gust response tests were made at selected speeds and altitudes to check the analog computer results. Maneuvers such as dives, loops, and barrel rolls showed that rapid changes in the aircraft parameters did not degrade system performance. The results can be summarized as follows:

1) Stability and control were excellent through most of the flight envelope. Correspondence between model output and aircraft pitch rate was good for pitch-rate command frequencies up to at least 10 rad/sec.

2) For the landing condition, the system was slightly underdamped (damping ratio 0.6 against the desired 0.7), and the system developed error when the cyling frequency was higher than 3 rad/sec. This performance deterioration was attributed to low elevator effectiveness (M_{δ_e}) and insufficient elevator rate capability.

3) At high altitudes and high Mach numbers, the system tended to be somewhat overdamped.

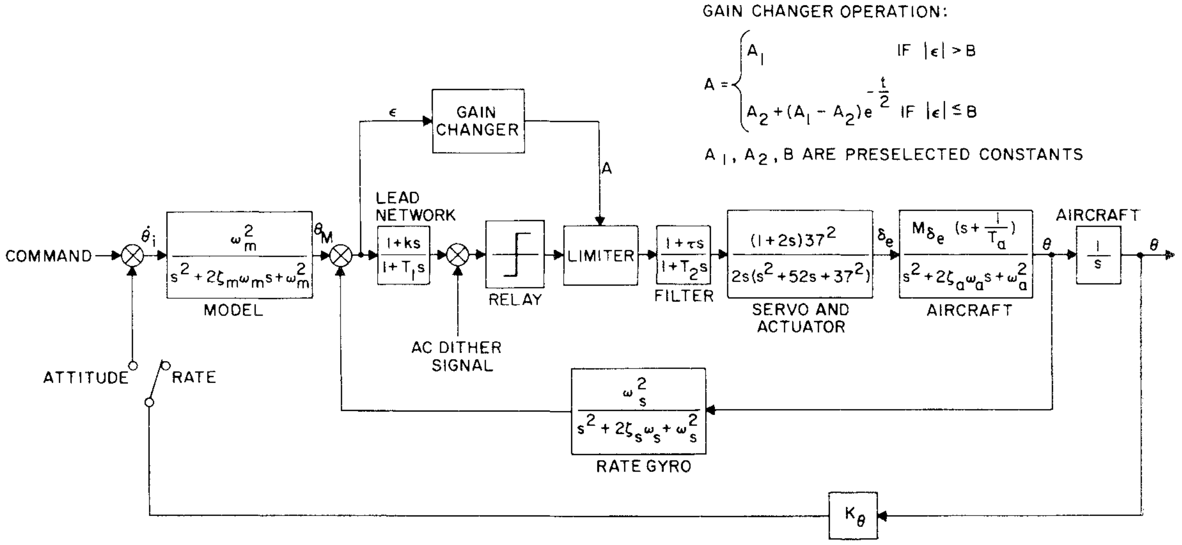


Fig. 3 Adaptive control system for F-94C.

4) At the low altitude and high dynamic pressures, there was a low-amplitude limit cycle caused by high elevator effectiveness (M_{δ_e}).

A summary plot of the performance results is presented in Fig. 4. Lack of time prevented a complete evaluation of the gain-changing concept, but the conclusions were drawn from these tests that an adaptive control system capable of providing invariant prescribed dynamic response over a very large flight envelope is not only feasible but also practical, and that the adaptive inner loop provides a good base for outer-loop modes and presents a means of eliminating the requirements for complex gain scheduling. Simple means are available for improvement of this adaptive system. Further details are given in Ref. 3.

MH-90 Three-Axis Adaptive System

Following the successful feasibility demonstrations in the F-94C, Honeywell developed a complete three-axis Automatic Flight Control System (AFCS) utilizing advanced adaptive concepts. The system employed the more refined gain-changing technique described earlier, rather than the simple bi-stable element device.

The general goals of the MH-90 development and flight test program (1958–1959) were to 1) prove the feasibility of the adaptive concept for use in a three-axis AFCS for a supersonic airplane; 2) provide satisfactory outer-loop control (attitude hold, heading hold, altitude control, and control stick steering) without the use of air data scheduling; and 3) make the operation of the adaptive system pilot-acceptable with regard to small-amplitude oscillations. An F-101A aircraft was selected as the basic test vehicle.

Figure 5 is a functional block diagram of the MH-90 pitch axis. The inputs are normal acceleration, pitch rate, pitch attitude, pitch control signal from the turn and pitch controller, longitudinal stick force, total Mach number, pitch beep trim, up elevator, and blended altitude control inputs of altitude displacement and altitude rate. Pressure altitude, pressure altitude rate, normal acceleration, and up-elevator signals are utilized to derive the blended altitude inputs. Pitch-axis control is obtained through use of a limited-authority series servo and a full-authority parallel servo. Both servos are utilized at all times when the system is engaged.

Pitch-axis functions

The following functions are provided by the pitch-axis autopilot:

1) Damping: Basic damping in the pitch axis is obtained through use of a normal accelerometer and a pitch-rate gyro. The normal-acceleration signal is predominant at low frequencies, and all commands are essentially normal-acceleration commands except at low airspeeds. The pitch-axis gain is controlled by a gain changer that operates in the same manner as the gain changer shown in Fig. 2. The series servo follow-up is high-passed for nominal integrating action, but utilizes a low-gain, steady-state follow-up to keep the operating point at the center of its limited authority ($\pm 1.5°$ stabilator). The commands to the pitch-axis model are provided with two sets of limits; stick force command limits are set at 4.0 and −1.0 g's absolute, whereas all other command limits are set at 2.5 and 0.25 g's absolute. These limits protect against most electronic failures and excessive g commands from flight test inputs or altitude control engagement during rapid descents or climbs.

2) Speed stability: A low-gain Mach control is provided in the pitch axis to achieve a long-term stability characteristic.

3) Pitch control stick steering: The pitch axis utilizes a stick force transducer to command aircraft normal acceleration plus pitch rate.

4) Pitch attitude hold: The pitch attitude hold mode of the MH-90 system is obtained by effectively changing to a pitch-rate inner loop. This effective pitch-rate inner loop results from addition of a negative normal-acceleration signal

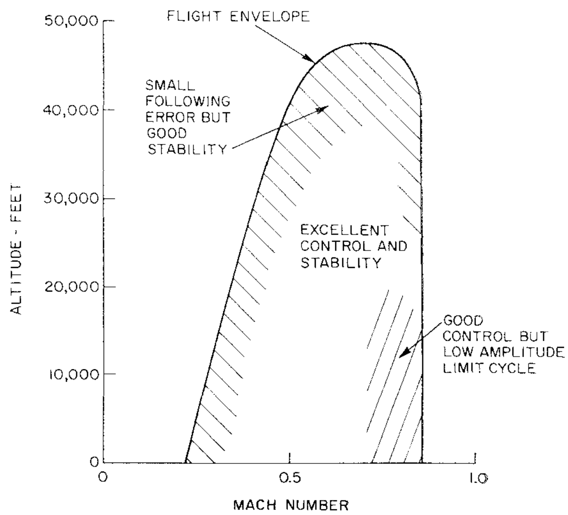


Fig. 4 Summary of F-94C flight test results: pitch-rate control with minimum system and no gain changer.

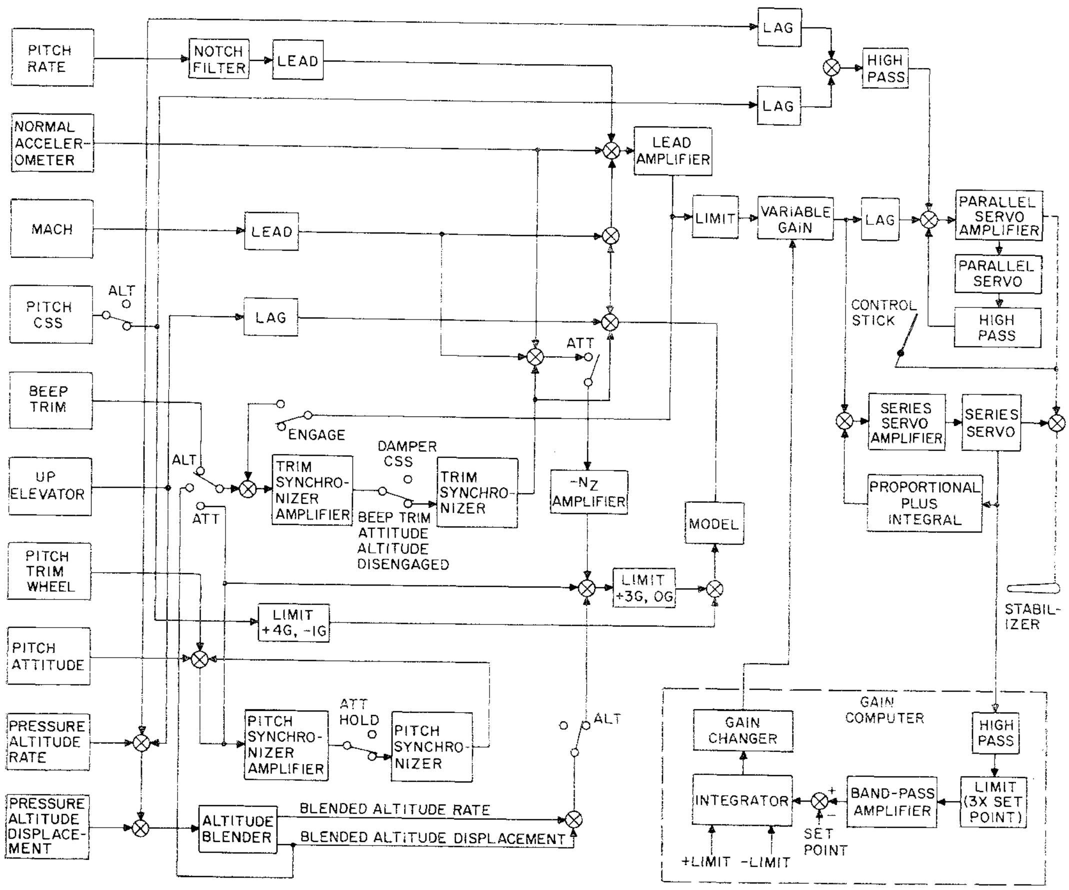


Fig. 5 MH-90 pitch-axis configuration.

that is summed with the pitch-attitude error signals when the pitch-attitude mode is engaged.

5) Altitude control: The blended altitude control in the system provides the necessary filtering for a tight altitude hold mode without erratic inputs from the pressure-sensing elements. The inputs (pressure altitude, pressure-altitude rate, normal acceleration, up elevator, and trim) to the blended altitude are summed with the derived altitude displacement and derived altitude rate with the proper gains to provide a second-order characteristic.

Performance results

The operation of The limit-cycle adaptive system in the F-101A largely confirmed predictions from theoretical analysis.[4] The responses to command inputs were uniform over the entire flight regime, and the limit cycle was undetected by the many pilots who evaluated the system. Only the operation of the gain changer itself was different from that expected from theoretical analysis. Instead of providing a continuous limit cycle, the limit cycle appeared and disappeared with an associated fluctuation in system gain. However, this fluctuation in system gain was small and had little effect on system performance, because average gain was well above that required to assure uniform response to command inputs.

The effect of random noise inputs (like that experienced in turbulent air) did not degrade system performance. Structural elastic mode coupling was experienced because of the large bandwidth of the adaptive system, but was eliminated by appropriate filtering. Typical pitch-axis responses and gain changer performance are illustrated in Fig. 6.§

§ The modified MH-90 has been in use for the past three years at the Aerospace Research Pilot School, Edwards Air Force Base. The F-101A has been modified as a variable stability space trainer for astronauts to simulate a wide range of entry space vehicle configurations.

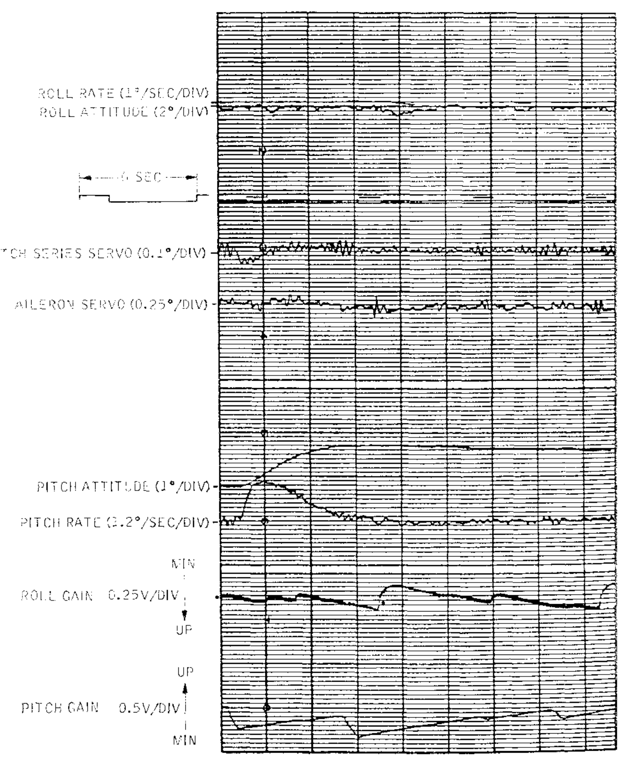


Fig. 6 MH-90 pitch attitude step — 10,000 ft, Mach 0.9 (double 0.35-sec lag model, $\zeta = 1$, $\omega = 3$).

MH-96 Adaptive System for the X-15

A number of self-adaptive techniques were investigated for possible application to the control of manned space vehicles such as the X-15. Among the criteria established as a guide in selection of a configuration for this application were the following: the controller should provide reasonably uniform inner-loop dynamic response; the self-evaluation process must be positive enough to compensate for aerodynamic changes occurring during re-entries and static instabilities that may occur as a function of angle of attack and/or sideslip; the controller should provide information for logical transition from aerodynamic control to blended aerodynamic-reaction control, and vice versa; and the concept must lend itself to a reliable hardware mechanization.

The basic concepts developed for the F-101A were selected for the X-15 because the system concept had already been proven through extensive flight testing in a supersonic aircraft. Furthermore, the system readily lends itself to a fail-operational redundant mechanization to achieve high reliability and fail-safety. It was also important for the X-15 that external inputs not be required for operation of the self-evaluation loop.

System mechanization

The system mechanization, represented by the pitch-axis diagram in Fig. 7, is an expansion of the MH-90 F-101A adaptive autopilot in the following areas: the X-15 system is dual-redundant throughout, with a fixed-gain backup mode; the system automatically blends reaction control and aerodynamic control; the system incorporates filters for body bending and tail-wags-dog effects; the system is designed for compatibility with the full-rolling tail of the X-15; and the system provides an angle-of-attack hold mode as well as attitude and heading hold modes.

Other unique features of the system include the automatic trim mechanization, input command signal limiting, and vernier control capability for both angle of attack and pitch attitude. In addition, the system incorporates a yaw rate-to-aileron feedback for improved roll axis stability at extreme angles of attack. A comprehensive discussion of the entire MH-96 development program (1959-1962) is presented in Refs. 5–11 and is summarized in Ref. 12; therefore, a detailed description of the system is not presented here.

Test results

To date, 43 flights have been made with the MH-96 AFCS installed in the Number 3 X-15 research vehicle, including the record manned aircraft flight to 354,000 ft. In all these flights the adaptive system performed to plan and no malfunctions occurred.¶ Some important details of the flight test activity follow:

1) Prior to the first adaptive flight test, the possibility of coupling with a poorly damped control surface aeroelastic mode at a frequency of 12 cps was discovered during flight test of a conventional damper system in one of the other X-15 vehicles. Through computer studies, it was determined that considerable filtering would be required if the adaptive system with its high gains were to avoid this problem. The required filtering was designed, fabricated, and installed. This resulted in these effects: first, the limit-cycle frequency was reduced from 4.0 to 2.5 cps because of unavoidable filter lags at the limit-cycle frequency; second, in an attempt to reduce the lags at the limit-cycle frequency, a peaking in system gain at approximately 6 cps resulted. This, in turn, required a reduction in the maximum system gain to avoid the tail-wags-dog effect, a limit cycle that is purely inertial in origin.

¶ See also Refs. 13 and 14 for comments on the X-15 accomplishments with the MH-96.

2) The gain in each axis varies over a range of approximately 20/1, somewhat less than planned initially because of the necessity to avoid the tail-wag-dog effect. Thus the gain in each axis reaches its maximum value at a dynamic pressure of approximately 300 psf instead of the planned-for 140 psf.

3) The gain-changer operation was actually smoother than anticipated from analog computer and simulator studies because of a smaller backlash in the Number 3 X-15 control system than that experienced on the simulator and described analytically.

4) The limit cycle associated with the gain-changer operation has not been found objectionable and on most flights has not been detected by the pilot.

5) Operation of the automatic blending of the aerodynamic and reaction controls has been so smooth that the pilot is unaware of the transition.

Figure 8 shows the gain change occurring as a function of time for flight number 5. Superimposed upon this is the aerodynamic surface effectiveness (M_{δ_e}) in pitch, showing that the gain changes as an inverse function of surface effectiveness.

X-20 (Dyna Soar) Flight Control System

The MH-132 three-axis flight control system development was started in January 1961 and terminated in December 1963. At the time of the X-20 project cancellation, development had been carried through fabrication of prototype models and partial testing of these models on Boeing and Honeywell simulators.**

Numerous demanding constraints influenced the design of the MH-132 system. The more important were optimum control of thrust vector rockets during abort to maximize recovery capability, minimum expenditure of reaction control fuel, no degradation of system performance for any single failure, control of a statically unstable vehicle in pitch at some flight conditions, accurate g limiting, very stringent handling quality requirements, and control of a flexible vehicle with widely varying modal frequencies complicated by the tail-wags-dog phenomenon.

The pitch-axis block diagram is given in Fig. 9. The system augmented the glider's natural aerodynamic stability and provided maneuvering control over the entire flight regime. Glider control was provided by pilot or inertial guidance system commands through either adaptive or fixed gain systems. The augmentation system kept forces acting on the glider within tolerable limits through command signal limiting, sideslip control, and normal-acceleration control. Four operating modes were provided:

1) Manual Direct (MD): This mode gave the pilot direct control over the aerodynamic surface servos, thrust vector servos, and reaction control jets. The stick and pedal-to-control-surface displacement gain could be varied by the pilot with a gain switch. Three levels of gain were provided as a function of Mach number.

2) Augment (AUG): This mode provided adaptive stability augmentation and control in all three axes to improve the handling qualities of the vehicle over the entire flight regime. The effectiveness of the stick rate-commands was governed by the adaptive yaw and pitch-axis gain computers.

3) Pilot-selectable, fixed gain (FG): This control loop was identical to the augment mode, except that the adaptive gain computers were by-passed. Three levels of preset gains in all three axes were slaved to the gain switch under control of the pilot. This was the same switch that controlled the manual direct gains and, hence, selectable gains were referenced to the same Mach regimes as manual direct.

** Reference 15 documents the X-20 flight control system development program and contains a bibliography of the relevant Boeing, Honeywell, and U. S. Air Force Aeronautical Systems Division (ASD) reports.

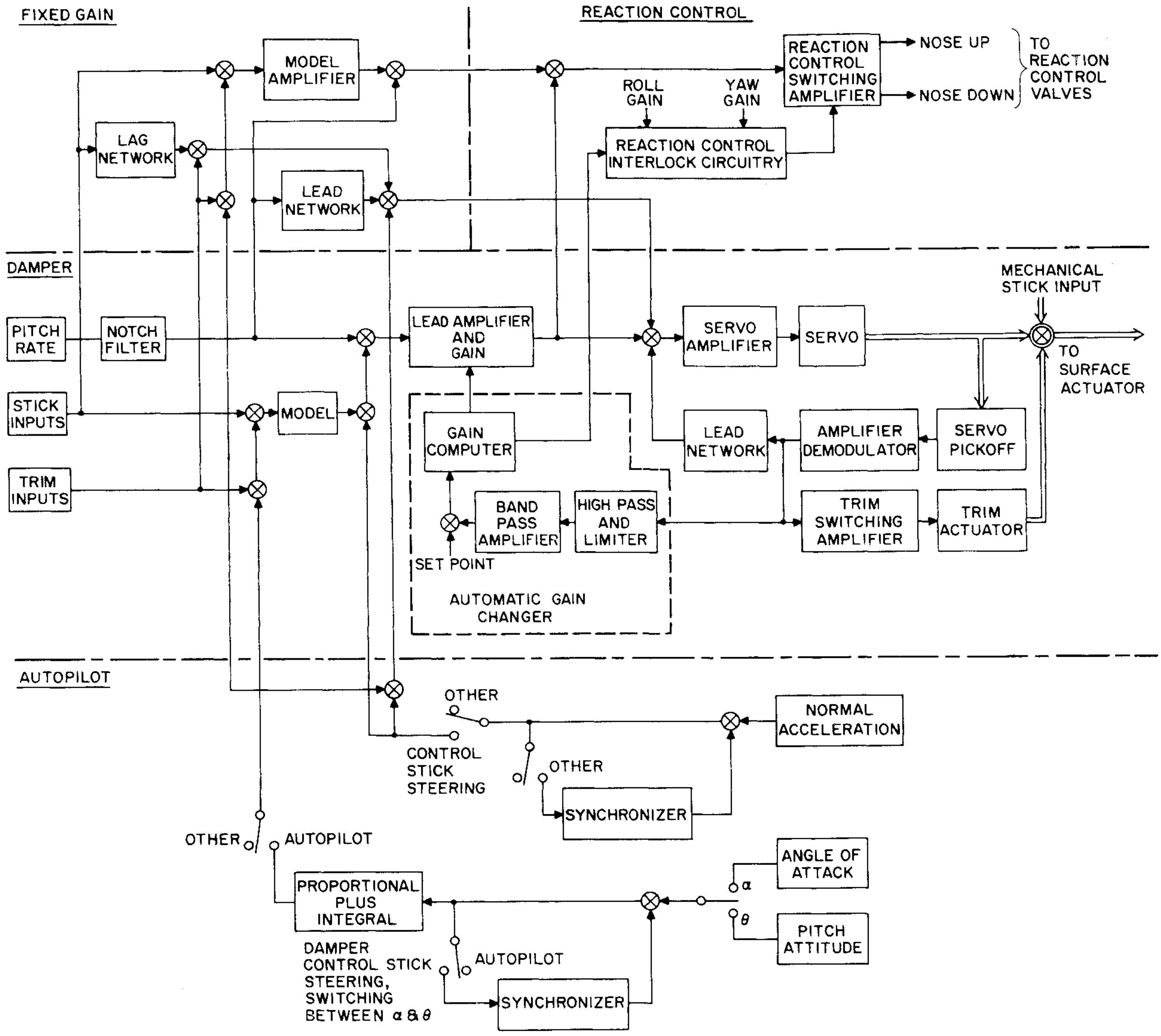


Fig. 7 MH-96 pitch-axis configuration.

4) Automatic: In this mode, angle-of-attack commands and angle-of-attack references were provided by the inertial guidance system. As in the augment mode, command authority was governed by the adaptive gain computers.

Gain controller design

The following design criteria were chosen for gain changer optimization: when subjected to random gusts and normal stick commands, an average gain of at least 50% linear critical should be held; it must be impossible to excite the gain changer such that it would hold a gain higher than linear critical; gain changer response speed should be fast enough to stabilize the statically unstable glider when starting up from minimum gain; the nominal limit-cycle amplitude should be $\pm 0.1°$; down-logic authority should exceed up-logic authority by a factor of at least two; and body-bending frequencies should have a negligible effect on the gain changer.

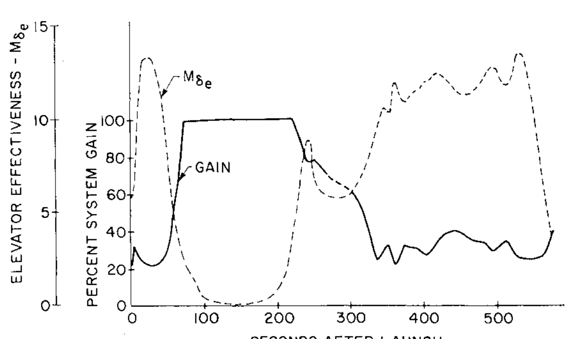


Fig. 8 Surface effectiveness and gain variations during a typical X-15 flight.

The X-20 control system was initially expected to use the X-15 gain controller. During the course of study, it became apparent that simple down-logic filtering was not adequate, so up-logic filtering was added. This was necessary because:

a) In the X-20 augmented mode, the control authority is proportional to the adaptive gain. Therefore, gain holding was essential to satisfactory control of the X-20 vehicle.

b) Gain could not be held when the system was subjected to random noise, gust, and stick inputs without a large increase in the nominal limit-cycle amplitude. Furthermore, body-bending and tail-wags-dog considerations necessitated reduction of limit-cycle frequencies from a nominal 4 to 1.8 cps and 3.0 cps in pitch and yaw, respectively. Addition of actuator flow hysteresis has the effect of making limit-cycle amplitude, frequency, and forward-loop gain interdependent such that to hold a nominal limit-cycle amplitude of $\pm 0.1°$ surface, the pitch frequency varied from 1.2 to 1.7 cps, and the yaw frequency varies from 2.6 to 2.8 cps.

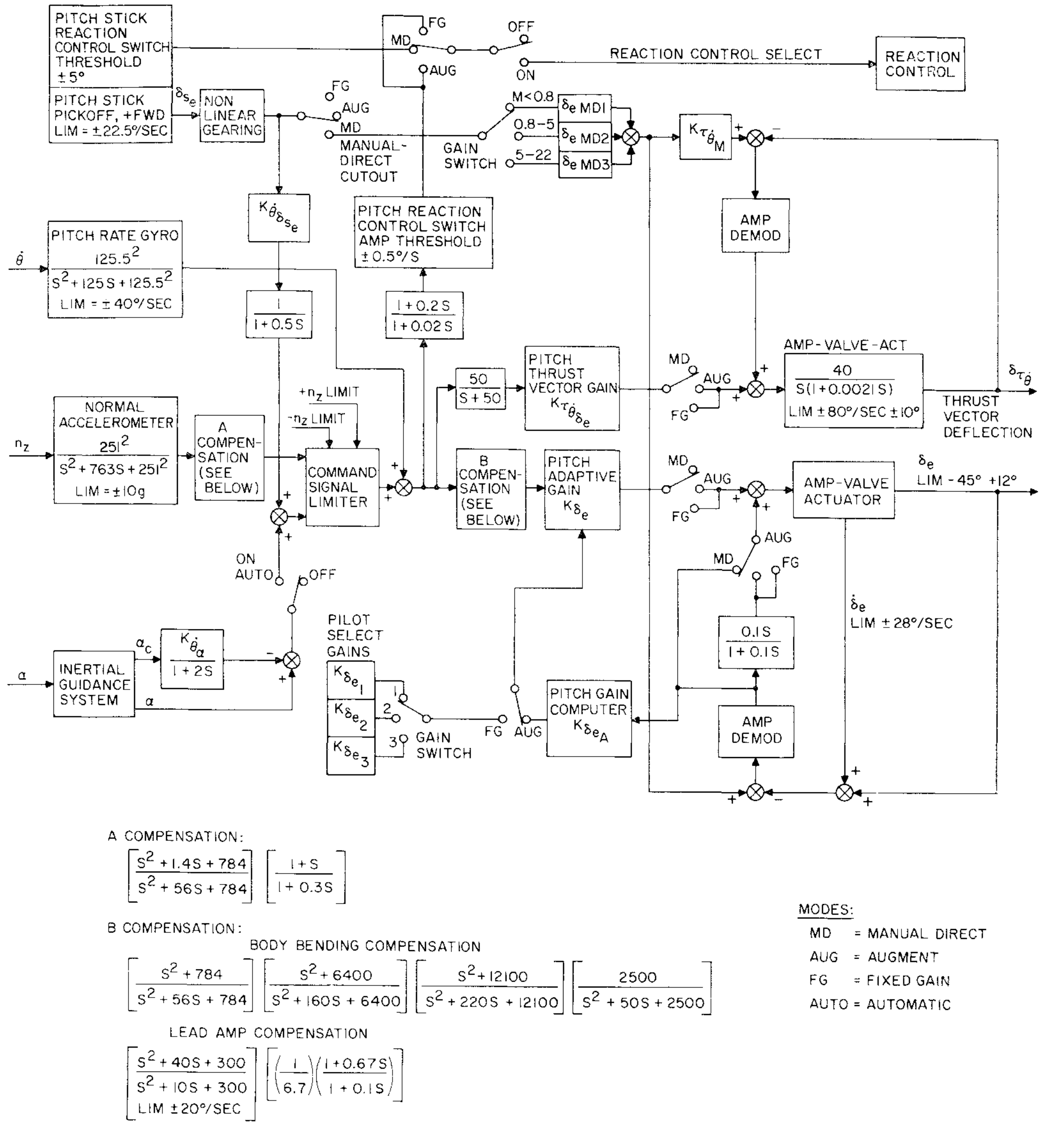


Fig. 9 MH-132 pitch-axis configuration.

c) The stable gain range for the abort vehicle became too narrow for any but a highly adaptive control system to stabilize.

A block diagram of the pitch-axis gain changer is shown in Fig. 10. Servo actuator rate plus displacement was fed to up-and-down-logic bandpass filters, the outputs of which were rectified and summed with the set point to provide a d.c. signal. This signal was integrated (proportional plus integral) and the integrator output drove an electronic gate through a nonlinear gain, the width of which was proportional to the forward-loop control system gain.

The primary difference between the pitch and yaw gain changers was the peaking frequency of the down-logic bandpass filters, 12.5 rad/sec for pitch and 20 rad/sec for yaw. This difference was accomplished by placing the underdamped second-order natural frequency at 13 and 20 rad/sec in the pitch and yaw axes, respectively. The up-logic band-passes were identical, peaking at 2.3 rad/sec.

The set point voltages indicated in Fig. 10 provided nominal limit-cycle amplitudes of about 0.2° surface depending upon the limit-cycle frequency at which a given flight condition operated. The smoothing filters were 0.5 and 0.2 sec first-order lags in the pitch and yaw axes, respectively; gain computer dynamics were identical, consisting of a $(2 + s)/s$ proportional plus integral with a gain of 1/20. The nonlinear gain is shown in Fig. 10.

The pitch axis used 12-v rate limits and the gain change went from minimum to maximum gain in 16 sec with saturated limits; the yaw axis used 10-v rate limits and required 20 sec to traverse its gain range. Up-logic authority was limited to 20 v or about twice the rate limit saturation level. Down-logic authority was limited at 40 v, twice the up-logic authority.

Performance

Simulator studies using actual flight control system hard-

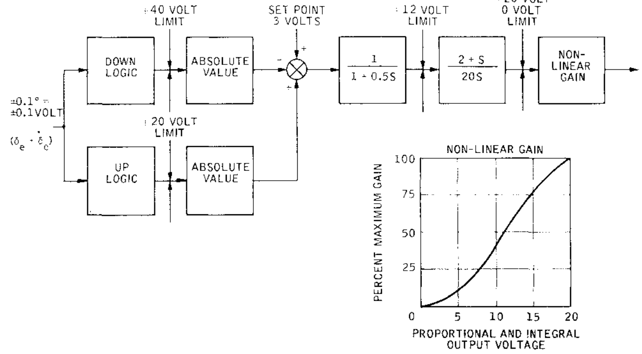


Fig. 10 MH-132 pitch-axis gain changer.

ware demonstrated that the gain changer design satisfied the performance requirements.

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