DISCUSSION OF THE HONEYWELL ADAPTIVE FLIGHT CONTROL SYSTEM FOR HIGH-PERFORMANCE AIRCRAFT

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1. INTRODUCTION

Encouraged by the F-94C flight test results and bolstered by a better analytical understanding of nonlinear control systems, we at Honeywell initiated two new adaptive control system development programs:

1. The development and flight test of a complete three-axis adaptive flight control system for application to high-performance aircraft.

2. The development and flight test of a simplified adaptive control system.

It is appropriate to discuss in this paper only the first of these two systems as this will provide a comprehensive coverage of the Honeywell adaptive flight control concepts.

Since the F-94C flight tests, much effort has been directed toward extending the adaptive concept to all axes of control and in overcoming the deficiencies found in the flight test of the F-94C system. As a result, a complete adaptive flight control system has evolved and is being readied for flight tests in a F-101A fighter aircraft.

2.0 DISCUSSION OF THE ADAPTIVE INNER LOOP

The model and the adaptive controller form the kernel of the Honeywell adaptive flight control system. Figures 3 and 4 are block diagrams of the pitch and lateral axes showing these units in their proper perspective to the complete system.

Being common to all axes of the system, the model and adaptive controller will be discussed in some detail. Consider the block diagram shown in Figure 1.

2.1 Model

The model in the adaptive flight control system provides the system with the selected standard of performance. It represents what the pilot wants in his aircraft handling characteristics. Hence, the model is an analog simulation of an ideal aircraft.

Specifically, in the pitch axis where normal acceleration is the major adaptive loop feedback, a second order model with a damping ratio of unity and a natural frequency of 2 radians per second is used. In the roll axis a first order lag with a time constant of 0.5 second is used.

Electric commands to the model are obtained from a stick force transducer located on the control stick of the airplane. As the pilot applies force to the control stick, an electrical voltage, which is proportional to the applied force, is generated and transmitted to the model. The output of the model represents the desired airplane response. In other words, the output of the model is the specific performance the pilot would like to have for the given input.

2.2 ADAPTIVE CONTROLLER

If the aircraft response could be made to duplicate the response of the model, then it will, of course, respond as the pilot would like it to respond. This is precisely the job of the adaptive controller, that is, to force the aircraft to follow closely the output of the model.

Studies conducted at Honeywell have shown that the " following error" can be kept reasonably small if the inner loop natural frequency is at least 5 to 10 times that of the model. Hence, the function of the adaptive controller is to provide the compensation and gain adjustment necessary to meet this criteria at all flight conditions.

The adaptive controller which contains no moving parts consists of a compensating network, a modified bi-stable element, an electronic integrator and an automatic amplitude modulator (see Figure 1). The compensating network is a phase lead network which operates on the error signal to provide anticipation and to maintain the frequency of the limit cycle relatively constant over the flight regime of the aircraft. How this is accomplished is discussed in the previous paper by Luther Prince.

The modified bi-stable element is essentially a high gain linear amplifier with limited output. The gain and limits are variable in the manner shown in Figure 2.

The electronic integrator is used to obtain a proportional plus integral signal in the forward path of the inner loop.

The automatic amplitude modulator senses servo motion at the limit cycle frequency only and varies the gain of the modified bi-stable element to maintain the forward loop gain at the highest stable value for all flight conditions. Hence the automatic amplitude modulator compensates for changes in aircraft control surface effectiveness.

Now the system operates with an error sufficiently small to stay within the linear portion of the modified bi-stable element. In accomplishing this, a small controlled residual motion exists at the frequency corresponding to the neutral stability point. The amplitude of this residual motion is kept constant at the servo output by the automatic amplitude modulator. If the amplitude of motion is larger than that designated by a bias voltage, the gain of the modified bi-stable element is decreased, and vice versa.

In the yaw axis there is a deviation from the above discussion as the gain of the modified bi-stable element in yaw is adjusted by the roll automatic amplitude modulator. Hence, no automatic amplitude modulator is included in the yaw axis. No characteristic residual motion exists in the yaw axis as the gain of the modified bi-stable element is adjusted to be less than that required to sustain a limit cycle.

2.3 GENERAL CHARACTERISTICS

2.3.1 COMMAND AND GUST INPUTS

Since the system gain for small error signals is maintained at that gain required to produce a limit cycle at a frequency at least five to ten times that of the model, the system output will follow the model very closely with the error signal rarely becoming large enough to exceed the small linear band of the modified bi-stable element.

However, if the aircraft is excited by a gust, the transient error in the inner loop becomes quite large, and, as can be seen in Figure 2, the effective gain of the modified bi-stable element is sharply reduced during the transient. This is a desirable and necessary feature of the system; because it increases the system phase margin, and hence damping, for external disturbances without compromising system " following error" for signals fed through the model.

2.3.2 CHARACTERISTIC RESIDUAL MOTION

The size of the characteristic residual motion is set just large enough to overcome the various thresholds of the system. In the case of the F-101A, the amplitude of the surface residual motion will be kept under 0.1 degree at a frequency of 6 cps. The resulting pitch and roll rate and acceleration experienced by the aircraft are well under the pilots threshold. The pitch attitude residual motion is kept under 1 mil.

3.0 GENERAL DISCUSSION OF BLOCK DIAGRAMS

The objective set up for the F-101A system is optimum performance with no air data scheduling of parameters.

3.1 PITCH AXIS

As it can be seen in Figure 3, three outer loop modes in the pitch axis are provided in the F-101A system. They are control stick steering, attitude and altitude hold. Automatic glide slope and flare-out systems can be added easily.

To eliminate the need for scheduling in certain outer loop controls (such as, altitude hold, Mach hold, etc.), a pitch rate inner loop is desirable; and for certain other outer loop controls (such as, altitude hold, flight path commands, control stick steering, etc.), a normal acceleration inner loop is preferred. The F-101A system mechanization is such that it allows the effective utilization of either a normal acceleration inner loop or a pitch rate inner loop depending upon the mode of operation. A consequence of this is the complete elimination of air data scheduling in all modes of operation.

In aircraft handling qualities, a pilot prefers a constant stick force per " g" characteristic. For control stick steering then, normal acceleration plus high-passed pitch rate are fed back as the primary inner loop signals. The ratio of normal acceleration to pitch rate is adjusted to provide adequate damping and an essentially uniform normal acceleration response throughout the aircraft flight regime.

For the attitude hold mode, the adaptive inner loop system is essentially switched to a pitch rate system which is as earlier mentioned, the ideal inner loop system for attitude control. The addition of a high-passed pitch attitude feedback and a negative normal acceleration signal, fed through the model, effectively change the inner loop to a pitch rate system. It can be shown mathematically that the result of adding these two feedbacks is to effectively remove the high-pass from the pitch rate signal and to attenuate the normal acceleration feedback in the range of model control frequencies. Hence, it is possible to achieve uniform dynamics over the flight range without resorting to air data scheduling. A low gain integration through the trim synchronizer is used to ensure a droopless system.

For the altitude hold mode, the normal acceleration adaptive inner loop is preferred because both rate of change of altitude and normal acceleration are functions of forward velocity. No pitch attitude loop is in the system when on altitude hold. For stability purposes this signal is replaced by altitude rate from an inertially augmented altitude controller which will provide a good altitude rate signal. A low gain integration on the altitude displacement signals is provided as in the pitch attitude case.

Input command signal limiting is achieved by a diode limiter just ahead of the model.

In the pitch axis, the series servo and parallel servo are driven by the output of the adaptive controller through a splitter network. The function of the splitter network is to separate the incoming signal into a high frequency band and a low frequency band. The high frequencies are fed to the series servo and the low frequencies to the parallel servo. In this manner, the characteristic residual motion and high-frequency damping is accomplished by the series servo while the parallel servo will, for the most part, reflect only the gross surface motions encountered during a maneuver. Selection of the splitter network time constant is governed by the available series servo authority and desired stick feel.

3.2 ROLL AXIS

As shown in Figure 4, roll rate is used for the primary inner loop feedback and four outer loop modes are included: control stick steering, altitude and heading hold, and heading select.

For the control stick steering mode, summing the stick force signal with roll rate only, provides a constant stick force per degree per second roll rate throughout the flight regime.

Roll attitude hold is provided by merely feeding a roll attitude signal to the constant characteristic roll rate adaptive loop. Roll attitude is automatically engaged when the roll stick force is below a certain limit.

The heading hold and heading select modes are obtained by adding a heading feedback to the roll attitude mode. The heading hold mode is automatically engaged when the bank angle is less than 7 degrees and when the stick is centered laterally. Heading select is obtained through the heading synchronizer by a manual input selector. A unique feature of the heading select mode is that the gain of the heading error signal is a nonlinear function of heading error. Thus, the heading response at high speeds is improved because the commanded bank angle will be held at its limit value until the aircraft is nearly at the selected heading.

As in the pitch case, a command signal limiter is provided to prevent excessive outer loop commands.

All signals out of the adaptive controller are fed into the two series type servo actuators located at each aileron.

3.3 YAW AXIS

The yaw adaptive control loop (Figure 4) utilizes high-passed yaw rate, and lateral acceleration to provide dutch roll damping and transient turn coordination for the yaw axis. No outer loop modes are used in the yaw axis.

Since the rudder pedals apply direct mechanical inputs to the rudder actuator, manual inputs may be made by the pilot. This manual input establishes a transient command in lateral acceleration which will eventually be reduced to zero, within the limits of the series servo authority, through the integral action of the adaptive controller.

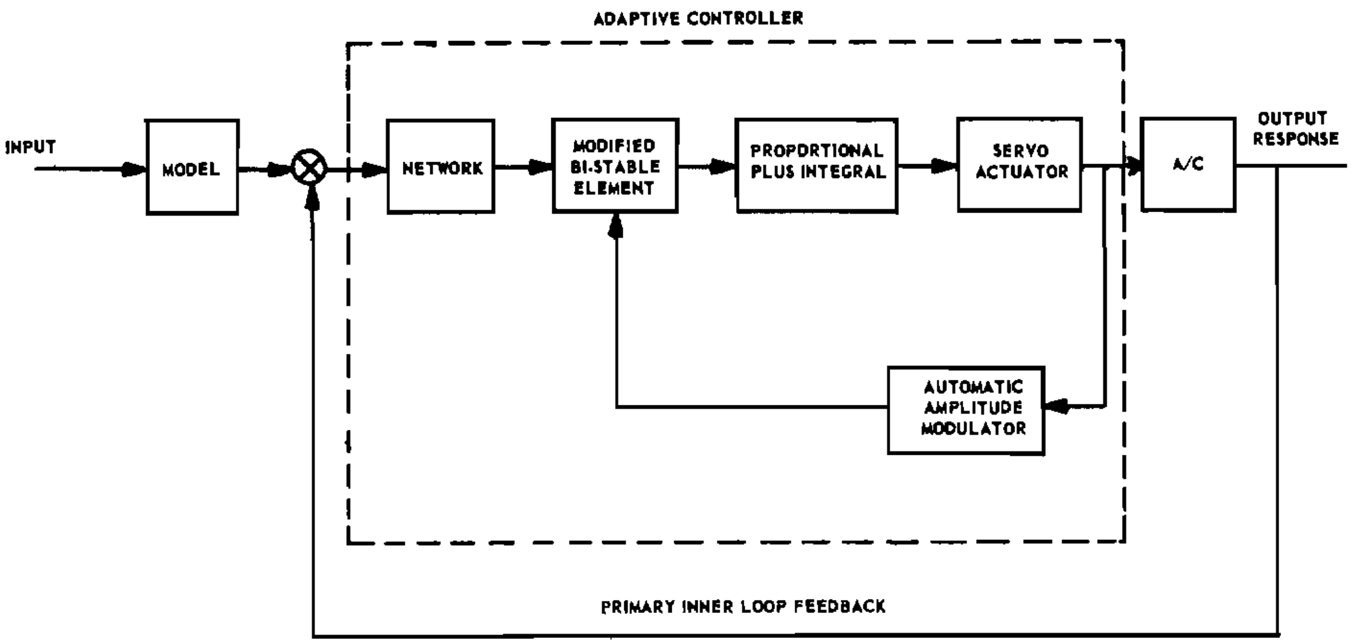

Figure 1. Typical Adaptive system

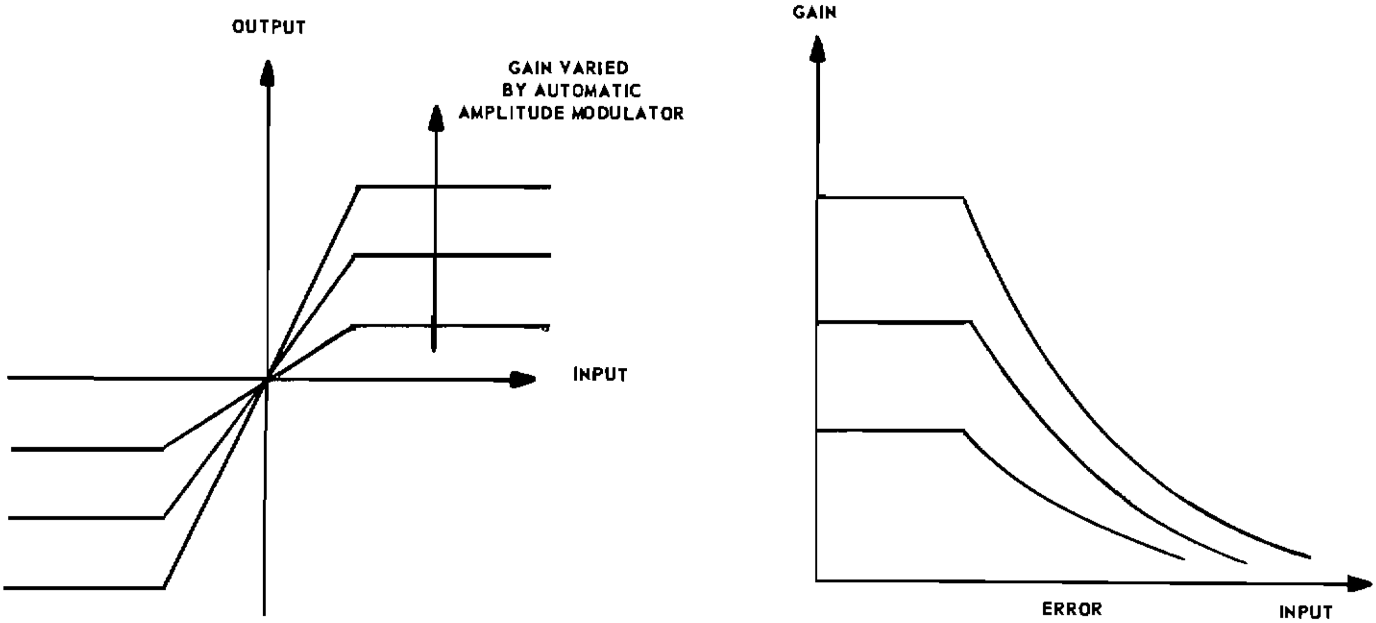

Figure 2. Characteristics of Modified Bistable Element

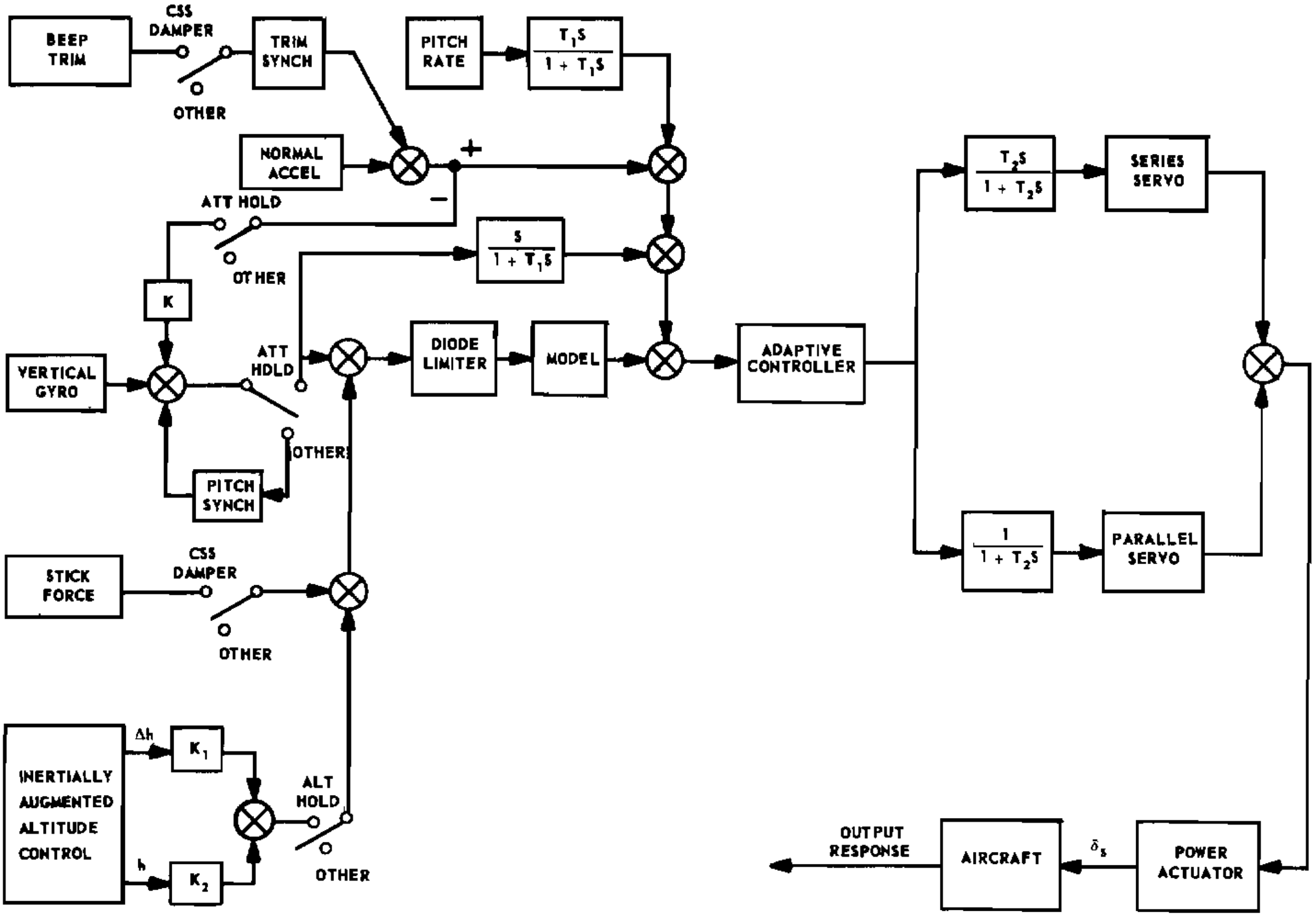

Figure 3. Pitch Axis Block Diagram - F101A

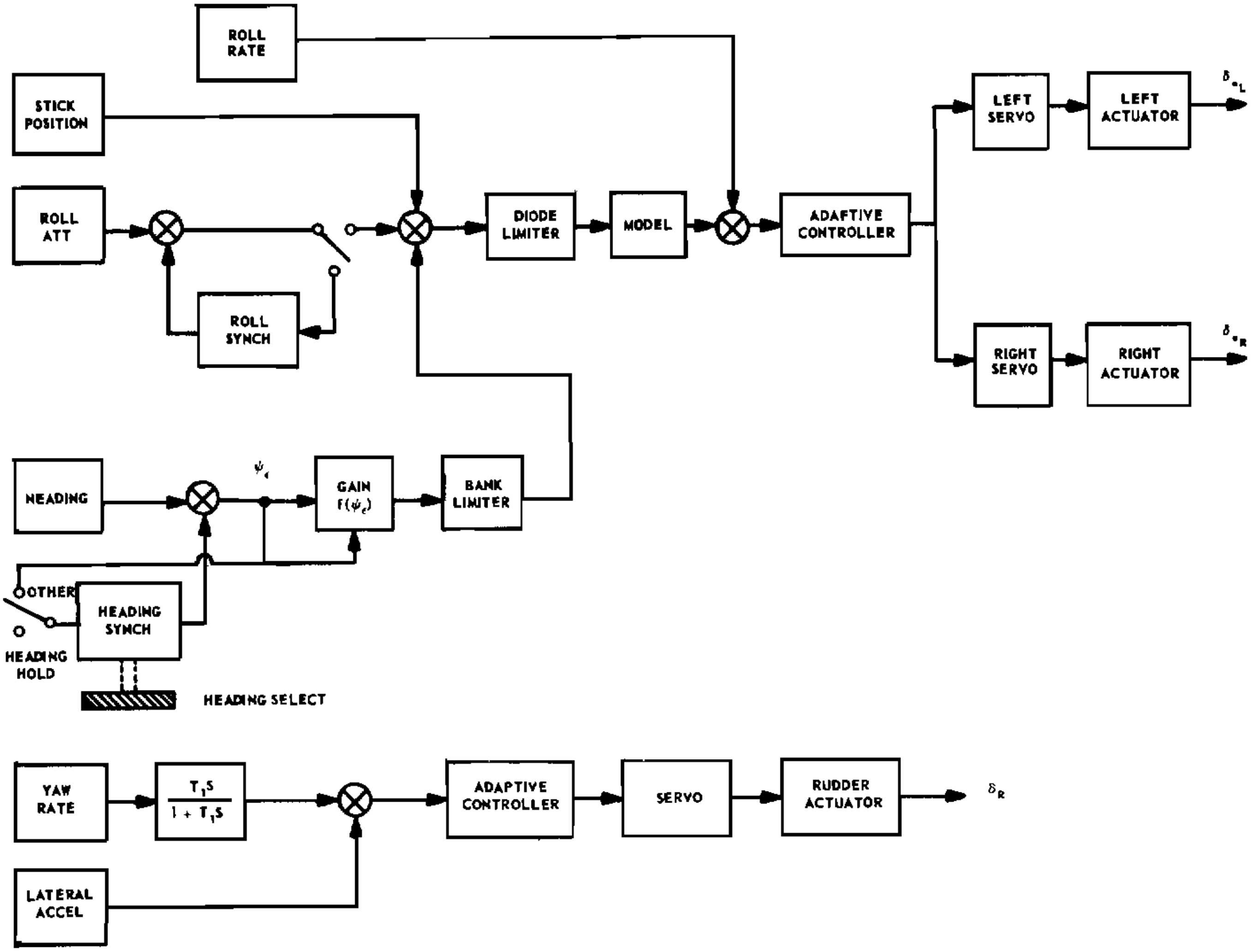

Figure 4. Yaw and Roll Axis Block Diagram - F101A