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EVALUATION OF A VTOL FLIGHT-DIRECTOR CONCEPT DURING CONSTANT-SPEED INSTRUMENT APPROACHES

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16. Abstract

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SUMMARY

A flight investigation was conducted to determine the potential instrument-approach capability afforded by a helicopter type VTOL flight-director system used in conjunction with a relatively sophisticated control augmentation system. Simulated instrument approaches were flown at a constant speed of 45 knots along a straight-in 6° glide path to a 50-ft (15 m) altitude.

The results indicate that the pilot can perform acceptable approaches on a consistent basis and uses relatively small control inputs. The flight-director indicator provides a good indication of the necessary control action but, at the same time, requires considerable attention by the pilot to assure himself that he is making the proper control inputs. This situation tends to detract from the pilot's ability to scan the remaining instruments which, in turn, results in his having some doubt as to the overall status of the approach. The tests indicated that modifications to the display sensitivity, logic mechanization, and display integration should tend to alleviate the scanning problem. In addition, a reduction in sensor noise should also result in a lower pilot workload.

INTRODUCTION

Although flight directors have come into widespread use for performing instrument approaches with conventional aircraft, their application to VTOL aircraft has been very limited. Early attempts at adapting the flight-director concept to VTOL aircraft (for example, refs. 1 to 4) were hampered by the poor low-speed handling characteristics of the vehicles employed. Without adequate stabilization, the vehicle motions that were reflected back into the flight-director commands produced high-frequency commands with which the pilot could not cope.

With the advent of high-gain control augmentation systems, VTOL aircraft can be provided with virtually any level of stabilization required for a given task. It appears desirable, therefore, to reevaluate the application of flight directors to VTOL aircraft in which some form of stabilization is used to lower the frequency content of the required control inputs. A flight investigation was therefore initiated to determine the potential instrument-approach capability afforded by a VTOL-oriented flight director used in conjunction with a high-gain control augmentation system. The tests were conducted with a CH-46C research helicopter employing a "fly-by-wire" attitude command system designed to provide inner-loop stabilization which allows the pilot to concentrate more fully on following the guidance information. Power (collective), roll, and pitch flight-director commands were presented for the control of glide slope, localizer, and speed, respectively. The present tests involved only the constant-speed approach capabilities afforded by the system, although the pitch command could be easily programed to direct a decelerating approach to a hover. The approaches were flown under simulated instrument (hooded) conditions at a constant ground speed of 45 knots along a straight-in 6° glide path to an altitude of 50 ft (15 m).

DESCRIPTION OF EQUIPMENT

Test Aircraft

A photograph of the helicopter used in the present flight investigation is shown in figure 1. The helicopter has been equipped with a control augmentation system which utilizes a central analog computer to process the pilot's control input signals and various sensor signals. The computer outputs position the control surfaces of the helicopter by means of electrohydraulic actuators.

The control augmentation system provides attitude stabilization in pitch and roll and an automatic turn-following system in yaw. The system also suppresses the response of the vehicle to external disturbances with a level of resistance which is essentially independent of the vehicle response characteristics to pilot control inputs. This feature is discussed in the appendix which describes the mechanization of the control augmentation system.

The control response characteristics provided in the four controlled degrees of freedom are given in table I. The pitch and roll characteristics were selected on the basis of the results reported in reference 5, whereas the turn-following characteristics were based on the results of reference 6.

In pitch and roll, pilot control inputs produce attitude changes proportional to control position; in other words, the pilot's control position determines the vehicle steady-state attitude. The relatively high control sensitivity employed is dictated by the levels of attitude stiffness (natural frequency) and attitude sensitivity chosen. During the initial flight check of the control system the pilot noted that, although the attitude stiffness, attitude sensitivity, and time to achieve the commanded attitude were satisfactory, the initial response was too abrupt. Furthermore, there was a tendency for the aircraft vibrations to couple with the control system through the pilot's control stick. These problems were cured by filtering the pilot's control input signal with a 0.1-sec first-order filter.

In yaw, the vehicle would automatically follow stick-initiated turns and pedal inputs could be used to produce intentional sideslip. The vertical degree of freedom is independent of the control augmentation system and controlled through the basic collective control system of the helicopter.

Guidance System

The flight tests were conducted at NASA Wallops Station with the tracking radar system described in reference 7. Aircraft position relative to a fixed point on the ground was sensed by the radar and transmitted to the aircraft via a telemetry link. This information was processed in the central onboard analog computing equipment to determine the aircraft position and rates relative to the desired approach path.

Instrumentation

During the research flights, data were recorded at the radar ground station and on board the aircraft. The ground-base data consisted of plots of altitude against range, cross-range against range, and range-rate (i.e., X-component of ground speed) against range. On board the aircraft, control positions, flight-path deviations, range-rate error, position of the flight-director command needles, and aircraft attitudes were recorded on magnetic tape. Angular rates, accelerations, and other standard measurements were recorded on an on-board oscillograph.

DISPLAY SYSTEM

Pilot's Instrument Panel

Figure 2 is a photograph of the evaluation pilot's instrument panel. The flight instruments included an attitude-director indicator (ADI), moving-map display, altimeter, instantaneous vertical speed indicator (IVSI), airspeed indicator, and a power (collective) lever position indicator. Control position commands for pitch, roll, and power were generated in the onboard computer and presented on the ADI as indicated in figure 3. The gains on the flight-director signals were established on a fixed-base simulator and verified in flight. The gains employed for the present tests are given in table II.

The flight-director logic programed into the computer dictated that glide-slope corrections be made with the power control, localizer corrections with the roll control, and speed corrections with the pitch control. The pitch command was generated as illustrated in figure 4. The telemetered range signal was differentiated and compared with the desired reference speed of 45 knots to form a range-rate (ground-speed) error signal.

This signal was compared with a filtered pitch-control-position signal to form the pitch-control-command signal used to drive the horizontal needle of the ADI. The filter, in effect, bled off the control input signal as a function of time; thus, this signal was prevented from masking position errors under steady-state conditions. The bleed-off rate used for each control signal is given in table II in terms of the first-order time constant τ.

Figure 5 is a block diagram illustrating the logic used to generate the roll command. As indicated in the diagram, the roll-control-command signal was produced by summing the lateral deviation, lateral-deviation rate, and the filtered roll-control-position signal.

The power-command logic (fig. 6) was identical to the roll-command logic except that the deviation signal (glide-slope deviation) was formed on board the helicopter from the telemetered range and altitude signals.

Moving Map

The moving-map display (a detailed description is given in ref. 8) presented a diagram of the approach path which moved vertically on the viewing screen to indicate range and laterally to indicate lateral deviations. Heading was displayed by a rotating-aircraft symbol. During the present investigation, range was displayed on two scales – namely, 1000 ft/in. (120 m/cm) from 10 000-ft to 2500-ft (3048 m to 762 m) range and 100 ft/in. (12 m/cm) from 2500-ft range to the landing pad. The lateral-map scale was held constant at 100 ft/in.

Altimeter

The altimeter displayed the telemetered altitude signal – that is, the absolute altitude relative to the landing pad. A dual linear scale was employed to provide a sensitive indication below 100 ft (30 m) while still maintaining an overall indication of 0 to 1200 ft (0 to 366 m).

RESULTS AND DISCUSSION

Tracking Performance

The approach performance obtained with the system is indicated by figure 7, where altitude, lateral deviation, and range-rate (ground-speed) error are plotted as a function of range for eight typical approaches. The figure covers approximately the final 60 sec of the approach during which time precision becomes most important. As indicated by the figure, the approach performance was fairly consistent and tended to improve somewhat as the helicopter neared the breakout point.

Figure 8 shows time histories of control activity, flight-director control commands, aircraft motion, range-rate error, and flight-path deviation corresponding to one of the constant-speed approaches presented in figure 7. The time histories are arranged to show, in order, the controlled degree of freedom, the error in the variable being controlled, the control command displayed to the pilot, the pilot's response to the control command, and the aircraft response to the control input.

The pitch and roll control inputs were found to be quite small, particularly during the final portion of the approach. In addition, the pedals were, for all practical purposes, not used; this is probably due to the fact that, with the turn-following system, pedal inputs are used only to produce sideslip and are not required to coordinate turns. An indication of the size of the control inputs used is provided by the data shown in table III wherein the standard deviations and maximum control inputs for the final 60 sec of each approach are tabulated. These data were based on sampling the pilot's control inputs 20 times per sec.

Pilot Acceptance Factors

The pilot commented that the task of tracking the three simultaneous control commands presented on the flight director was quite time consuming and left very little time for scanning the remaining instruments. As a result of this situation, there was always some doubt in his mind as to the actual status of the approach. The factors which caused the tracking task to be so time consuming were brought to light during a follow-on study (ref. 9) in which the pilot was commanded to fly a decelerating approach to a hover. Only the factors which have a direct bearing on the present results are discussed in the following sections.

Display sensitivity.- In order to achieve as high a degree of precision as possible, the gains on the flight-director needles were set relatively high. As a result, the needles were extremely active throughout the entire approach; thus, the pilot was directed to make numerous small control inputs. Although the high needle activity tends to give the pilot the impression that the approach is sloppy, in reality it may be more precise than necessary. It would appear desirable, therefore, to either reduce the gains for the entire approach or make the gains a function of range.

Logic mechanization.- In mechanizing the flight-director logic a linear function between the error in the parameter being tracked (e.g., glide path, localizer, or range rate) and the corresponding needle displacement (control command) on the flight director was employed. This use meant that twice as much error in the tracked parameter would double the needle deflection. Although this appears logical, it tends to overwork the pilot by causing him to fly in a manner which is more precise than he would under visual conditions. To illustrate this point, consider the visual flight condition wherein the pilot does not constrain himself to follow an approach path that is precisely defined at each point in space. Rather, he modulates his approach angle and speed so as to arrive at a particular end point while avoiding any obstacles in his path. Although computer techniques exist which would permit this type of control under instrument conditions, the flight-director logic could become extremely complex. It is thought that a similar easing of pilot workload might be achieved simply by allowing an error to exist within some small range. Thus, if the desired speed is 45 knots, he might be directed to make a disproportionately small control input when the actual speed is between, say, 43 to 47 knots. A similar criterion should probably exist for control and tracking of glide path and localizer so that the pilot would be commanded to fly within a corridor rather than along a line.

Sensor noise.- The principal source of noise on the flight-director display came from the tracking radar system. The velocity input to the flight director was filtered by an amount equivalent to a first-order filter having a time constant of about 1.2 sec which includes filtering provided by the response characteristics of the flight-director instrument itself. Even with this much filtering, the random needle motions were considered objectionable and contributed significantly to the pilot's reluctance to track the needles tightly. In essence, the pilot was forced to act as a final filter.

Scan pattern.- The physical location of the displays created a scan pattern which was too large for the pilot to cover rapidly. Indications are that the displays must be integrated to the extent that the pilot can frequently cross-check his situation information. The cross-check capability is necessary for the following two reasons. First, the pilot must be assured that the flight-director commands are valid. The situation information is his only source for making this judgment since the motion cues which he experiences at low speed have to be accounted for. Second, an adequate cross-check of situation information should ease the pilot workload by allowing him to exercise judgment as to how tightly the flight-director commands should be tracked.

CONCLUDING REMARKS

A flight investigation was conducted to determine the potential instrument-approach capability afforded by a helicopter-type VTOL flight-director system used in conjunction with a relatively sophisticated control augmentation system. Simulated instrument approaches were flown at a constant speed of 45 knots along a straight-in 6° glide path to a 50-ft (15 m) altitude. The results indicate that the pilot can perform acceptable approaches on a consistent basis and uses relatively small control inputs. The flight-director indicator provides a good indication of the necessary control action but, at the same time, requires considerable attention by the pilot to assure himself that he is making the proper control inputs. This situation tends to detract from the pilot's ability to scan the remaining instruments which, in turn, results in his having some doubt as to the overall status of the approach. The tests indicated that modifications to the display sensitivity, logic mechanization, and display integration should tend to alleviate the scanning problem. In addition, a reduction in sensor noise should also result in a lower pilot workload.

Langley Research Center,
National Aeronautics and Space Administration,
Hampton, Va., April 24, 1970.

APPENDIX

MECHANIZATION OF THE CONTROL AUGMENTATION SYSTEM

Introduction

The control augmentation system was mechanized by utilizing the equipment and techniques described in reference 10. As noted in that reference the vehicle response to pilot control inputs is determined primarily by a set of equations of motion (referred to as the models), which describe the desired response in the controllable degrees of freedom. The vehicle response to external disturbances, on the other hand, is determined by the closed-loop-system characteristics which are utilized to match the response of the vehicle to that of the model. Although the models and response matching technique are treated separately in the following discussion, it should be kept in mind that both portions of the system affect the pilot's overall impression of the flying characteristics of the vehicle.

Symbols

a_Y — acceleration along the body Y-axis, feet per second^2 (meters per second^2)

$K_1, K_2, \ldots, K_{10}$ — computer model signal gains

K_a — error signal gain

K_b — integrated error signal gain

p_c — computed rolling angular velocity, radians per second

$\dot{p}_c$ — computed rolling angular acceleration, radians per second^2

q_c — computed pitching angular velocity, radians per second

$\dot{q}_c$ — computed pitching angular acceleration, radians per second^2

q_h — helicopter pitching angular velocity, radians per second

r_c — computed yawing angular velocity, radians per second

$\dot{r}_c$ — computed yawing angular acceleration, radians per second^2

s — Laplacian operator

$\delta_X, \delta_Y, \delta_Z$ — control deflection about body X-, Y-, and Z-axis, respectively, inches (centimeters)

γ — desired approach angle

θ — pitch attitude, radians

τ — time constant, seconds

ϕ — roll attitude, radians

Models

Figure 9 illustrates the computer mechanization of the pitch, roll, and yaw models in block diagram form. The models for the pitch and roll degrees of freedom are identical; for both, the angular acceleration computation involves a summation of the pilot's control input (filtered), an attitude signal obtained from a vertical gyro, and a computed angular velocity signal obtained by integrating the angular acceleration of the model.

In mechanizing the control system, cancellers (devices which nulled the various sensor outputs while the system was disengaged) were incorporated into the attitude command system to eliminate transients during system engagements. The trim relationship between the pilot's control position and the vehicle attitude was therefore determined by the conditions existing at the time the attitude command system was engaged. During the present tests, all engagements occurred in straight-and-level flight at 45 knots with the pilot's control stick in neutral. There were no provisions made to alter the trim-attitude—stick-position relationship other than disengaging the system and then reengaging it with a new trim-attitude—stick-position relationship. Stick force trimming, however, was available at all times.

For the yaw degree of freedom, the computation of yawing angular acceleration involved summing a signal proportional to pedal displacement, a signal proportional to lateral control displacement, a body-mounted lateral accelerometer signal (which is proportional to the helicopter sideward velocity), and a computed angular rate signal.

Response Matching

The technique used to match the response of the helicopter to that of the model is identical to the technique reported in reference 10. As illustrated in figure 10, the technique involves forming a control surface input signal by summing the angular rate error, integrated rate error, and the model angular acceleration. The unstable-rate-gyro feedback loop around the helicopter is employed to reduce the basic helicopter to an angular acceleration system (approximately); this modification permits the use of computed angular acceleration of the model as the lead signal. In reference 10, the helicopter inherent damping was accounted for in the lead network by proper shaping of individual sensor inputs. The present method for forming the lead signal greatly simplifies the computer program.

The level of resistance to external disturbances is primarily a function of the rate-error-signal gain as described in reference 10. For the present test, these gains were $5.5 \frac{\text{rad/sec}^2}{\text{rad/sec}}$, $3.7 \frac{\text{rad/sec}^2}{\text{rad/sec}}$, and $3.0 \frac{\text{rad/sec}^2}{\text{rad/sec}}$ in pitch, roll, and yaw, respectively. Long-term resistance is provided by the integrated-rate-error loop and also the attitude gyro loop which is closed through the model. It should be noted that any disturbance sensed by the attitude gyro will affect both the computed angular acceleration and the computed angular rate of the model. Both of these loops, therefore, will be active in suppressing external disturbances.

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TABLE I.- CONTROL RESPONSE CHARACTERISTICS

Pitch and roll:		
Control power		2 rad/sec^2
Control sensitivity	$0.6\ \frac{\text{rad/sec}^2}{\text{in.}}$	$\left(0.24\ \frac{\text{rad/sec}^2}{\text{cm}}\right)$
Attitude sensitivity	0.15 rad/in.	(0.06 rad/cm)
Damping to inertia ratio		3.0 sec^{-1}
Undamped natural frequency		2.0 rad/sec
Yaw:		
Control power		0.25 rad/sec^2
Control sensitivity	$0.2\ \frac{\text{rad/sec}^2}{\text{in.}}$	$\left(0.08\ \frac{\text{rad/sec}^2}{\text{cm}}\right)$
Directional stability	$0.004\ \frac{\text{rad/sec}^2}{\text{ft/sec}}$	$\left(0.01\ \frac{\text{rad/sec}^2}{\text{m/sec}}\right)$
Damping to inertia ratio		0.7 sec^{-1}
Yaw due to lateral control	$0.065\ \frac{\text{rad/sec}^2}{\text{in.}}$	$\left(0.03\ \frac{\text{rad/sec}^2}{\text{cm}}\right)$
Height control characteristics:		
Thrust-to-weight ratio		>1.1
Height control sensitivity	0.15 g/in.	(0.06 g/cm)
Normal velocity damping to mass ratio		0.38 sec^{-1}

TABLE II.- FLIGHT-DIRECTOR GAINS

Command	Input	[a]Full-scale signal	Sensing
Pitch	Range-rate (ground-speed) error	±25 ft/sec (±7.62 m/sec)	Up deflection commands aft stick
	Pitch control position [b]τ = 10 sec	±2 in. (±5.08 cm)	
Roll	Lateral deviation	±250 ft (±76.2 m)	Right deflection commands right stick
	Lateral-deviation rate	±25 ft/sec (±7.62 m/sec)	
	Roll control position [b]τ = 10 sec	±2 in. (±5.08 cm)	
Power	Glide-slope deviation	±150 ft (±45.72 m)	Up deflection commands an increase in power
	Glide-slope-deviation rate	±15 ft/sec (±4.57 m/sec)	
	Power control position [b]τ = 2 sec	±1 in. (±2.54 cm)	

[a]Full-scale deflections of the pitch, roll, and power command needles are ±1.0 in. (±2.54 cm), ±1.0 in. (±2.54 cm), and ±0.85 in. (±2.16 cm), respectively.

[b]τ is the first-order time constant defining the rate at which these signals are bled off.

TABLE III.- CONTROL USAGE DURING THE FINAL 60 SECONDS OF THE APPROACH

Run	Longitudinal control				Lateral control				Rudder pedals				Collective control			
	Maximum		Standard deviation		Maximum		Standard deviation		Maximum		Standard deviation		Maximum		Standard deviation	
	in.	cm	in.	cm	in.	cm	in.	cm	in.	cm	in.	cm	in.	cm	in.	cm
1	0.3	0.76	0.1	0.25	0.5	1.27	0.1	0.25	0.1	0.25	0	0	0.9	2.29	0.6	1.52
2	.4	1.02	.2	.51	.6	1.52	.2	.51	.0	.0	0	0	1.6	4.06	.6	1.52
3	.4	1.02	.1	.25	.5	1.27	.1	.25	.0	.0	0	0	.6	1.52	.3	.76
4	.4	1.02	.2	.51	.6	1.52	.2	.51	.0	.0	0	0	1.2	3.05	.7	1.78
5	.4	1.02	.1	.25	.5	1.27	.1	.25	.0	.0	0	0	1.2	3.05	.6	1.52
6	.7	1.78	.2	.51	.5	1.27	.2	.51	.1	.25	0	0	.7	1.78	.3	.76
7	.3	.76	.1	.25	.4	1.02	.1	.25	.1	.25	0	0	.6	1.52	.2	.51
8	.5	1.27	.2	.51	.7	1.78	.2	.51	.3	.76	0	0	.6	1.52	.2	.51

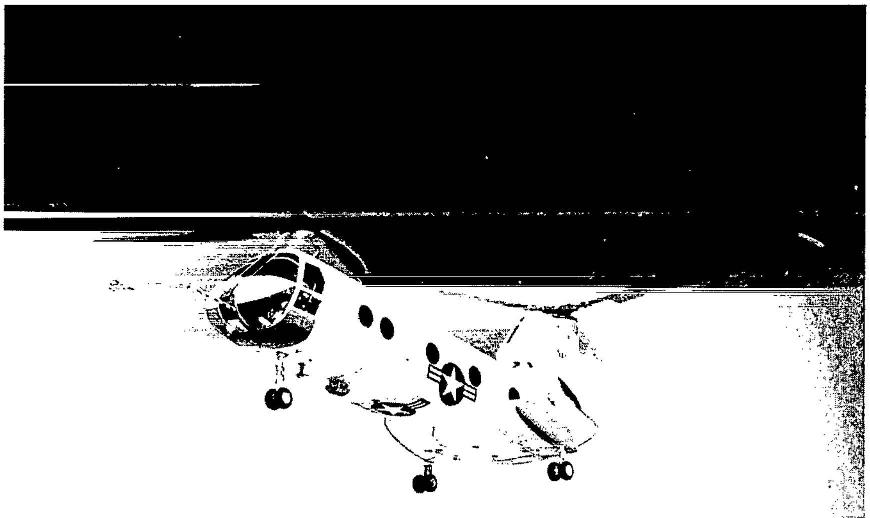

Figure 1.- Test helicopter. L-68-9364

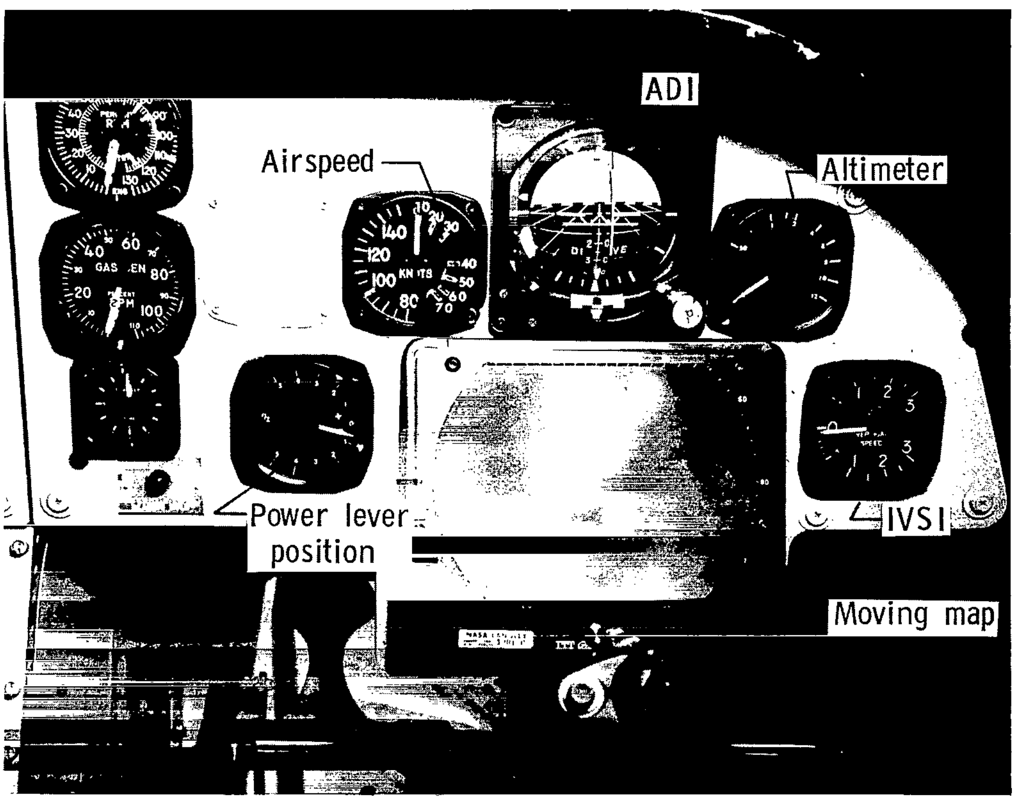

Figure 2.- Test instrument display. L-69-3332.1

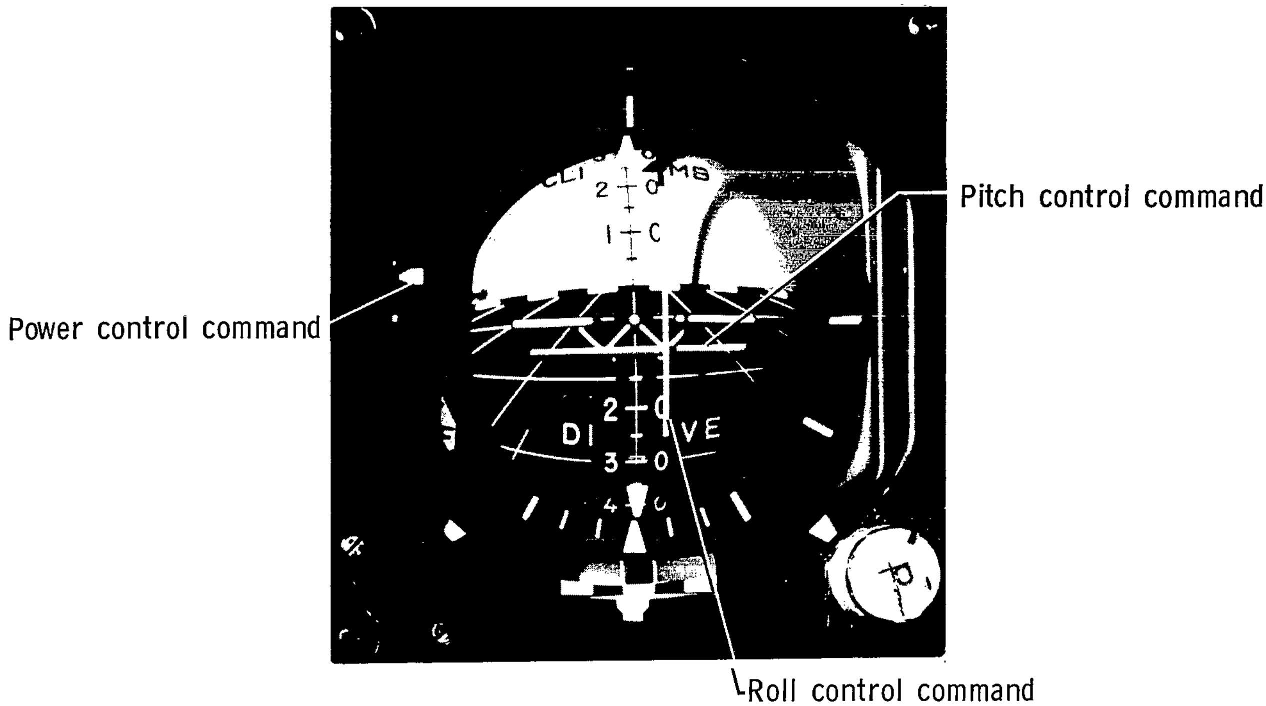

Figure 3.- ADI control commands. L-70-1623

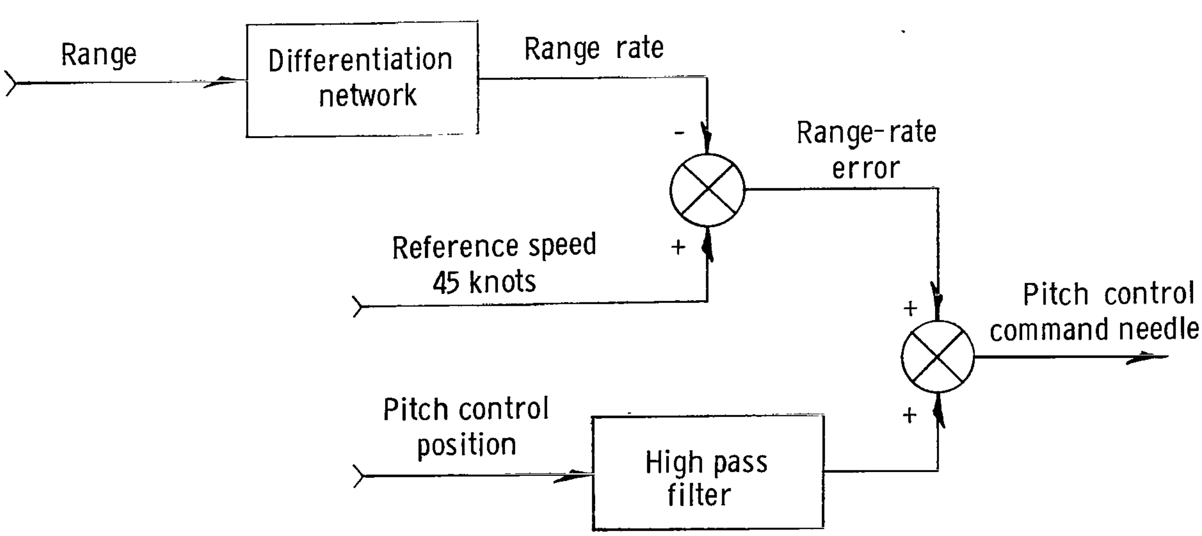

Figure 4.- Pitch-command logic.

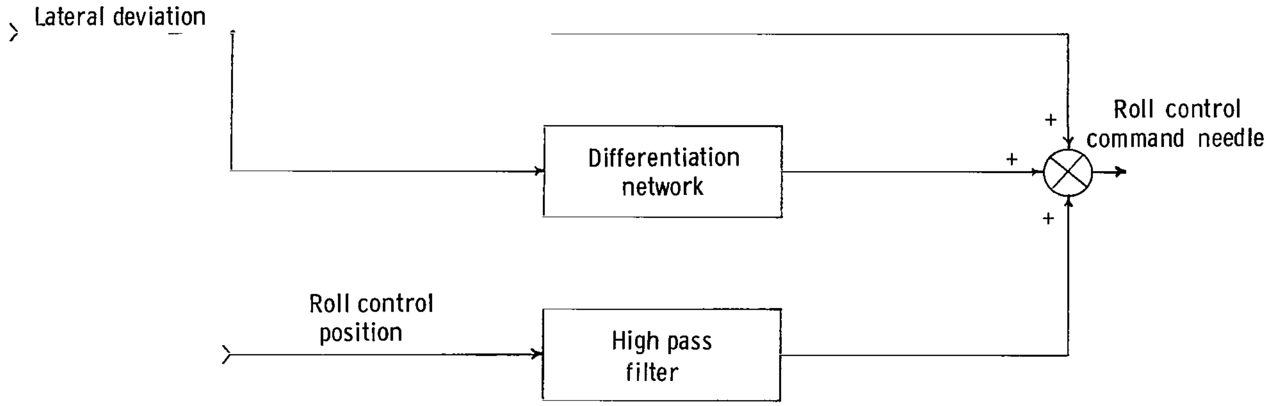

Figure 5.- Roll-command logic.

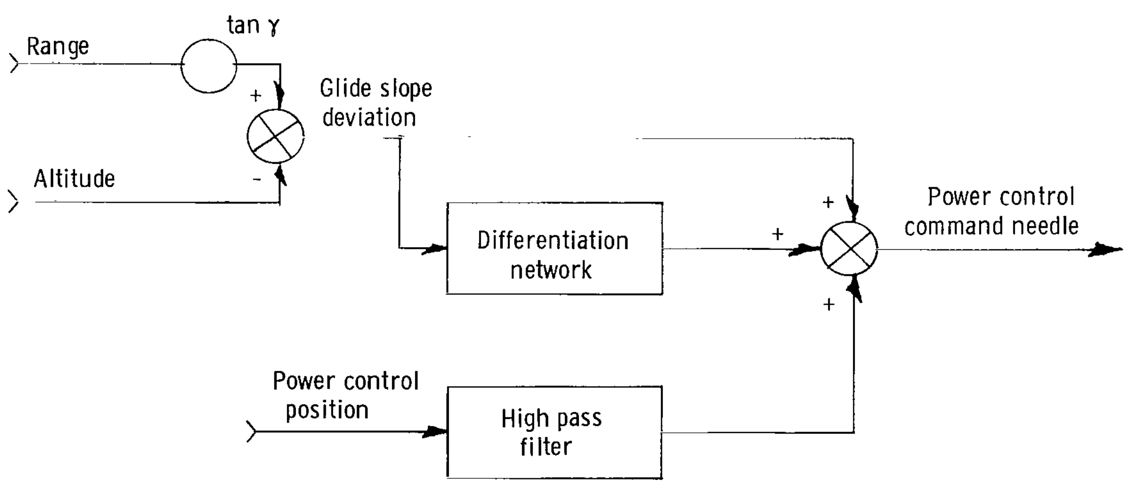

Figure 6.- Power-command logic.

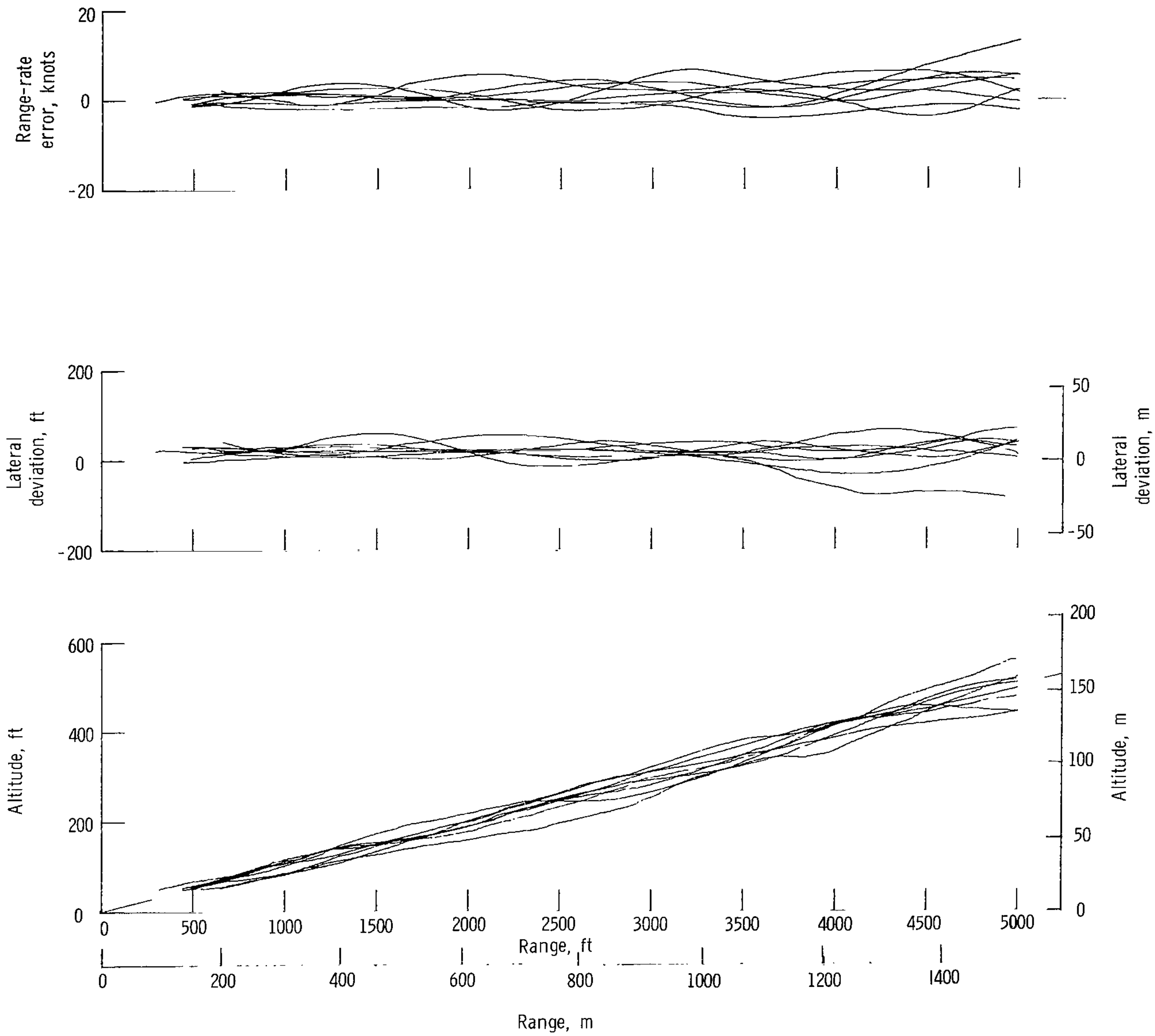

Figure 7.- Approach performance.

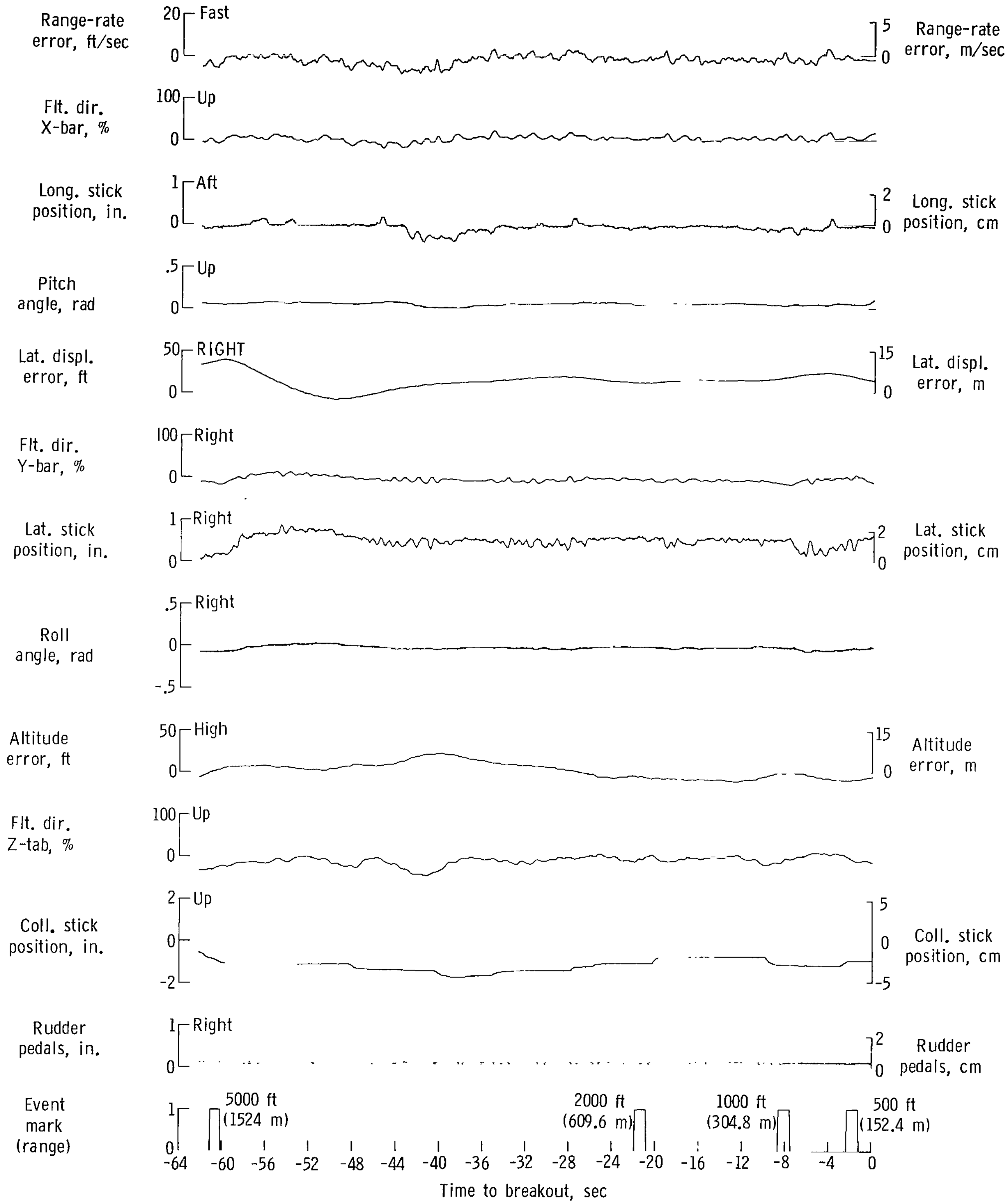

Figure 8.- Time histories of typical approach.

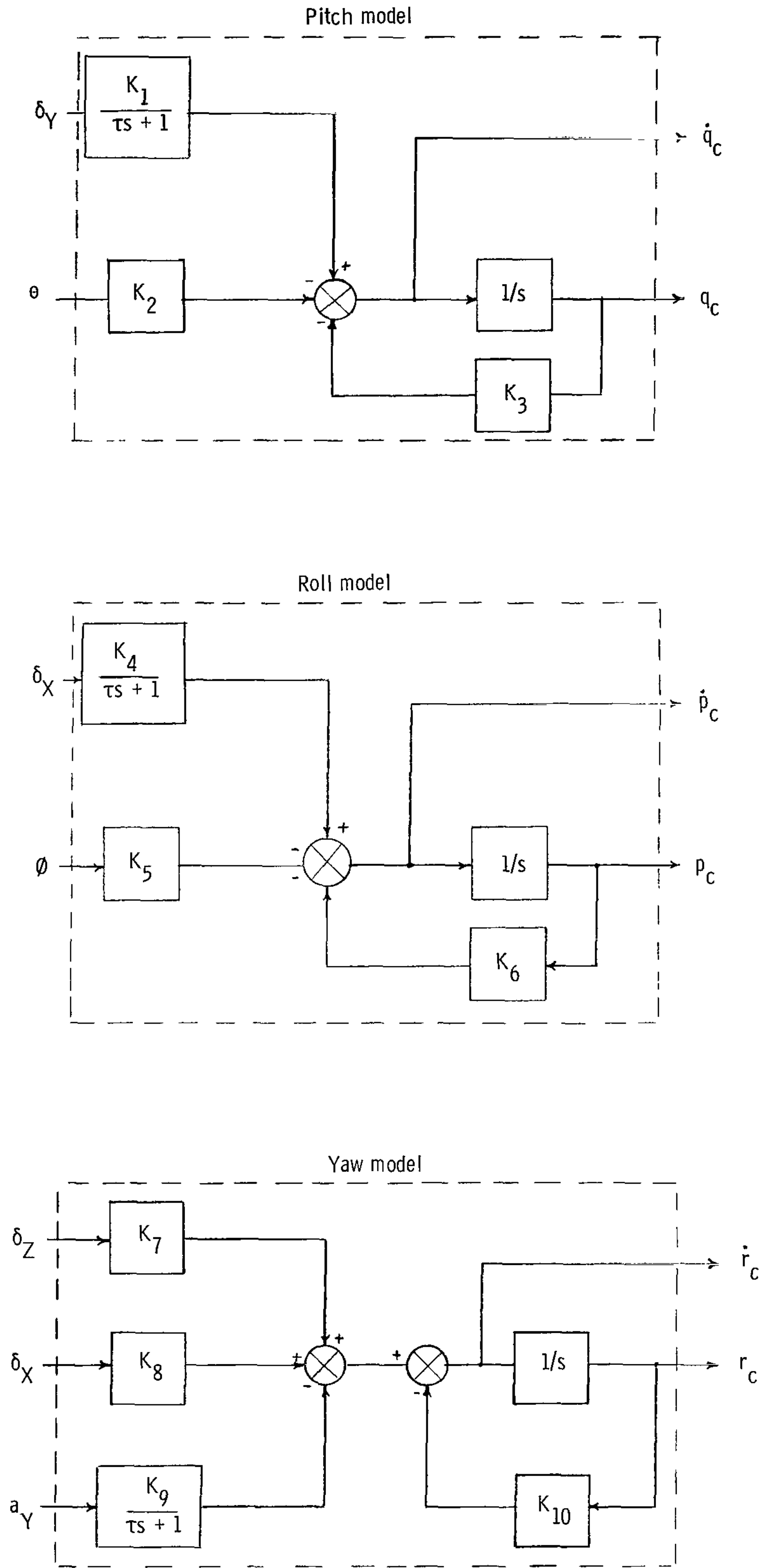

Figure 9.- Computer mechanization of the pitch, roll, and yaw degrees of freedom.

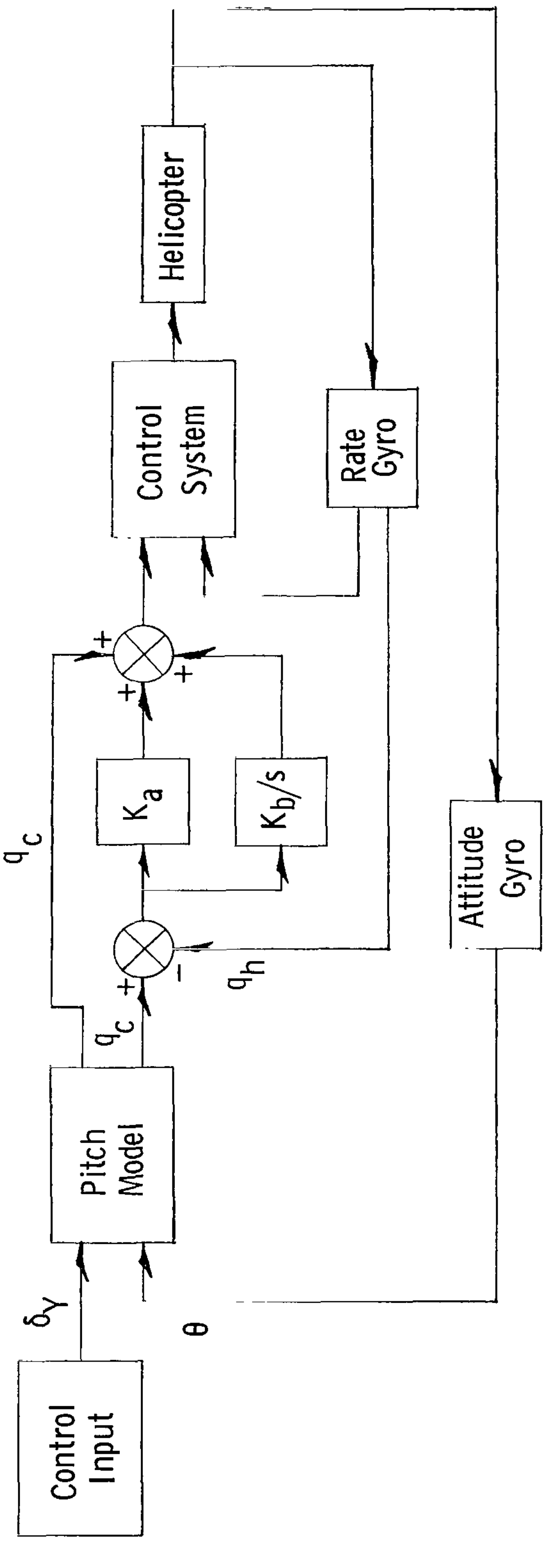

Figure 10.- Technique used for matching aircraft to model response (pitch axis shown).

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