APPLICATION OF THE MODEL TECHNIQUE TO A VARIABLE-STABILITY HELICOPTER FOR SIMULATION OF VTOL HANDLING QUALITIES

By John F. Garren, Jr., and James R. Kelly

NASA Langley Research Center
Langley Station, Hampton, Va., U.S.A.

Presented at the
27th Meeting of the AGARD Flight Mechanics Panel

Rome, Italy
October 11-12, 1965

NATIONAL AERONAUTICS AND
SPACE ADMINISTRATION
WASHINGTON

APPLICATION OF THE MODEL TECHNIQUE TO A VARIABLE-STABILITY HELICOPTER FOR SIMULATION OF VTOL HANDLING QUALITIES

By John F. Garren, Jr.,* and James R. Kelly*

NASA Langley Research Center

SUMMARY

34999

In order to provide a means for accurate in-flight simulation of V/STOL aircraft, a computer model technique has been adapted to a variable-stability helicopter. Unlike the stability-derivative simulation technique, which is usually employed in variable-stability aircraft, the model approach produces a response which is essentially independent of the dynamics of the test vehicle. The aircraft response, therefore, is a function only of the evaluation pilot's control inputs and the dynamics which are programed into the analog computing equipment.

In-flight time histories of the helicopter response and the corresponding commanded response are presented to illustrate the effectiveness of the technique. The results indicate that the model technique does, in fact, provide a feasible, accurate, and flexible approach to in-flight simulation. Author

INTRODUCTION

The critical need for improving the validity and coverage of existing VTOL handling-qualities criteria is well recognized. In the absence of VTOL aircraft suitable for conducting the required studies, the bulk of handling-qualities data must come from simulation. It is essential, moreover, that the particularly critical areas be explored by means of airborne simulation because of intangible influences of the pilot environment and flight task. In the past, however, airborne simulation has been hampered by an inability to represent a wide range of characteristics with a sufficient degree of accuracy. The inaccuracies have stemmed from a lack of information relative to the characteristics of the basic vehicle, as well as from the complexity involved in altering some of the more important characteristics as discussed in reference 1.

The miniaturization of analog computing equipment during recent years has made it possible to circumvent these problems by applying ground-based simulation techniques to variable-stability aircraft. When applied to airborne simulation this method is commonly referred to as the model simulation technique. In this technique the equations of motion, which represent the aircraft

*Aerospace Technologist.

characteristics being simulated, are programed into an onboard computer. In the computer, the desired response is generated in accordance with the programed equations on the basis of control inputs and motion-sensor feedbacks. Differences in the desired response and the vehicle's actual response are used to form an error signal which drives the control surfaces so as to eliminate the discrepancy.

The modification of a prototype helicopter for adaptation of the computer-model simulation technique was initiated by the NASA in 1961. Following development and documentation of the simulation capability in 1962, actual research flights began in 1963. The purpose of this paper is to present a description of this technique as it has been applied to low-speed VTOL research. Limitations of this technique encountered under operating conditions are discussed. In-flight time histories are presented to illustrate the effectiveness of the model technique. A general description of the variable-stability helicopter is also included.

LIST OF SYMBOLS

X,Y,Z	forces along the respective body axes, lb
L,M,N	moments about the body X, Y, and Z axes, respectively, lb-ft
I_X, I_Y, I_Z	moments of inertia about the respective body axes, slug-ft^2
m	mass of aircraft, slug
c	used as subscript to designate commanded response (i.e., the computer response)
h	used as subscript to designate actual response (i.e., the response of the test helicopter)
$\delta_X, \delta_Y, \delta_Z, \delta_P$	control deflection about the three body axes indicated and along the body Z axis, respectively, in.
p	rolling angular velocity, rad/sec
q	pitching angular velocity, rad/sec
r	yawing angular velocity, rad/sec
u	forward component of velocity, ft/sec
v	side component of velocity, ft/sec
w	normal component of velocity, ft/sec
α	angle of attack, rad

$\gamma_1,\gamma_2,\gamma_3,\gamma_4$ components of simulated gust velocity, ft/sec

z absolute altitude, ft

Unless otherwise indicated, when any of the defined symbols are used as subscripts, the derivative with respect to that parameter is indicated. Dots over a symbol indicate a time derivative with respect to that parameter.

GENERAL SIMULATION CONSIDERATIONS

Equations of Motion

The terms which are of primary interest because of their first-order effects on the response of VTOL aircraft are shown below in the form of linear, quasi-static equations of motion:

$$\dot{q} = \frac{M_{\delta_Y}}{I_Y}\delta_Y + \frac{M_q}{I_Y}q + \frac{M_\alpha}{I_Y}\alpha + \frac{M_u}{I_Y}u + \frac{M_{\delta_P}}{I_Y}\delta_P + \frac{M_\gamma}{I_Y}\gamma_1 \tag{1}$$

$$\dot{p} = \frac{L_{\delta_X}}{I_X}\delta_X + \frac{L_p}{I_X}p + \frac{L_v}{I_X}v + \frac{L_r}{I_X}r + \frac{L_\gamma}{I_X}\gamma_2 \tag{2}$$

$$\dot{r} = \frac{N_{\delta_Z}}{I_Z}\delta_Z + \frac{N_r}{I_Z}r + \frac{N_v}{I_Z}v + \frac{N_{\delta_X}}{I_Z}\delta_X + \frac{N_\gamma}{I_Z}\gamma_3 \tag{3}$$

$$\dot{w} = uq + f(\theta,\varphi) + \frac{Z_{\delta_P}}{m}\delta_P + \frac{Z_w}{m}w + \frac{Z_z}{m}z + \frac{Z_\gamma}{m}\gamma_4 \tag{4}$$

$$\dot{v} = ur + f(\varphi) + \frac{Y_v}{m}v \tag{5}$$

$$\dot{u} = vr + f(\theta) + \frac{X_u}{m}u \tag{6}$$

The coefficients in the equations need not be constant since nonlinear elements are available in the computing equipment to permit variation in the coefficients with airspeed and altitude or other parameters. Similarly, varied relationships between control position and the corresponding accelerating moment can be handled.

Because in the aircraft there are only four independent sources for producing moments and forces (moments about each of the three body axes and a lift force or thrust along the body -Z axis), it is possible to alter the basic vehicle response for only the degrees of freedom expressed by the first four equations. In the absence of independent sources for producing forces along the body -X and -Y axes (which would necessitate installation of additional propulsion systems) it is necessary to live with the characteristics of the basic aircraft for the latter two degrees of freedom, equations (5) and (6). It may be seen from inspection of the latter two equations, however, that only the last term of each, the drag, is a function of the particular aircraft; the other terms are independent of aircraft configuration so that the basic vehicle is otherwise inherently correct. Fortunately, therefore, the inability to alter the response for these two degrees of freedom represents only a minor limitation, particularly at low speeds where drag effects are usually quite small.

From the pilot viewpoint, not duplicating the drag term results in a deviation from the correct relationship between aircraft attitude and the steady-state linear velocity. Strictly speaking, there can also exist slight differences in the long-term response following control inputs, but such differences are so small as not to be perceived by the pilot in the majority of cases. For example, if one wished to simulate a tilt-wing VTOL through the conversion from hover to cruise flight, the fuselage attitude would not be correctly duplicated even though the dynamics and response to control inputs would be essentially correct. From a handling-qualities standpoint such effects are probably minor in comparison with the other parameters which are being studied and, therefore, do not currently justify the increase in complexity which would be associated with adding sideward and forward (and rearward) thrusting engines.

Mechanization of Equations

The solutions for the first three equations of motion as listed previously are desired in terms of angular velocities about the respective axes; from the fourth equation, the normal acceleration is required. These solutions are obtained in real time and are the responses which are used to command the vehicle motion. In order to obtain the solution for the first three equations, the outputs from the pilot controls and from various motion sensors are summed in accordance with the specified equations of motions to produce a voltage proportional to the desired angular acceleration which is integrated in turn, to yield a signal proportional to the desired angular velocity. For illustrative purposes the mechanization of the lateral-directional equations of motion, equations (2) and (3), is shown schematically in figure 1. It should be noted that the coefficient values, which are set on the potentiometers, have been normalized with respect to aircraft inertias so that the effect of inertia on the periods and time constants is automatically taken into account. The method of mechanization shown in the figure also accounts for the proper degree of coupling, or interaction, between the axes.

A further example of the mechanization which is required is the switch-over from air-referenced to ground-referenced signals at speeds below about 30 K where some sensors, such as the sideslip vane, become unreliable. The switch to provide this function is shown in the schematic diagram. It is pointed out, moreover, that for speeds above 30 K, the signal proportional to the aircraft sideward velocity v is generated by passing the signal from the angle-of-sideslip vane through a potentiometer which is set for the intended forward speed. For test conditions where the speed will not be held constant, the potentiometer can be replaced by an element which accounts for changes in velocity. Variation of the coefficients with speed can be handled in much the same manner.

DESCRIPTION OF EQUIPMENT

Test Vehicle

A photograph of the NASA variable-stability helicopter is shown in figure 2. The gross weight of this vehicle is 15,500 pounds and its maximum speed is 140 knots. The vehicle has been demonstrated to rearward and sideward velocities up to 25 knots and to a normal acceleration of 1.5g. It should be noted that this aircraft is a prototype and these restrictions are not typical of production models. A more complete documentation of the characteristics of the basic aircraft is given in reference 2.

As in the case of any simulator, the maximum accelerations and velocities which can be simulated are limited to the corresponding capabilities of the test vehicle. These limitations for the angular degrees of freedom for the NASA test helicopter are given below.

Axis	Maximum angular acceleration, rad/sec^2	Maximum angular velocity, rad/sec
Pitch	1.7	3.3
Roll	1.3	1.8
Yaw	.25	>2.0

The angular acceleration in yaw is considered marginal although its near-zero angular velocity damping permits extremely high angular velocities to be developed. At any rate, for the simulation of dynamics pertinent to aircraft as large as, or larger than, the test helicopter, these acceleration and rate capabilities have generally proven to be adequate. Difficulties are sometimes encountered, however, in the simulation of higher response associated with small aircraft.

Variable-Stability System

The variable-stability system (ref. 3) installed in the helicopter is composed of three major components; namely, a modified control system, an analog computer, and a sensor group. The location of each component in the overall system is illustrated in figure 3. The block in the figure labeled "signal plugboard" provides the interface between the various components. Although the function of the latter two components has been discussed previously, additional information is included in this section.

Modified control system.- The pilot controls on the right-hand side of the cockpit, consisting of the conventional center stick, rudder pedals, and collective stick were modified to a "fly-by-wire" system; that is, the motion of these controls produces only electrical voltages which can, in turn, be used to drive the control surfaces. The left-hand controls are unmodified and are continuously monitored by the safety pilot whose duty it is to take over in the event of a malfunction or other emergency. Further modification included provision for the conversion of electrical voltages to control surface displacements. There are four separate but identical channels in the system - one each for the pitch, roll, yaw, and vertical degrees of freedom.

Since fail-safe features were not designed into the various components of the variable-stability system, several safety provisions were incorporated in the modified control system. These provisions included a control-limiting system, a safety pilot override feature, and several disengage modes. The control-limiting system permits a variation in the authority of the variable-stability system from 10 percent to 100 percent. For example, when the authority is set at 10 percent, the system is capable of commanding only 10 percent of the total control surface travel. Although the simulations are normally run at 100 percent authority, the initial engagement on each flight is made at a reduced authority (about 30 percent). The disengage modes, which revert control of the aircraft to the safety pilot in the event of an emergency, include electrical disengage buttons on each of the pilots' controls as well as a redundant mechanical switch on the instrument panel. As a precaution against hardover failures, the override feature allows the safety pilot to overpower commands by the variable-stability system without having to first disengage the system.

Analog computer.- The computing equipment which is located in the cabin is shown in figure 4 along with the signal plugboard. This computing equipment consists of two off-the-shelf computers, which are slaved so that both may be operated from a single control panel. The equipment, as shown, is sufficient for programing three degrees of freedom, but is being expanded to handle the fourth. The computing elements currently installed in the NASA helicopter include forty (40) operational amplifiers, sixteen (16) integrators, forty-eight (48) potentiometers, and twenty-four (24) nonlinear components. In addition to computing the command response, several complementary functions are performed. Some elements are used in establishing quasi-static conditions at the instant of engagement. This function, commonly referred to as canceling, starts the outputs from all the sensors at zero, which also prevents transients upon engagement of the system. Still another function served by the computer is the correction of various motion sensors for their location relative to the aircraft center of gravity.

Sensors.- Insofar as possible, the locations of the various sensors were chosen with regard to their respective function. The angular velocity and angular acceleration sensors may be mounted in any convenient location so that their position was selected on the basis of minimum vibration. Most other sensors, on the other hand, are sensitive to location with respect to the aircraft center of gravity; and, where a center-of-gravity location is not feasible, corrections to their outputs must be made. For example, the angle-of-attack sensor must be mounted ahead of the aircraft to minimize rotor downwash effects. In this position, however, the vane is sensitive to pitching angular velocity. By properly summing the output from the vane with the output from the longitudinal rate gyro, however, the true angle of attack is obtained. Similarly, corrections are required for the angle-of-sideslip vane and for the linear accelerometers. In addition to contributing to the solution of the command response, the sensor outputs are recorded for correlation with the pilot ratings and comments, and, in some cases, they are used to actuate cockpit displays.

RESPONSE COMMAND METHOD

The method used for generating a signal proportional to the desired, or commanded response was discussed in a preceding section. This section describes the technique which forces the vehicle to obey the commanded response. The basic command technique, as illustrated in figure 5, is nothing more than a closed-loop servomechanism employing a rate error signal. In this case the computer provides the commanded angular rate q_c and the feedback, or actual rate, q_h is provided by a rate gyro which is mounted in the aircraft.

Limitation of the Basic Command Method

Effect of time delays.- Inspection of simultaneous time histories of the commanded and the actual response for systems of the type shown in figure 5 generally indicate a time lag, or phase shift between the two, even though the time histories might appear essentially identical in other respects. Any such delays are, of course, undesirable and if they appraoch 0.2 second or so, they tend to become discernible to the pilot as a delay in the response to control inputs. A more subtle effect of the time delay than the mere phase shift between the commanded and the actual response is the error which it produces in the commanded response itself.

There are then, in fact, three distinct responses which should ideally be identical - the theoretical response, the commanded response, and the actual response. Although one is tempted to assume that the commanded response is automatically the same as the theoretical response, this is strictly true only for simple dynamics (for example, zero-order and first-order responses) where the solution of the equations of motion does not involve the motion sensors. A zero-order response results from an acceleration system in which case the only input to the summing amplifier in figure 1 would be the output from the pilot control. Similarly, no motion sensors are used in computing a first-order response, a rate system, in which case there would be two inputs to the summing amplifier - the pilot control and a rate feedback from the computed angular velocity. For most other instances the motion sensors are required for solution of the commanded response and time delays inherent in the basic command method are fed back through the sensors which cause the commanded response to deviate from the theoretical response. This, in turn, causes the actual response, which closely duplicates the commanded response, to be in error. The error, thence, tends to be self-generating, so that to accurately duplicate the theoretical response for long intervals of time, say 30 seconds, would require that the vehicle-following time delay not exceed a few hundredths of a second. In general, high-frequency responses having low damping ratios are the most adversely affected by the time delay.

For the evaluation of handling qualities during precision tasks, such as instrument approaches and hovering over a spot in turbulence where the pilot control frequency is on the order of one input per second, exact duplication of the theoretical response for long periods of time is not mandatory. In such cases, where the pilot may be considered as an active element in the control loop, the interval of prime importance, as shown by several handling-qualities investigations during recent years, is the first 2 or 3 seconds of the response following the control input. It does not appear unreasonable, however, to require that the system have an accuracy of no less than 80 percent during the first 10 seconds following a disturbance, either by the pilot or by the simulated turbulence.

Source of time delays.- Since the reduction of time delays is the key to achieving an accurate response, it is necessary to understand their source. One source is the dynamics of the basic test vehicle which typically contributes time lags on the order of 0.1 second. The second source is directly dependent on the error-signal gain (amplification of the error signal) which can be attained. The error-signal gain is defined here as the angular acceleration which the helicopter develops to cancel a unit error in angular velocity and is defined as: $G = \frac{\dot{q}_h}{q_c - q_h}$. It is noted, therefore, that G has units of $\frac{1}{\text{second}}$ and, in the absence of control system time delays, can be considered as approximately representing the reciprocal of the time required for the vehicle to achieve 63 percent of any commanded rate. For example, assuming a static gain of 10/sec for G, the actual response will lag the commanded response by 0.1 second. It should be pointed out that the time delays from each of the sources mentioned are not directly additive since the error signal overcontrols in an effort to reduce the basic vehicle time delay.

Since the time delay associated with the closed-loop dynamics is the reciprocal of the error signal gain, it would be desirable to attain an infinite gain. As in the case of any practical system, the characteristics of the various loop-elements limit the maximum allowable gain to some finite value, beyond which the control system will limit cycle, i.e., a self-sustained oscillation of high frequency and constant amplitude will exist in the control loop. Gains on the error signal which were attainable in flight using the basic system of figure 5 resulted in time delays, the worst of which was about 0.3 second while the best was somewhat less than 0.1 second. Analytical studies based on this worst case indicated that a wide range of damping ratio values, including a damping ratio of zero, could be satisfactorily simulated only for periods greater than 7 seconds. Even at low speeds it is possible for small aircraft, i.e., aircraft with low moments of inertia, to exhibit periods somewhat shorter than this. It was apparent therefore that some modification to the basic technique was necessary.

Modification of Basic Command Technique

Lead.- Aside from the long-term inaccuracies which tend to accumulate when there is an inadequate gain on the rate error signal, the lag produced in the initial or transient response following control inputs is no doubt the most adverse effect from a handling-qualities standpoint. Time delays on the order of 0.3 second are within the pilot capability of observation and would therefore result in pessimistic pilot ratings for the simulated characteristics. In order to overcome this delay, a lead network, as shown in figure 6, was added to the basic technique. This input is scaled so as to produce the correct initial acceleration following motion of the control.

Additional lead can be provided to further reduce the lag in the actual response by feeding the motion sensor outputs into the control system through the lead network. The inputs from the sensors are scaled so that the stability derivatives of the basic test vehicle are artifically altered to match the stability derivatives being simulated. Such use of lead is the sole simulation method used in conventional airborne simulators, so that the present simulation technique is in actuality a hybrid of the pure model technique and the conventional or stability derivative technique.

Inasmuch as the use of lead in this application, however, is only a second-order refinement, the matching of the basic vehicle characteristics to the desired characteristics need be only approximate to yield the desired result. Assume, for example, that the characteristics of the basic test vehicle are only approximately known so that the best simulation which can be achieved using only lead is about 60 percent. Assume, further, that the basic, or unmodified model simulation technique is capable of compensating for 80 percent of the difference between the desired response and the inherent response of the basic test vehicle for some range of characteristics. By combining the two methods the expected accuracy can be figured approximately as 60 percent plus 0.8 (40 percent), or 92 percent. In other words, even by adding compensation that is uncertain by a margin of 40 percent, the error in the simulation is reduced from 20 percent to 8 percent. It is seen, therefore, that the lead need not be precise to be effective.

Integrator loop.- There are many characteristics in the basic test vehicle which contribute minor errors to the simulated response, but which cannot be compensated for by using lead (partly because of a lack of appropriate sensors and partly because of the added complexity). Trim changes of the basic vehicle and inertia coupling effects are representative examples of such characteristics. An effective compensation network for this purpose, which was suggested by National Research Council personnel who also use it in their variable-stability helicopter, is the integrator network shown in figure 6. As the name implies, this network integrates any error in the angular velocity and therewith feeds in additional control.

As in the case of the gain on the rate error signal, only a limited gain can be tolerated on the integrated-rate error signal. In fact, the addition of the integrated signal represents a compromise in that it becomes necessary to reduce the rate-error-gain since the overall error gain is related to the square root of the sum of the squares of these individual gains. Furthermore, the overall gain which can be achieved is somewhat lessened because of the phase characteristics introduced by the integrator loop. Addition of the integrator loop provides, nonetheless, a net improvement in the overall response, particularly for the low-frequency response. Also, with regard to elimination of external disturbances, the integrator provides a long-term attitude memory whereas the pure rate error signal provides only high damping.

In general, then, the lead improves the high-frequency response; the integrator loop, the low-frequency response; and the basic rate error signal operates over the entire spectrum.

Results Using Modified Techniques

Figure 7 is a flight time history obtained using the modified technique. The trace labeled "q_c" is the commanded pitching angular velocity and the one labeled "q_h" is the actual angular velocity. Both traces are recorded on approximately the same gain. The timing marks, the vertical lines, are at 1-second intervals. An estimate of time delay in the vehicle-following may be obtained by comparing the time between corresponding peaks. A careful inspection of this figure reveals that the overall time delay does not exceed 0.1 second.

As discussed in an earlier section, the basic technique was substantially limited in the range of characteristics which could be accurately simulated because of an appreciable time delay resulting from limitations on the error signal gain. The modified technique, on the other hand, provides a capability for simulating oscillatory responses with periods as short as 4 seconds at very low damping ratios, and even shorter periods at higher damping ratios. The primary obstacle to further increasing the range of response which can be simulated is the time delay associated with the basic vehicle. The present capability, however, has thus far proven adequate for the simulation of characteristics pertinent to low-speed handling qualities for VTOL aircraft.

An additional criterion against which the model simulation technique was judged was its ability to eliminate external, or unprogramed, disturbances such as trim changes of the basic test helicopter. As is the case with other tandem-rotor helicopters, the test helicopter exhibits a strong tail-sitting tendency during decelerating flares, which must be countered by forward motion of the longitudinal control. In order to test the system capability in this respect, flight records of the evaluation pilot's control and of the helicopter control surfaces were obtained as the evaluation pilot flew the aircraft through the critical maneuver. These control-motion time histories are compared in figure 8. This figure illustrates clearly that the evaluation pilot's control remained trimmed near zero throughout the maneuver, despite the fact that the control system of the basic helicopter moved about 25 percent of its total travel to fight the trim change. Total travel is used here as the distance between the control travel limits. It was concluded from such tests that the model technique effectively eliminates any unprogramed disturbances.

APPLICATION OF THE MODEL SIMULATION TECHNIQUE

As implied in the introduction, the principal use of the NASA variable-stability helicopter is in the development of general criteria for VTOL handling qualities. For this purpose, rather than simulating the detailed characteristics of specific configurations, a wide range of different parameters are systematically evaluated for a variety of tasks. The merits of the model simulation technique for such studies are best emphasized by a discussion of a few of the recent contributions to handling qualities which it has made possible.

The problem of applying helicopter criteria to VTOL aircraft is particularly critical in the specification of control power, which, though relatively inexpensive in helicopters, must be provided at the direct expense of installed power in many VTOL configurations. It is thus important that minimum requirements be accurately determined. Since one of the fundamental requirements for control power is maneuvering, an extensive investigation (ref. 4) of maneuvering requirements was conducted during which trim changes and disturbances were eliminated by use of the model technique. Although differences in trim change characteristics and gust susceptibility have been the principal criticism of applying helicopter experience to VTOL aircraft, the results of this study showed close agreement with AGARD Report 408 (ref. 5) which was based largely on helicopter experience. The maximum difference for any of the axes was only 20 percent; the general validity of this portion of current criteria is thus more firmly established.

Precision tasks, such as hovering over a spot and square hovering patterns, were also performed during the control power investigation and it was soon apparent that wide variations in either control power or sensitivity (angular acceleration per unit control) had little, if any, effect on the pilot ratings. In the absence of disturbances, the visual precision tasks became trivial even for low values of damping, and the aircraft could be "balanced" with very little pilot effort. The overwhelming effect of disturbances on the precision hovering task is clearly evident by comparison of these results with previous studies where disturbances, could not be eliminated. The ability to isolate the various parameters and to examine their effects individually has contributed to a clearer understanding of the overall handling-qualities picture.

The effects of trim change, static stability, and simulated turbulence on these results will be the subject of future investigations. In the case of the yaw axis, the effects of static directional stability have already been examined with this equipment and are reported in reference 6.

A flight investigation conducted with an on-off type control using this equipment provides another example of the potential offered by the model technique. This investigation was conducted to determine the feasibility of using an on-off control system (as opposed to the conventional proportional system) for V/STOL operation. The parameters which were investigated included the size of the control deadband, control power effects, and angular velocity damping effects. During these tests extremely low control powers were investigated and were found to give satisfactory maneuver capability. In fact, satisfactory control for maneuvering was obtained at about one-quarter the control power needed for proportional control systems. Total moments equivalent to less than 3 percent of the total available control-surface travel of the basic aircraft were explored (values this low were not satisfactory, however). Since the basic aircraft exhibits trim changes on the order of 25 percent of its total control-surface displacement, as discussed previously, this study could not have been accomplished had not the simulation technique been capable of eliminating trim changes.

Although the equipment, as installed in this aircraft, is best suited for establishing general handling-qualities criteria, such an aircraft can be used effectively as a training device to better acquaint test pilots with the effects of various stability derivatives on flying qualities. Also, by simulating aircraft that are still in the design stage, it often is possible to detect potential problem areas and to determine the direction and magnitude of changes to correct the deficiency. As still another application, it is always advantageous to simulate the characteristics of newly built and yet unflown aircraft so that the pilot can gain experience with new or unusual characteristics in the presence of a safety pilot who can revert to normal characteristics. In such an application, the aircraft equations of motion and stability characteristics, which are estimated by theoretical calculations and wind-tunnel testing, are programed into the onboard computers. If the desired degree of detail requires variation in the characteristics as a function, say, of airspeed, function generators in the computer make this possible.

CONCLUDING REMARKS

A model-controlled simulation technique has been adapted to a relatively sophisticated variable-stability helicopter for study of low-speed handling-qualities requirements. The ability of the technique to wash out the stability of the basic helicopter and thus to command the computed response has been demonstrated from analysis of flight time histories. Some lag problems and steady-state errors were encountered because of limitations on the maximum error-signal gain which could be achieved. These problems were largely overcome, however, by introduction of lead networks which produce the correct initial response following control inputs and by an integrator network on the rate error signal which reduces long-term errors. The results indicate that the model technique does, in fact, provide a feasible, accurate, and flexible approach to in-flight simulation.

REFERENCES

1. Gould, D. G.: The Model-Controlled Method for Development of Variable-Stability Aircraft. AGARD Report 402, July 1962.

2. Kelly, J. R.; and Winston, M. M.: Stability Characteristics of a Tandem-Rotor Transport Helicopter as Determined by Flight Test. NASA TN D-2847, 1965.

3. Garren, John F.; and Kelly, J. R.: Description of an Analog Computer Approach to V/STOL Simulation Employing a Variable Stability Helicopter. NASA TN D-1970, 1964.

4. Garren, John F.; Kelly, J. R.; and Reeder, John P.: A Visual Flight Investigation of Hovering and Low-Speed VTOL Control Requirements. NASA TN D-2788, 1965.

5. Anon.: Recommendations for V/STOL Handling Qualities. AGARD Report 408, October 1962.

6. Garren, John F.; Kelly, J. R.; and Reeder, John P.: Effects of Gross Changes in Static Directional Stability on V/STOL Handling Characteristics Based on a Flight Investigation. NASA TN D-2477, 1964.

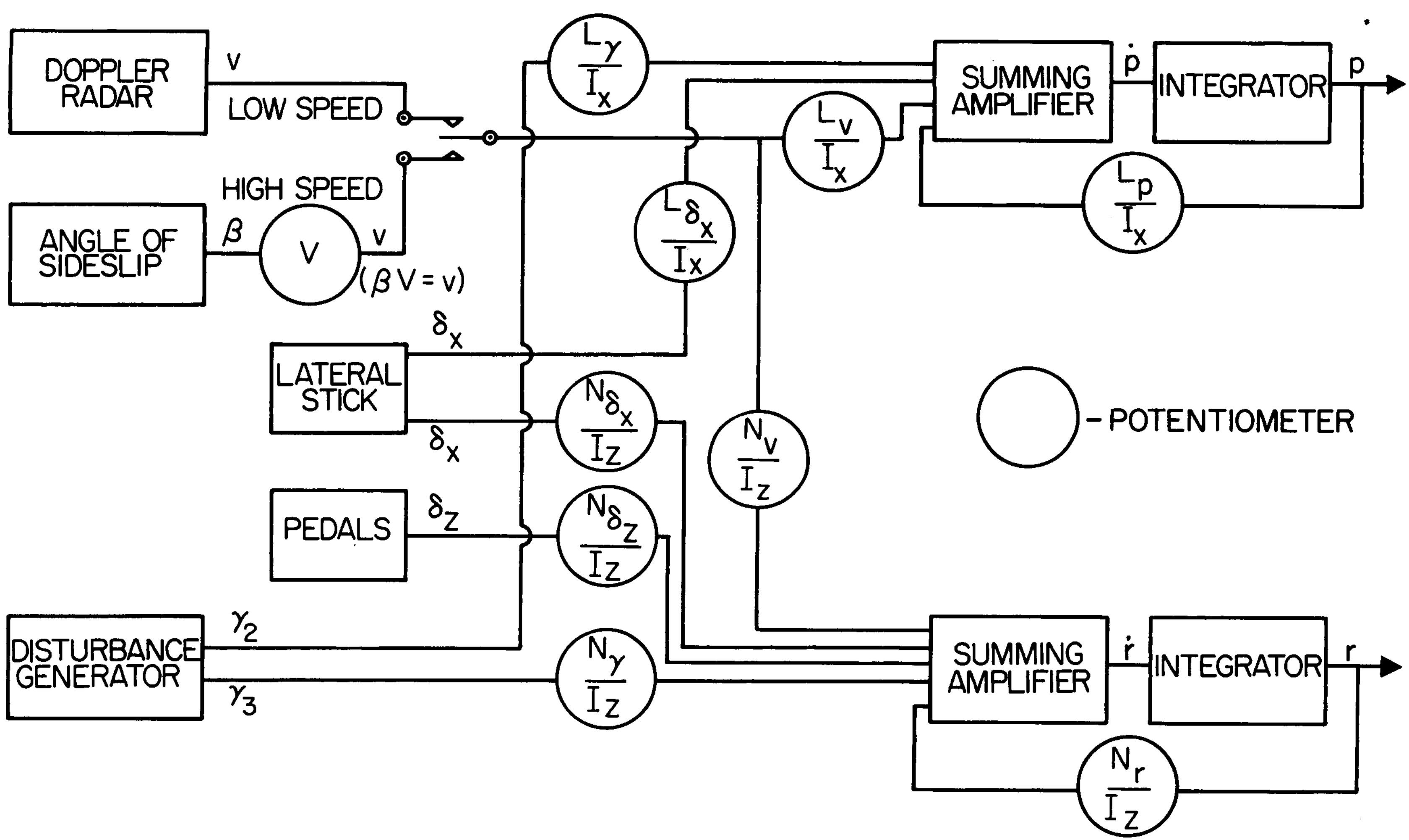

Figure 1.- Mechanization of lateral-directional equations.

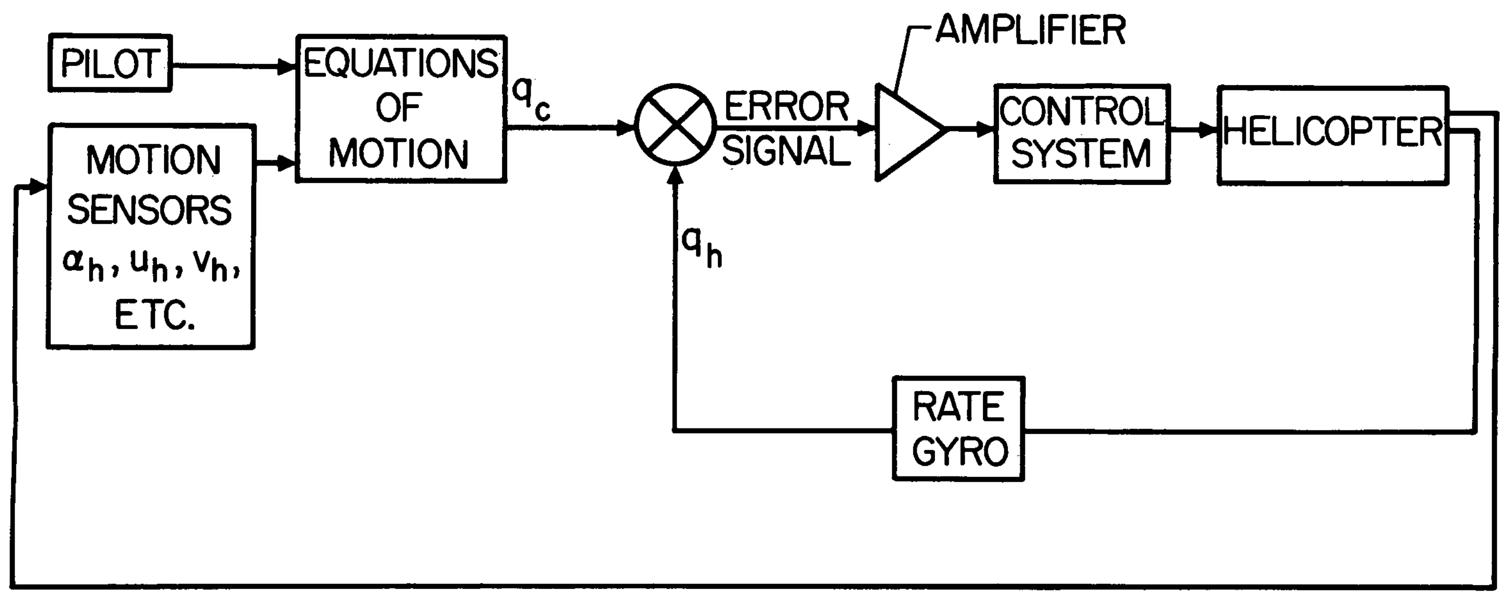

Figure 2.- Basic model technique.

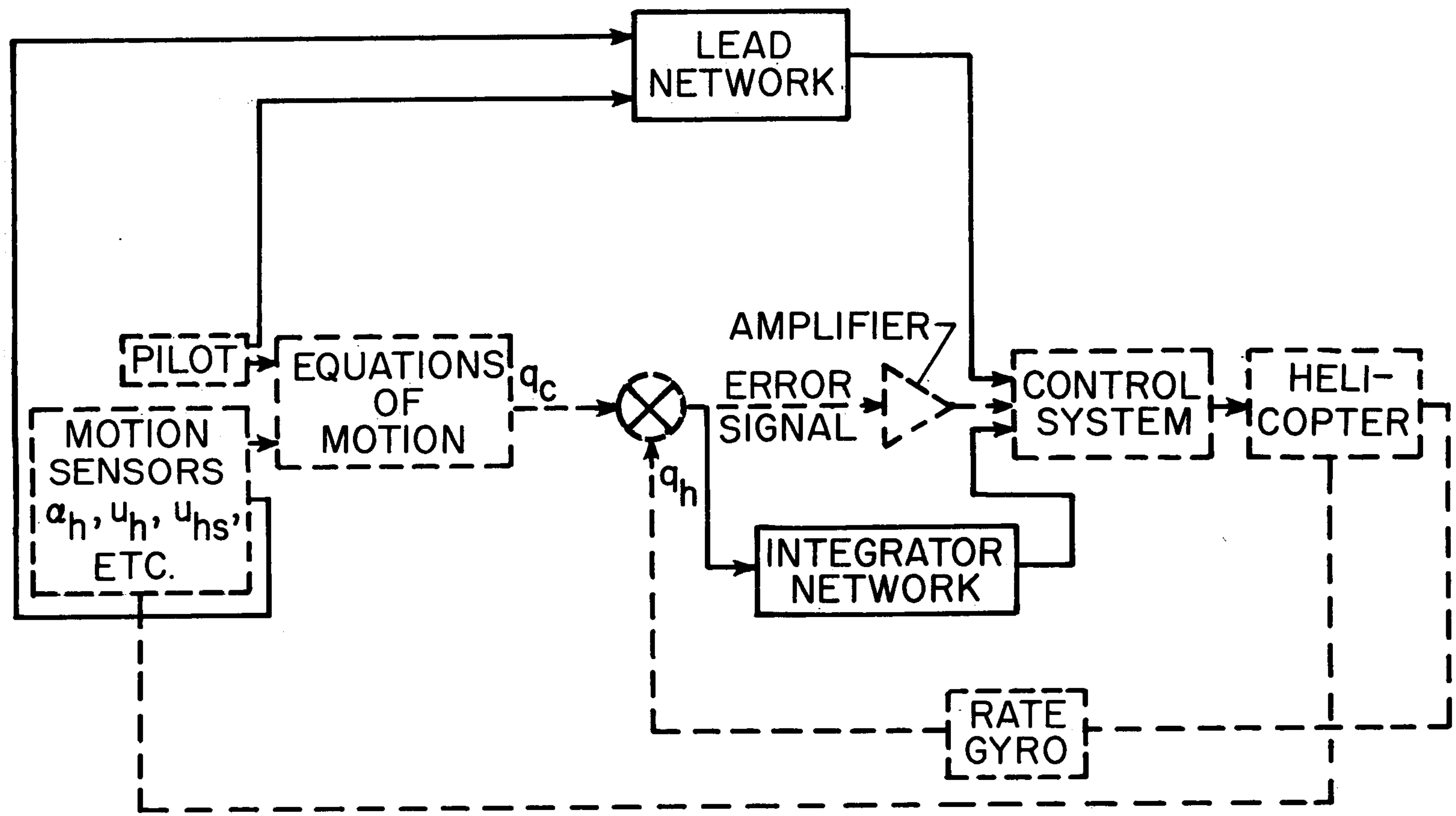

Figure 3.- Modified model technique.

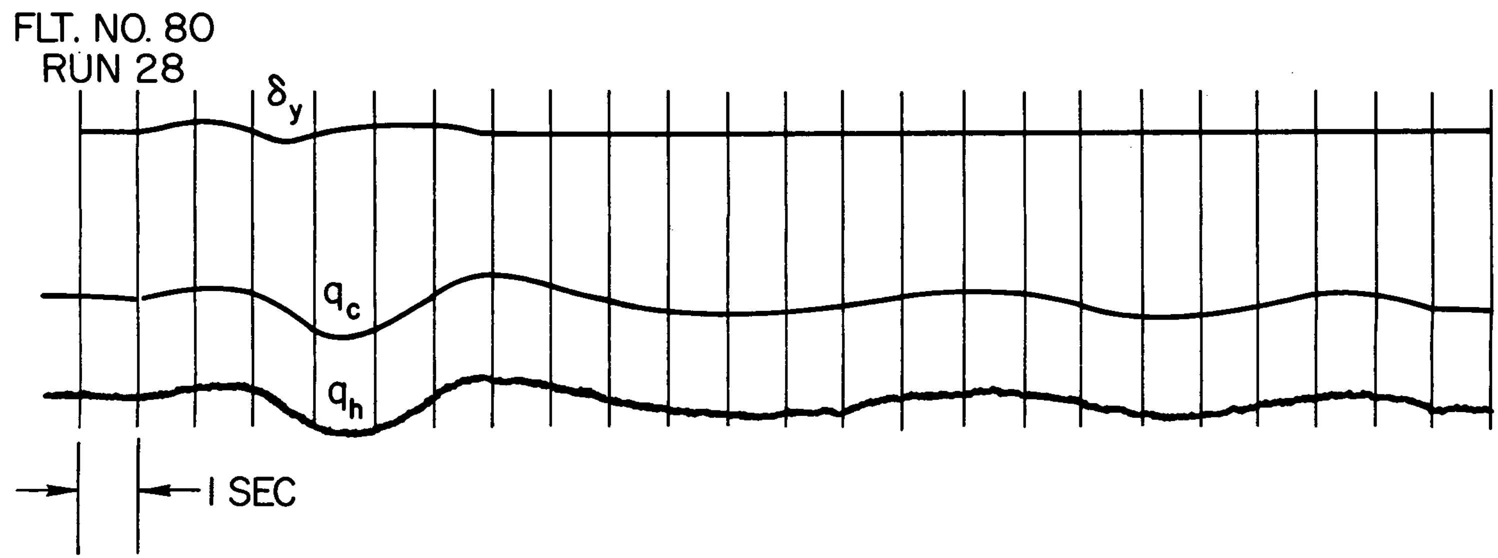

Figure 4.- Matching of a higher order model.

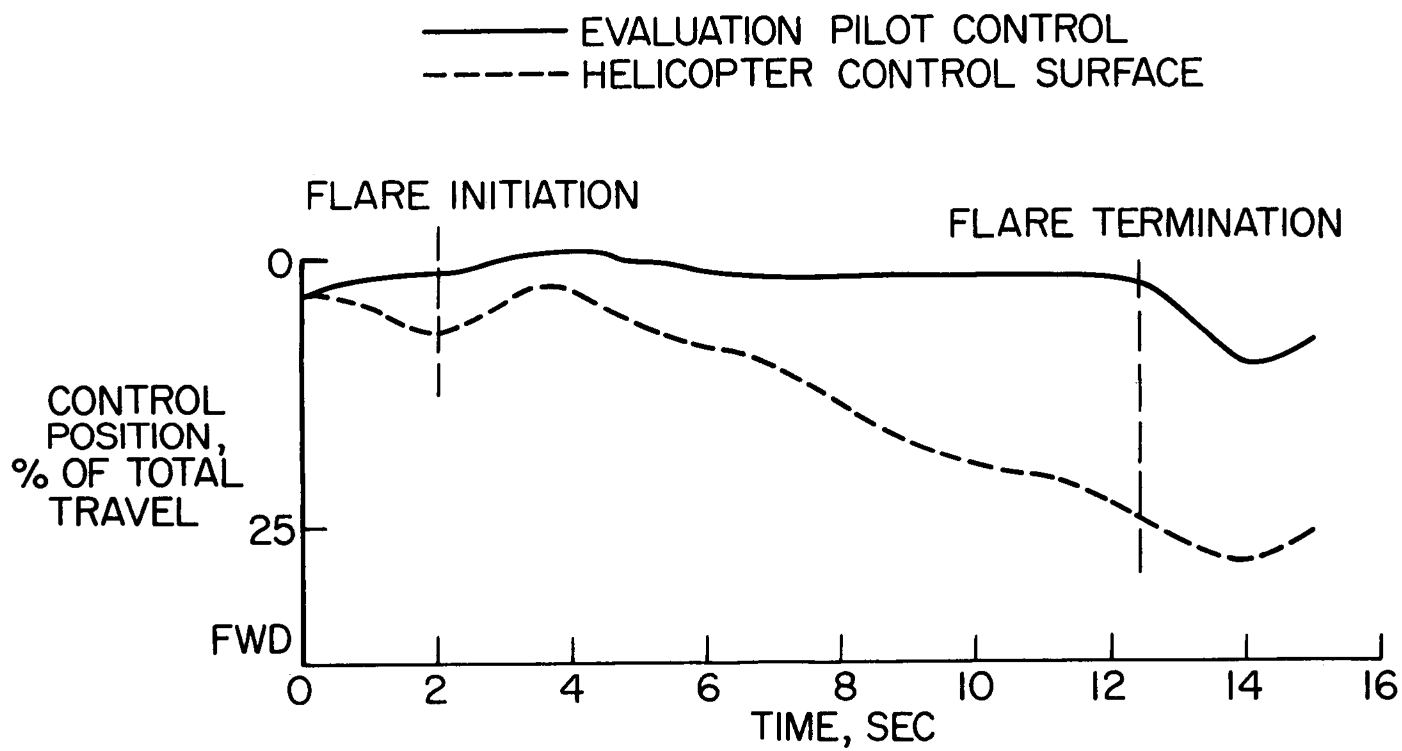

Figure 5.- Elimination of trim changes.

Figure 6.- Variable-stability helicopter.

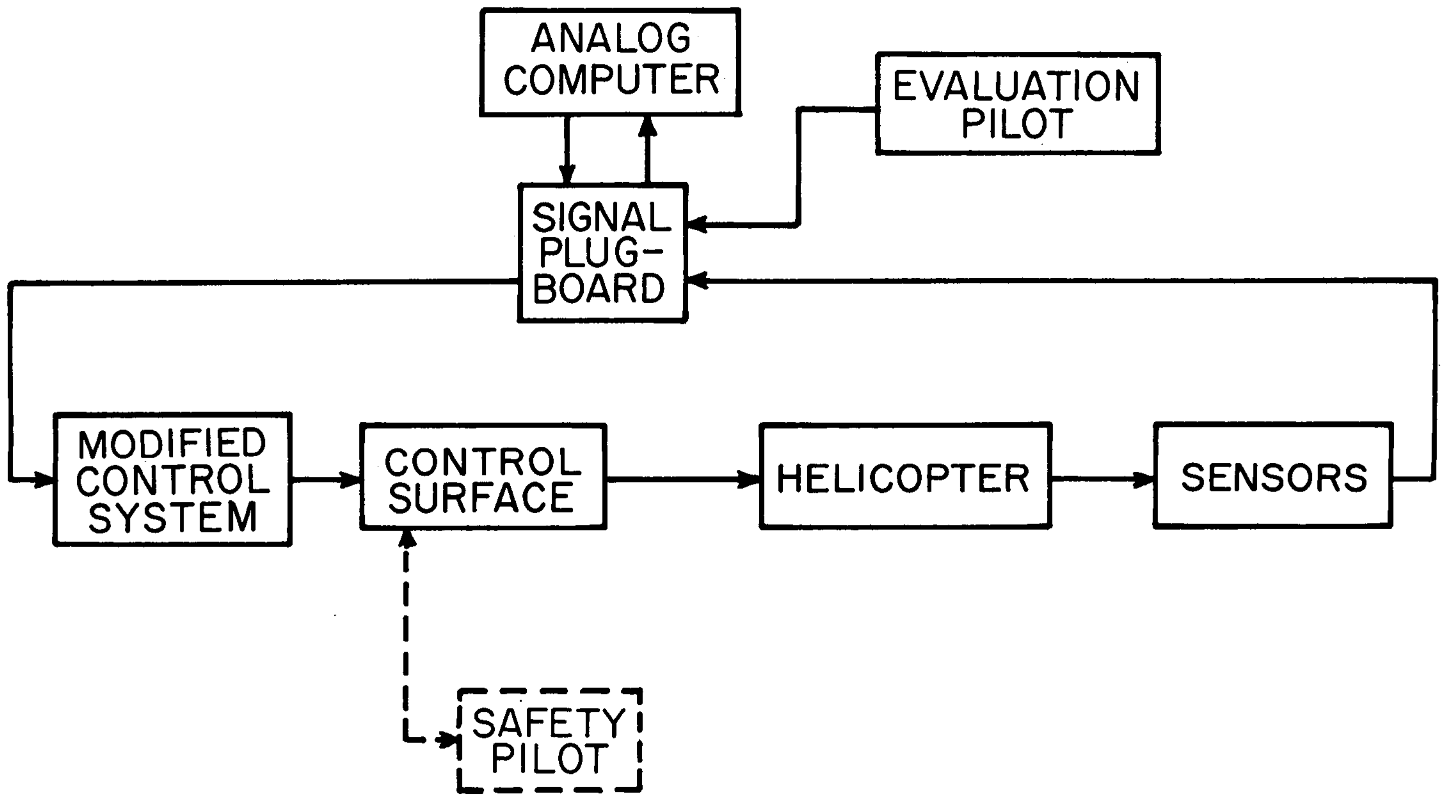

Figure 7.- Block diagram of variable-stability system.

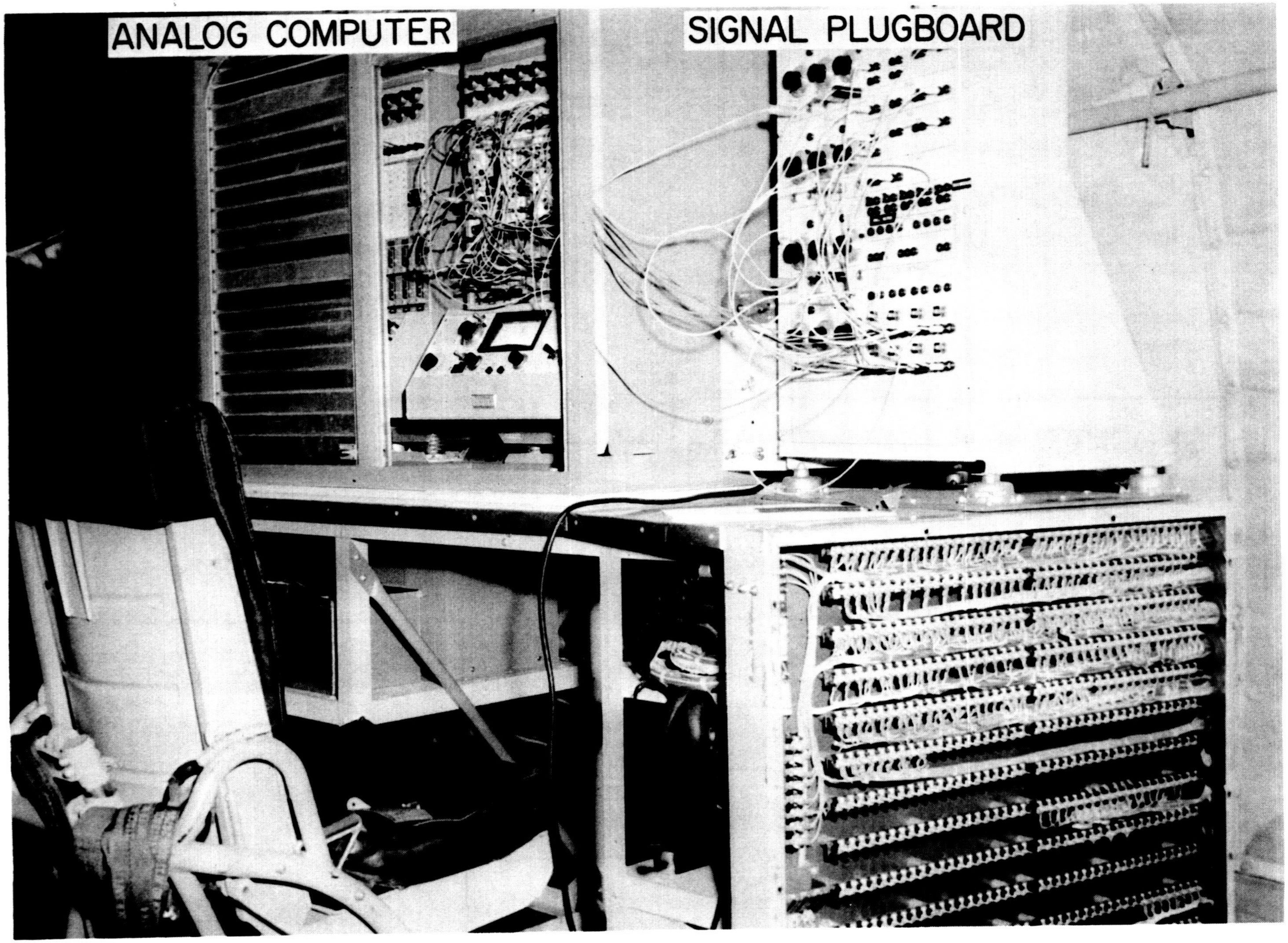

Figure 8.- Analog computing equipment and signal plugboard.