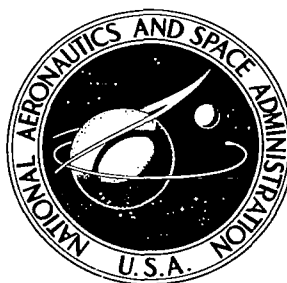


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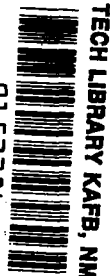


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**COMPUTER SIMULATION INCORPORATING  
A HELICOPTER MODEL FOR EVALUATION  
OF AIRCRAFT AVIONICS SYSTEMS**

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*Hampton, Va. 23665*

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NASA Technical Memorandum X-3491

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Aaron J. Ostroff and R. Brian Wood  
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COMPUTER SIMULATION INCORPORATING A HELICOPTER MODEL FOR  
EVALUATION OF AIRCRAFT AVIONICS SYSTEMS

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SUMMARY

A computer program has been developed to integrate avionics research in navigation, guidance, controls, and displays with a realistic aircraft model. A user oriented program is described that allows a flexible combination of user supplied models to perform research in any avionics area. A preprocessor technique for selecting various models without significantly changing the memory storage is included. Also included are mathematical models for several avionics error models and for the CH-47 helicopter used in this program.

INTRODUCTION

Avionics research is being conducted at the Langley Research Center in areas related to navigation, guidance, control, and displays (ref. 1). An overall simulation is required to evaluate and integrate the various areas of this research. The purpose of the present paper is to describe the simulation program and its capabilities.

One requirement is to develop a computer program that gives each researcher the capability of evaluating an avionics area independently from other research areas without rigid time schedules. Further, algorithms that are developed should be included into the simulation with minimal impact on the overall program. The preceding requirement has been met by using a modular type design. A set configuration representing all of the avionics systems envisioned has been defined, with separate modules representing the various measurement sensors and avionics functions. Potentially, 10 different user supplied models can be selected for each of the modules in the simulation. The choice of models is accomplished by a series of defaults; overriding can be accomplished with the use of input data cards. A specially designed preprocessor combines the input data and default data in order to assemble a complete program. This program includes a six-degree-of-freedom CH-47B helicopter model (ref. 2). After a model has been developed, it is added to the program subroutine library for use by other researchers.

The computer program must be simple to use and easily understood since users of various disciplines will be running the program. NAMELIST type data are used for the input, and only the data associated with the particular simula-

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\*Vought Corporation, Hampton, Virginia.

tion being run are required. This feature is accomplished by the series of defaults previously described.

Another requirement that the computer program should have is the capability of replacing any computer software module by its equivalent hardware and of evaluating the hardware in the overall simulation system. In certain cases, pilot capability is also required. These features dictate the need for a real-time simulation. Two programs have been developed and are maintained, one for batch type research and the other for real-time capability. These programs are maintained as nearly similar as possible within the limits of the Langley computer complex. The real-time program can be combined with hardware to meet this requirement. The computer software module can be deleted by a simple input data card that selects a dummy model with zero iteration frequency. The equivalent input communication is carried out through electrical input/output channels to the actual hardware.

Elements of structured type programming have been used where applicable. The mainline and much of the programming in the subroutines use top-down structuring. With the Langley FORTRAN compilers, structured programming tends to be inefficient with respect to time and, to some degree, memory storage. The efficiency problem has been traded off to maintain single-input, single-output type programming as much as possible.

An overlay program has been developed to minimize core storage. In general, the program can be run with approximately 60 000 octal words. The iteration frequency of every module is selectable by the user through a simple data card. Print options are also selected by input data cards, with each module having at least one option.

#### SYMBOLS

$A_1, A_2, A_3, A_4$	matrix components representing stability derivatives (eq. (15))
$B_1, B_2, B_3, B_4$	matrix components representing control derivatives (eq. (15))
$C_n^a$	matrix transformation from body frame to geographic frame
$C_n^b$	matrix transformation from geographic frame to body frame
$d_W$	correlation distance, m (eq. (12))
$E$	error term
$F$	$9 \times 9$ system matrix (eq. (10))
$F_X, F_Y, F_Z$	summation of all forces along longitudinal, lateral, and vertical body axes, respectively
$F_1, F_2$	general terms for longitudinal and lateral perturbation linear and angular accelerations (eqs. (15), (16), and (17))

$\vec{f}^b$  vector for acceleration of body relative to inertial space,  $m/sec^2$   
 (eq. (19))

$G$   $9 \times 6$  forcing matrix (eq. (10))

$g$  scalar magnitude of gravity,  $m/sec^2$

$\vec{g}^n$  gravity vector in geographic frame,  $m/sec^2$

$H_s$  station altitude, m

$I_{XX}, I_{YY}, I_{ZZ}$  aircraft moments of inertia about longitudinal, lateral, and vertical axes, respectively,  $kg-m^2$

$I_{XZ}$  aircraft product of inertia in longitudinal plane,  $kg-m^2$  (JXZ for computer example in appendix)

$I_1, I_2, I_3$  inertia products (eqs. (21))

$L_H, l_H$  aircraft latitude and longitude, rad

$\dot{L}_H, \dot{l}_H$  derivatives of aircraft latitude and longitude, rad/sec (eq. (24))

$L_s, l_s$  latitude and longitude of ground station, rad

$M_X, M_Y, M_Z$  rolling, pitching, and yawing moments, respectively

$m$  aircraft mass, kg

$N$  random number

$p, q, r$  aircraft roll rate, pitch rate, and yaw rate, respectively, rad/sec

$P_I, Q_I, R_I$  measured roll rate, pitch rate, and yaw rate, respectively, rad/sec

$Q$  white-noise strength (eq. (9))

$\vec{q}$   $6 \times 1$  forcing function vector (eq. (10))

$R$  range measurement, m (eq. (4))

$R_e$  radius of Earth, m

$R_H$  aircraft altitude relative to mass center of Earth, m (eq. (25))

$S$  scale factor

$u_a, v_a, w_a$  measured airspeed components along longitudinal, lateral, and vertical body axes, respectively, m/sec

$u_{a,N}, v_{a,N}, w_{a,N}$  nominal airspeed components along longitudinal, lateral, and vertical body axes, respectively, m/sec

$u_H, v_H, w_H$  aircraft groundspeed components along longitudinal, lateral, and vertical body axes, respectively, m/sec  
 $u_W, v_W, w_W$  wind velocity components along longitudinal, lateral, and vertical body axes, respectively, m/sec  
 $\dot{u}_I, \dot{v}_I, \dot{w}_I$  measured accelerations along longitudinal, lateral, and vertical body axes, respectively, m/sec<sup>2</sup>  
 $\vec{V}^n$  aircraft velocity vector in geographic frame, m/sec  
 $\dot{\vec{V}}^n$  aircraft acceleration vector in geographic frame, m/sec<sup>2</sup> (eq. (18))  
 $W$  wind magnitude, m/sec (eqs. (26) and (27))  
 $\dot{X}_{a,V,H}$  aircraft airspeed magnitude, m/sec (eq. (26))  
 $\dot{X}_H$  aircraft north groundspeed, m/sec  
 $\dot{X}_{V,H}$  aircraft groundspeed, m/sec (eqs. (12) and (26))  
 $x_1, x_2$  state perturbations (eq. (15))  
 $\dot{Y}_H$  aircraft east groundspeed, m/sec  
 $Z_H$  aircraft altitude, m (eq. (25))  
 $\dot{Z}_H$  aircraft vertical velocity, m/sec (eq. (24))  
 $\Delta F_X, \Delta F_Y, \Delta F_Z$  longitudinal, lateral, and vertical perturbation forces, respectively in body frame, m/sec<sup>2</sup> (eq. (14))  
 $\Delta M_X, \Delta M_Y, \Delta M_Z$  rolling, pitching, and yawing perturbation moments, respectively, in body frame, rad/sec<sup>2</sup> (eq. (14))  
 $\Delta p, \Delta q, \Delta r$  perturbations in roll rate, pitch rate, and yaw rate, respectively, in body frame, rad/sec (eq. (14))  
 $\Delta R_d$  range-difference measurement, m (eq. (7))  
 $\Delta u_a, \Delta v_a, \Delta w_a$  longitudinal, lateral, and vertical airspeed perturbation components, respectively, in body frame, m/sec (eq. (13))  
 $\delta L, \delta \ell$  navigator measured position errors in latitude and longitude, rad (eq. (11))  
 $\dot{\delta L}, \dot{\delta \ell}$  derivatives of  $\delta L$  and  $\delta \ell$   
 $\vec{\delta X}$  perturbation state vector (eq. (11))  
 $\dot{\vec{\delta X}}$  derivative of perturbation state vector (eq. (10))



$\delta Z$  navigator measured position error in altitude, m (eq. (11))  
 $\dot{\delta Z}$  derivative of  $\delta Z$   
 $\delta_B, \delta_C, \delta_S, \delta_R$  longitudinal, collective, lateral, and directional control stick positions, respectively, m (eq. (14))  
 $\vec{\delta}_T$  perturbation control stick vector, m (see fig. 3)  
 $\epsilon_n, \epsilon_e, \epsilon_d$  north, east, and downward platform tilt errors, respectively, rad (eq. (11))  
 $\eta$  angle of wind vector measured clockwise from north to tail of vector, rad (eqs. (26) and (27))  
 $\theta_{az}, \theta_{el}$  measured azimuth and elevation angles, rad (eqs. (6) and (5))  
 $\theta_H, \phi_H, \psi_H$  aircraft pitch, roll, and yaw Euler angles, respectively, rad  
 $\dot{\theta}_H, \dot{\phi}_H, \dot{\psi}_H$  derivatives of  $\theta_H, \phi_H,$  and  $\psi_H,$  respectively, rad/sec (eq. (22))  
 $\theta_I, \phi_I, \psi_I$  measured pitch, roll, and yaw Euler angles, respectively, rad  
 $\theta_N, \phi_N$  nominal pitch and yaw Euler angles, rad (eq. (19))  
 $\mu_1, \mu_2$  control-stick perturbations, m (eq. (15))  
 $\rho_j$  measured distance between master station and aircraft, m (eq. (8))  
 $\rho_k$  measured distance between slave station and aircraft, m (eq. (8))  
 $\sigma$  standard deviation  
 $\tau$  time constant, sec  
 $\psi_0$  initial aircraft groundtrack angle, rad (eqs. (26) and (27))  
 $\Omega_{en}^n$  skew symmetric matrix for angular velocity of geographic frame relative to an Earth fixed frame in geographic-frame coordinates, rad/sec (eq. (18))  
 $\vec{\omega}_{en}^n$  angular velocity vector whose elements comprise matrix  $\Omega_{en}^n,$  rad/sec  
 $\vec{\omega}_{ib}^b$  angular velocity vector of body frame relative to an inertial frame in body-frame coordinates, rad/sec  
 $\vec{\omega}_{nb}^b$  angular velocity vector of body frame relative to a geographic frame in body-frame coordinates, rad/sec (eq. (23))  
 $\dot{\vec{\omega}}_{ib}^b$  angular acceleration vector of body frame relative to an inertial frame in body-frame coordinates, rad/sec<sup>2</sup> (eq. (20))

Stability derivative notation:

$$B_A = \left. \frac{\partial B}{\partial A} \right|_{\text{trim}} \quad \text{where } B \text{ can be } F_X, F_Y, F_Z, M_X, M_Y, \text{ or } M_Z \text{ and } A \text{ can be } u_a, v_a, w_a, p, q, r, \delta_B, \delta_C, \delta_S, \text{ or } \delta_R$$

Subscripts:

a           airspeed

b,e,i,n   body frame, Earth frame, inertial frame, and navigational geographic frame, respectively

H           aircraft (specifically helicopter)

I           inertial

N           nominal

s           station

V           vertical heading frame

W           wind

## AVIONICS SIMULATION

### General Flow

The computer program is composed of many independent modules that represent either specific hardware onboard an actual aircraft (such as rate gyros), external navigation aids, or a physical function (such as winds). A simplified block diagram of the simulation is shown in figure 1. In general, the helicopter model

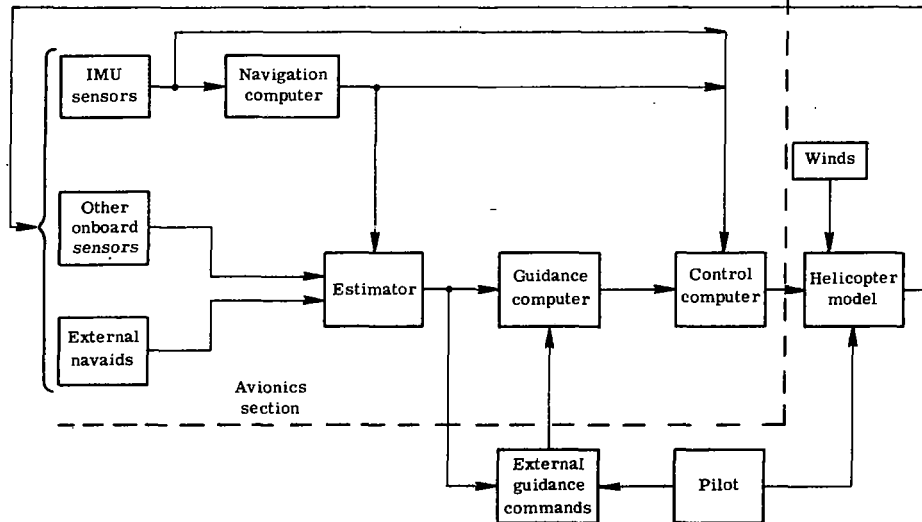


Figure 1.- Simplified block diagram of simulation.

calculates and feeds back perfect signals, such as linear and angular velocities, accelerations, positions, and angles. The sensor models use as inputs the helicopter derived signals and generate appropriate output variables to represent the measurements. For perfect measurements, the output is equivalent to the input. For models with errors, the output is a corrupted version of the input. Each of the three blocks for sensors actually represents several specific sensor modules that are described subsequently.

The outputs from the inertial measuring unit (IMU) sensors feed a navigator computer module. The presently available module represents an error model of the navigation computer, but the module can easily accommodate actual navigation computer equations.

All of the measurement data feed into an estimator module. A Kalman filter is the most likely candidate for the module, although any type of estimator may be included. Presently, a dummy model is installed with the output estimate being an exact duplicate of the input.

The guidance computer and external guidance commands are composed of several modules. Guidance commands can be generated internally by stored nominal trajectory data and automatic guidance steering laws. External commands are generated by the pilot or by air traffic control (ATC). The guidance outputs generally represent some type of perturbation error commands and are fed into the control computer. The control computer uses these commands to generate equivalent control stick commands. For the CH-47B helicopter, four control channels are used. The model is also capable of accepting input stick commands from the pilot.

The helicopter module consists of several submodules. The main sections include actuator and rotor nonlinearities and dynamics, force and moment calculations, equations of motion, and an update section to calculate all of the variables that are fed to the sensor modules.

#### Mainline

The mainline is essentially an executive that determines the time a module is to be executed. Each module is tested to determine the time to update the equations, whether it contains differential equations, and if the print option has been selected. The iteration frequency for each module is selected by either inputting the data or allowing the default value to be chosen. An initialization subroutine (TIMVARS) uses this frequency data to determine the update time increments and the initial starting times. For input frequencies of zero, the initial starting times are made very large such that the module is never called. The initial starting times for all other modules are set to zero, except for the external navigation aids which have selectable start and end times.

In the real-time version, if a module has differential equations, the module is entered and the differential equations are integrated independently of the update time. The entire module, including differential equations, is exe-

cuted when the update time is reached. In the batch program, the differential equations are integrated only during the module execution time.

### Inertial Measuring Unit (IMU) Modules

The IMU modules consist of four major modules, each of which have three measurements for attitudes, rates, linear accelerations, and angular accelerations. The last item is a dummy module with an iteration frequency that presently defaults to zero. This module is included to provide the capability of incorporating angular acceleration measurements if the need arises.

The three outputs from the attitude module are the body attitudes for pitch  $\theta_I$ , roll  $\phi_I$ , and heading  $\psi_I$ , all relative to a local level geographic Earth frame. The outputs from the rate gyro and linear accelerometer modules are body-axis measurements relative to inertial space. The three rate gyro outputs are the pitch rate  $q_I$ , roll rate  $p_I$ , and yaw rate  $r_I$ . The three accelerometer outputs are the specific force measurements  $\dot{u}_I$ ,  $\dot{v}_I$ , and  $\dot{w}_I$  in the longitudinal, lateral, and downward directions, respectively.

Two types of models are presently offered: a perfect sensor model with the output equal to the input and an error model with the output being a corrupted version of the input. All three error models have the capability of including both bias and random type errors. Both bias and random errors are calculated for every sensor by multiplying the output of a white-noise random-number generator with Gaussian distribution  $N$  by the standard deviation  $\sigma$  of the error; that is,

$$E = N\sigma \quad (1)$$

where  $E$  represents either type of error. The values of  $\sigma$  for both the bias and the random errors are input by the user, the default case being zero. Bias errors are initialized prior to the operate loop, whereas random errors are updated each time the error model is called.

Scale factor errors are also included in both the attitude-gyro and rate-gyro error models. The scale factor  $S$  for every gyro is initialized prior to the operate loop as

$$S = 0.01N\sigma + 1.0 \quad (2)$$

Note that all error models simplify to the perfect case if the input  $\sigma$  for each error is set equal to zero. The capability exists for incorporating several other types of user supplied models. This capability is discussed in the section entitled "Preprocessor."

### Other Onboard Sensors

The four modules included in the other-onboard-sensors block of figure 1 are air data, groundspeed, heading reference, and absolute altitude. Only perfect sensor models are presently included. The main reason for incorporating

these modules is to have the capability of adding models which represent these avionics systems with little impact on the overall computer program. The typical outputs from each module are defined and are in common storage, but the outputs and their definitions are subject to change, depending upon user preference. The default frequency is zero for each of these modules.

The presently defined air data outputs are the three body-axis components of true airspeed  $u_{AD}$ ,  $v_{AD}$ , and  $w_{AD}$ , barometric altitude  $Z_{BAD}$ , and angles of attack  $\alpha_{AD}$  and sideslip  $\beta_{AD}$ . Other typical air data outputs that might be considered, but are not part of the present module, include indicated airspeed, Mach number, and altitude rate.

The three outputs from the groundspeed module include the north, east, and downward components of Earth velocity. A typical onboard sensor that would give this type of measurement is Doppler radar.

The presently defined heading reference output is the ground-track angle relative to north; that is,

$$\psi_{\text{heading reference}} = \tan^{-1} \frac{\dot{Y}_H}{\dot{X}_H} \quad (3)$$

where  $\dot{X}_H$  and  $\dot{Y}_H$  represent the aircraft north and east groundspeeds, respectively. Another possible heading model that a user might incorporate is for the airspeed heading vector. Typical sensors for heading include a magnetic compass, directional gyro, and gyrocompass.

The absolute altitude output  $Z_{AA}$  is the altitude relative to the ground. A radar altimeter and an acoustic altimeter are typical types of sensors. This module also has the capability of inputting the height of the terrain directly below the aircraft. Presently, the terrain altitude is defaulted to zero, and  $Z_{AA}$  is the same as  $Z_{BAD}$ .

#### External Nav aids

Radio communication between ground-based stations and the aircraft provides external navigational data. The computer program has the capability of incorporating and using simultaneously up to nine different navaid modules in any combination, plus an additional capability of choosing any one of a possible nine models for each navaid. The three navigational measurements presently available are for range (DME), elevation and azimuth angles (MLS), and range difference (Loran, Omega). Each navigational type has two models from which to select. All models have calculations of the actual measurement based upon actual station coordinates (latitude, longitude, and altitude), actual helicopter coordinates, and noise related errors if the errors are selected. The second model of each navaid type also has a calculation of the nominal measurement based upon estimated station coordinates and desired aircraft coordinates. The nominal measurement would likely be used in a Kalman filter that is linearized about a nominal path. The station coordinates are input data for each of these modules.

Distance-measuring equipment (DME) is the range measuring external navaid presently installed. A total of 10 stations can be included in any run, with any 3 stations operating simultaneously. Call statements to the modules are functions of the iteration frequency and the start and stop time for any station measurement. Bias and random errors are included in the actual measurement and are calculated as described in equation (1).

The standard deviations for these errors and the time data for each station are inputs to the program. The actual calculation range measurement  $R$  is (ref. 3)

$$R = \left\{ R_e^2 \left[ (L_H - L_S)^2 + (\cos^2 L_S)(\ell_H - \ell_S)^2 \right] + (-Z_H - H_S)^2 \right\}^{1/2} + E \quad (4)$$

where  $L_H$ ,  $\ell_H$ , and  $Z_H$  are the aircraft coordinates (with  $Z_H$  positive down),  $L_S$ ,  $\ell_S$ , and  $H_S$  are the station coordinates,  $R_e$  is the Earth radius, and  $E$  is the noise error. If the aircraft is not within line of sight from the station, a possibility with spherical Earth, or is farther than the maximum DME range of 370 km (200 nautical miles), the measurement calculation will not be made. The maximum range can be varied by changing a DATA statement. A flag output is available whenever a valid measurement is made. If the print option for this module is selected, the program will print any of up to three measurements simultaneously, including the time of the last valid measurement for each station. Equation (4) is a good approximation within the DME measurement range. For the maximum range of 370 km, the error is approximately 0.15 percent high as compared to a more exact formulation that uses spherical trigonometry.

Microwave landing systems (MLS) are typical angle measuring external nav aids. The general equation for the actual elevation angle  $\theta_{el}$  is

$$\theta_{el} = \tan^{-1} \frac{-Z_H - H_S - 0.5R_e \left[ (L_H - L_S)^2 + (\cos^2 L_S)(\ell_H - \ell_S)^2 \right]}{R_e \left[ (L_H - L_S)^2 + (\cos^2 L_S)(\ell_H - \ell_S)^2 \right]^{1/2}} + E \quad (5)$$

The last term in the numerator is a correction factor since MLS systems generally provide angles relative to a coordinate frame that has the origin at the azimuth antenna. The equation for the actual azimuth angle  $\theta_{az}$  is

$$\theta_{az} = \tan^{-1} \frac{(\cos L_S)(\ell_H - \ell_S)}{L_H - L_S} + E \quad (6)$$

The angles calculated in equations (5) and (6) are applicable to a conical measuring type system.

The error  $E$  is different in each case. The inputs are similar to those for the range measurement, with the addition of a center-line angle for the azimuth measurement. A measurement will not be made if the azimuth angle is outside a  $60^\circ$  boundary from the center line, the elevation angle is not within the limits of  $1^\circ$  to  $20^\circ$ , or the maximum range of 55.6 km (30 nautical miles) is exceeded. The limits can be changed by modifying a DATA statement.

The range-difference models can be used for either Omega or Loran. Inputs include position coordinates for both one master and two slave stations (ref. 3), iteration frequency, start and end times for a complete measurement, and time lags for each of the slave station measurements. The time lags are appropriate for Omega where each station has an independent transmission time with a constant iteration period of 10 sec. The measurement for each line of position is a range difference  $\Delta R_d$  between the master and a slave as given by (ref. 3):

$$\Delta R_d = \rho_j - \rho_k + E \quad (7)$$

where

$$\rho_i = R_e \cos^{-1} [\sin L_H \sin L_S + \cos L_H \cos L_S \cos (\ell_H - \ell_S)] \quad (i = j, k) \quad (8)$$

and  $\rho_j$  is the distance between the master station and the aircraft and  $\rho_k$  is the distance between a slave station and the aircraft. A total of two range difference measurements are made. Measurement errors include a random error and a correlated error. The correlated error is appropriate for Omega and is modeled as a first-order time lag to represent a slowly varying bias. Both the time constant  $\tau$  for the correlated error and the standard deviation  $\sigma$  must be input to override the default values of zero. The white-noise strength  $Q$  is (ref. 3)

$$Q = \frac{2\sigma^2}{\tau} \quad (9)$$

and is calculated in the range-difference initialization module.

### Navigation Computer

The nine outputs from the navigation computer are the three position coordinates, three velocity coordinates, and three planform tilt errors when error models are included. Presently, three model choices are offered. The default model is the perfect case with the output equivalent to the input. The other two choices are error models. The second model (first error model choice) incorporates seven states and is representative of most present inertial navigation systems. The vertical channel is not included since it is unstable without an external reference. The last model includes all nine states and is included to provide the capability of future evaluation. In this section, the error model description is the nine-state model, since it is the most general case.

The general error model takes the form of a first-order, nine-state, vector differential equation (ref. 4) as follows:

$$\delta \dot{\vec{X}} = F \delta \vec{X} + G \vec{d} \quad (10)$$

where

$$\delta \vec{X}^T = [\epsilon_n, \epsilon_e, \epsilon_d, \delta \dot{L}, \delta \dot{\ell}, \delta L, \delta \ell, \delta \dot{Z}, \delta Z]^T \quad (11)$$

and

$\epsilon_n, \epsilon_e, \epsilon_d$  north, east, and downward platform tilt errors

$\delta\dot{L}, \delta\dot{\lambda}, \delta\dot{Z}$  latitude, longitude, and altitude rate errors

$\delta L, \delta \lambda, \delta Z$  latitude, longitude, and altitude position errors

F  $9 \times 9$  system matrix

G  $9 \times 6$  forcing matrix

$\vec{q}$   $6 \times 1$  forcing function vector of inertial system errors

The six inertial system errors consist of the three rate gyro errors and three accelerometer errors, all transformed from the body axis to a local level geographic frame. The inputs to this module are the nine initial errors which are defaulted to zero if the input data are not included.

### Estimator

The estimator module (see fig. 1) is for a Kalman filter or any other type of filter estimator to be installed. Inputs to the estimator are obtained from various sensors, external nav aids, and the navigation computer. Presently, two dummy models are installed. The default model assumes perfect estimates of the aircraft position and velocity and of the wind velocity and direction. This allows perfect states to be used for guidance and control while nonnavigational type errors are evaluated. The second dummy model assumes estimates that are identical to the outputs of selected error models, allowing corrupted signals to be fed back to the guidance and control system.

### Guidance System

Guidance commands can be generated in any of four major modules. The first module is for storing and calculating the nominal trajectory. This module contains all of the preflight data and path generating algorithms such as leg switching, velocity commands, altitude commands, turning commands, nominal position commands, spiral descent guidance commands, and others. The second module is for automatic guidance steering laws. This module contains correction commands that modify the nominal trajectory as a function of aircraft deviation from the commanded path. Typically, this includes modifying the nominal altitude rate due to deviations in altitude, modifying the nominal bank angle due to errors in heading and lateral position, and modifying the nominal acceleration or velocity due to errors in velocity and time. The latter is typically a four-dimensional time-constrained law.

The third and fourth sources of guidance commands are the air traffic control (ATC) and the pilot. Commands from either of these sources can override the earlier guidance commands, the pilot having highest priority. Presently, the contents of both these modules are empty.



Another function of the guidance computer is to generate error perturbation commands for the control system. Presently, only velocity perturbations, based upon differences between command velocities and measured velocities, are generated. The velocity perturbations are first generated in a geographic coordinate frame and then rotated to the body axis for use by the controller.

Generally, a guidance steering law model is related to a nominal trajectory model; for the present discussion, a model pair refers to the combination. There are two model pairs presently available. The first pair (default case) is for enroute type guidance and can handle data for up to 10 waypoints. Typical input data include waypoint position coordinates and desired aircraft velocity and altitude. It is also possible to input desired aircraft rate of descent and nominal time at each waypoint, although these data are not used in the present guidance laws. Separate submodules are used to calculate velocity commands (VELCOM), altitude rate commands (CLIMB), and turn commands (LATTURN). A submodule for calculating the nominal position (PSITION) is also included. Algorithms for calculating the correct leg of the trajectory and the times to change various guidance commands are included in submodule LEGSWCH. The only guidance steering law in the first model pair is an altitude hold submodule (ALTHOLD) that modifies the altitude rate as a function of altitude error.

The second model pair is for both the enroute phase of flight and for spiral descent guidance (ref. 5). The enroute guidance is identical to that in the first model pair. When the last waypoint is reached, the spiral descent guidance is automatically switched in. If only one enroute point is included in the simulation run, the program will start with the spiral descent guidance laws. The input data required are the coordinates of the landing pad and spiral center, airspeed, spiral turning radius, and turn direction. The last three items have default values.

### Control Computer

The control-computer module is used for any of the research control laws to be tested. Typical inputs include attitude and rate measurements, guidance commands, and velocity error perturbations. The outputs are four stick commands for the helicopter control inputs. The default model presently installed has a velocity control law.

### Winds

Three wind models are presently offered. The default model is the zero wind case and sets all wind variables to zero. The second model is for either a constant horizontal wind vector or a wind shear. The wind shear allows a variation of the horizontal wind vector with altitude. Up to four data points can be included for both wind velocity and wind direction relative to north. Straight line interpolation is used, with altitude the independent variable. If only one data point is entered, a steady wind will be used, and if no data points are entered, a zero wind will be calculated. The vertical wind is assumed zero in this model.

The third wind model contains the contents of the second model plus variable winds in both horizontal and vertical channels. The variable winds are modeled as exponentially correlated winds with specified correlation distances  $d_W$  and standard deviations  $\sigma$ , plus random gusts (ref. 3). The white-noise strength  $Q$  is calculated as shown in equation (9), where a separate  $\sigma$  is input for the horizontal velocity, horizontal direction, and vertical velocity. The three correlation distances have defaulted values but can be overridden by input data. The effective time constants  $\tau$  for the exponentially correlated winds are a function of  $d_W$  and the groundspeed  $\dot{X}_{V,H}$ ; that is,

$$\tau = \frac{d_W}{\dot{X}_{V,H}} \quad (12)$$

The random gusts for both horizontal velocity and vertical velocity are calculated as in equation (1). The third model reduces to the second model by setting the input standard deviations to zero, and both models reduce to the default model with all input data defaulted to zero. The iteration frequency of the wind module is automatically set to the aircraft model iteration frequency and cannot be adjusted.

#### AIRCRAFT SIMULATION MODEL

A Boeing Vertol CH-47B helicopter is being used as a tool to evaluate advanced avionics research concepts as part of the automatic avionics program (ref. 1). A model of the CH-47B is included in this computer program to evaluate the avionics concepts prior to flight. One model is presently offered, although the preprocessor capability and modular concept allow other models to be added in the future. A linear perturbation model, for computing aerodynamic forces and moments, has been selected over a nonlinear model (ref. 6) to minimize computation time. A unique technique for updating the trims and derivatives, to maintain small perturbations over the complete flight regime, is described in this section. The kinematic equations of motion are fully coupled nonlinear equations.

The helicopter model consists of several submodules as shown in figure 2. Typically, control inputs originate from the control computer, including a stability augmentation system, and from the pilot. These control forces are inputs to the actuator and rotor section which includes nonlinearities and dynamics. The control-stick trim nominals are combined with the equivalent total control-stick inputs to generate perturbation stick commands. The airspeed submodule combines airspeed trim nominals, winds, and actual helicopter velocities to calculate perturbation airspeeds. Perturbation airspeeds, perturbation stick commands, and perturbation angular velocities are combined with the stability and control derivatives to calculate the total perturbation forces and moments acting on the helicopter. The total helicopter forces and moments are calculated by adding nominals and perturbation forces and moments. The equations of motion contain 12 differential equations for linear and angular accelerations and velocities, and take into account spherical Earth. In the update module, all of the helicopter variables are updated for the next time period. Each of the helicopter submodules is described in more detail in the following sections.

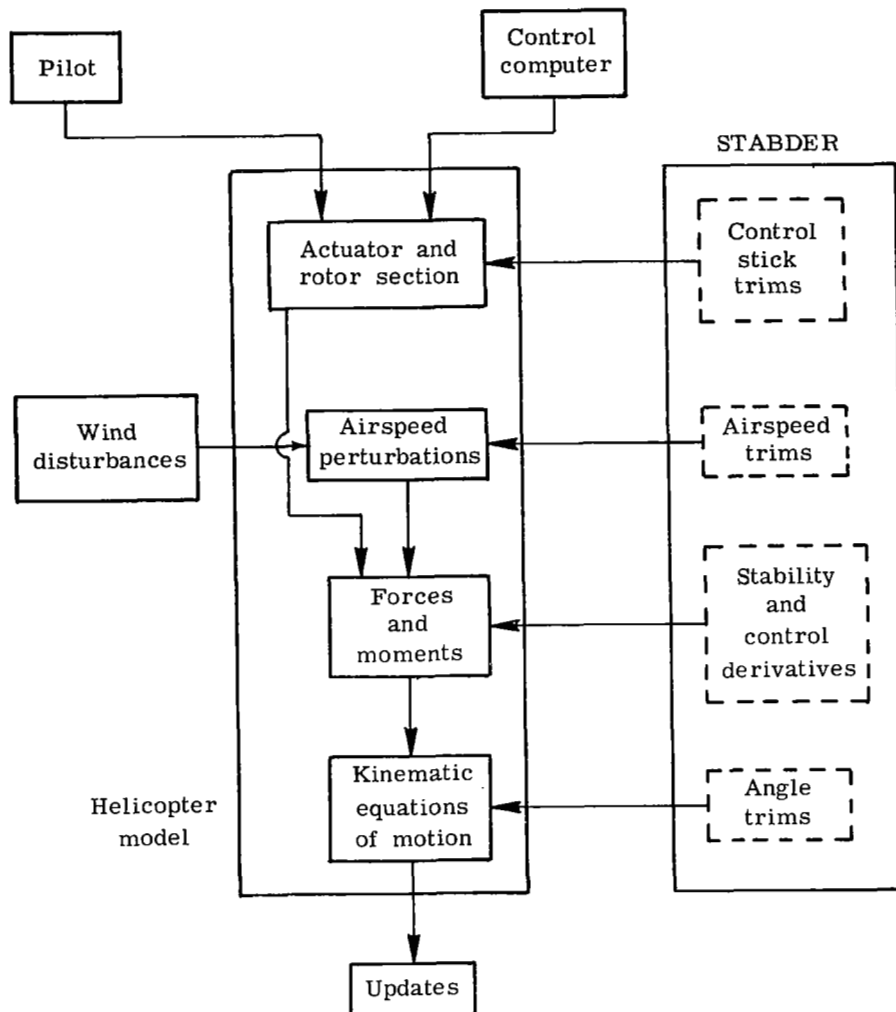


Figure 2.- Block diagram of helicopter module.

### Actuator and Rotor Section

An analysis has been made of the CH-47B control linkage system, from stick inputs to rotor motion, and the dominant system time constants and nonlinearities are taken into account. (See fig. 3.) One change in the research CH-47B model from a standard model is the addition of electrohydraulic actuators between the control sticks and the transfer bell cranks. The actuator position limit ( $+5.08$  cm from ref. 6) is the limiting mechanical displacement between the actuator input and rotor blades. It is possible for the pilot's control stick to be physically limited before the actuator limit occurs, but the stick limits would be modeled in the pilot module rather than the helicopter module. The dominant-system rate-limit constraint comes from the electrohydraulic actuator ( $+7.62$  cm/sec from ref. 6). Neither the upper boost nor the lower boost actuators (ref. 7) will reach their respective rate limits if the electrohydraulic actuator rate limit is not reached. The time constants of both the electro-

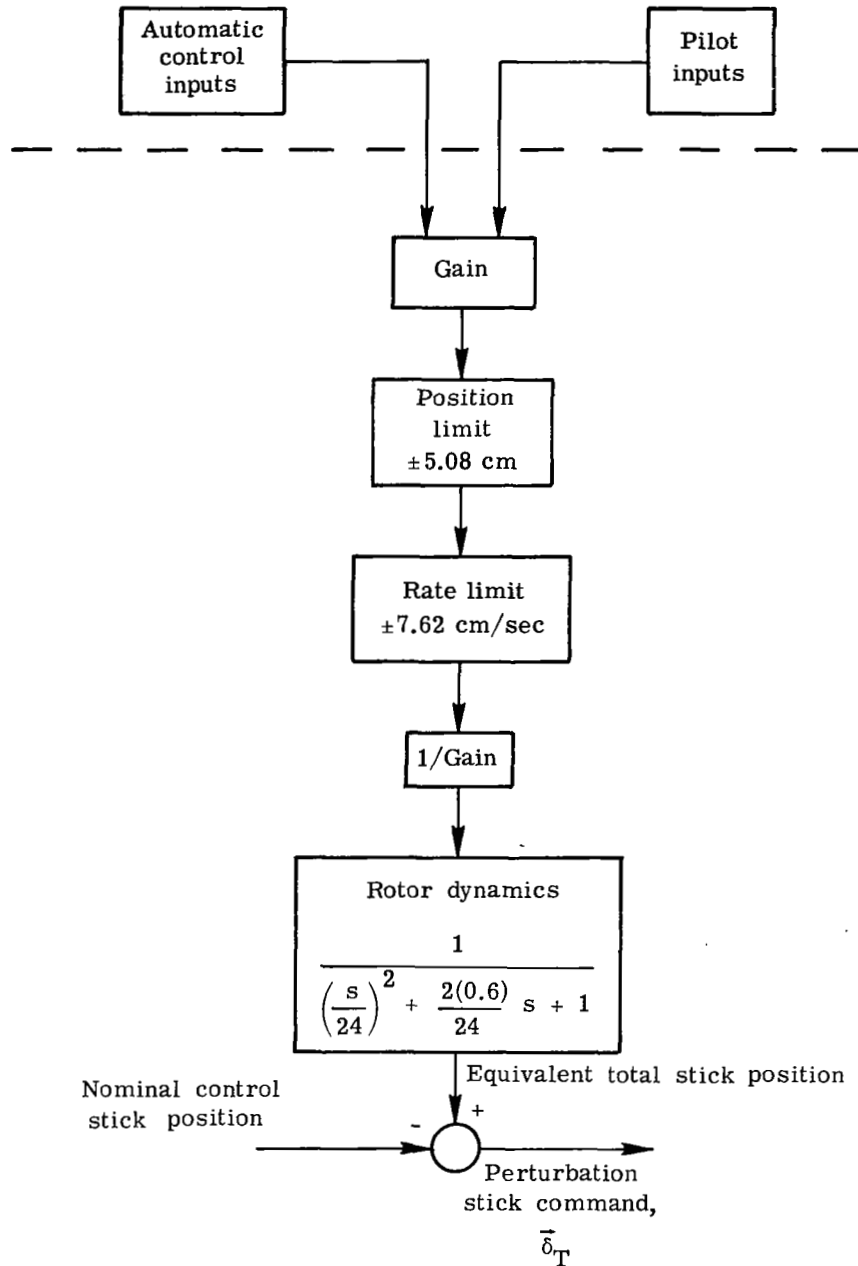


Figure 3.- Block diagram of CH-47B actuator and rotor section.

hydraulic actuators (0.0125 sec from unpublished data) and the lower boost actuators (0.0024 sec from ref. 8) are extremely small and are ignored. The upper boost actuators have an approximate second-order response with a damping ratio of 0.55 and a natural frequency of 50 rad/sec. (See ref. 6.) Their effect on the total system dynamics is minor when compared with the rotor flapping dynamics which has a damping ratio of approximately 0.6 and a natural frequency of 24 rad/sec (from unpublished data). The blocks in figure 3 labeled "Gain" and "1/Gain" are included to represent actual physical quantities. The gains for all four channels are selected such that a full electrohydraulic actuator motion

corresponds to a full pilot control stick throw. The second gain block is included to maintain the same total system steady-state gain used in calculating the trims and derivatives (ref. 2). The output from the rotor dynamics represents an equivalent total stick position. This value is summed with the corresponding nominal control stick position to get the perturbation stick command  $\delta_T$ .

### Airspeeds

The three body-axis airspeed perturbations  $\Delta u_a$ ,  $\Delta v_a$ , and  $\Delta w_a$  are calculated in the airspeeds submodule as

$$\begin{bmatrix} \Delta u_a \\ \Delta v_a \\ \Delta w_a \end{bmatrix} = \begin{bmatrix} u_H \\ v_H \\ w_H \end{bmatrix} - \begin{bmatrix} u_W \\ v_W \\ w_W \end{bmatrix} - \begin{bmatrix} u_{a,N} \\ v_{a,N} \\ w_{a,N} \end{bmatrix} \quad (13)$$

where  $u_H$ ,  $v_H$ , and  $w_H$  are the actual aircraft body-axis velocity components,  $u_W$ ,  $v_W$ , and  $w_W$  are the three wind components in body axis, and  $u_{a,N}$ ,  $v_{a,N}$ , and  $w_{a,N}$  are the nominal airspeed trim components in body axis. The approach for calculating the trims is described later in this section.

### Forces and Moments

Perturbation linear accelerations  $\Delta F_X/m$ ,  $\Delta F_Y/m$ , and  $\Delta F_Z/m$  and angular accelerations  $\Delta M_X/I_{XX}$ ,  $\Delta M_Y/I_{YY}$ , and  $\Delta M_Z/I_{ZZ}$  are calculated in the forces and moments submodule by linearly combining the aerodynamic and control force and moment contributions as follows:

$$\begin{bmatrix} \frac{\Delta F_X}{m} \\ \frac{\Delta F_Z}{m} \\ \frac{\Delta M_Y}{I_{YY}} \\ \frac{\Delta F_Y}{m} \\ \frac{\Delta M_X}{I_{XX}} \\ \frac{\Delta M_Z}{I_{ZZ}} \end{bmatrix} = \begin{bmatrix} \frac{F_{X,u}}{m} & \frac{F_{X,w}}{m} & \frac{F_{X,q}}{m} & \frac{F_{X,v}}{m} & \frac{F_{X,p}}{m} & \frac{F_{X,r}}{m} \\ \frac{F_{Z,u}}{m} & \frac{F_{Z,w}}{m} & \frac{F_{Z,q}}{m} & \frac{F_{Z,v}}{m} & \frac{F_{Z,p}}{m} & \frac{F_{Z,r}}{m} \\ \frac{M_{Y,u}}{I_{YY}} & \frac{M_{Y,w}}{I_{YY}} & \frac{M_{Y,q}}{I_{YY}} & \frac{M_{Y,v}}{I_{YY}} & \frac{M_{Y,p}}{I_{YY}} & \frac{M_{Y,r}}{I_{YY}} \\ \frac{F_{Y,u}}{m} & \frac{F_{Y,w}}{m} & \frac{F_{Y,q}}{m} & \frac{F_{Y,v}}{m} & \frac{F_{Y,p}}{m} & \frac{F_{Y,r}}{m} \\ \frac{M_{X,u}}{I_{XX}} & \frac{M_{X,w}}{I_{XX}} & \frac{M_{X,q}}{I_{XX}} & \frac{M_{X,v}}{I_{XX}} & \frac{M_{X,p}}{I_{XX}} & \frac{M_{X,r}}{I_{XX}} \\ \frac{M_{Z,u}}{I_{ZZ}} & \frac{M_{Z,w}}{I_{ZZ}} & \frac{M_{Z,q}}{I_{ZZ}} & \frac{M_{Z,v}}{I_{ZZ}} & \frac{M_{Z,p}}{I_{ZZ}} & \frac{M_{Z,r}}{I_{ZZ}} \end{bmatrix} \begin{bmatrix} \Delta u_a \\ \Delta w_a \\ \Delta q \\ \Delta v_a \\ \Delta p \\ \Delta r \end{bmatrix} + \begin{bmatrix} \frac{F_{X,\delta B}}{m} & \frac{F_{X,\delta C}}{m} & \frac{F_{X,\delta S}}{m} & \frac{F_{X,\delta R}}{m} \\ \frac{F_{Z,\delta B}}{m} & \frac{F_{Z,\delta C}}{m} & \frac{F_{Z,\delta S}}{m} & \frac{F_{Z,\delta R}}{m} \\ \frac{M_{Y,\delta B}}{I_{YY}} & \frac{M_{Y,\delta C}}{I_{YY}} & \frac{M_{Y,\delta S}}{I_{YY}} & \frac{M_{Y,\delta R}}{I_{YY}} \\ \frac{F_{Y,\delta B}}{m} & \frac{F_{Y,\delta C}}{m} & \frac{F_{Y,\delta S}}{m} & \frac{F_{Y,\delta R}}{m} \\ \frac{M_{X,\delta B}}{I_{XX}} & \frac{M_{X,\delta C}}{I_{XX}} & \frac{M_{X,\delta S}}{I_{XX}} & \frac{M_{X,\delta R}}{I_{XX}} \\ \frac{M_{Z,\delta B}}{I_{ZZ}} & \frac{M_{Z,\delta C}}{I_{ZZ}} & \frac{M_{Z,\delta S}}{I_{ZZ}} & \frac{M_{Z,\delta R}}{I_{ZZ}} \end{bmatrix} \begin{bmatrix} \delta_B \\ \delta_C \\ \delta_S \\ \delta_R \end{bmatrix} \quad (14)$$

The aerodynamic driving terms are perturbations in airspeeds  $\Delta u_a$ ,  $\Delta v_a$ , and  $\Delta w_a$  and angular velocities  $\Delta p$ ,  $\Delta q$ , and  $\Delta r$ , and the control force inputs are perturbations in the four control stick inputs  $\delta_B$ ,  $\delta_C$ ,  $\delta_S$ , and  $\delta_R$ . Equation (14) is partitioned to separate the longitudinal and lateral equations

$$\begin{bmatrix} F_1 \\ \text{---} \\ F_2 \end{bmatrix} = \begin{bmatrix} A_1 & A_2 \\ \text{---} & \text{---} \\ A_3 & A_4 \end{bmatrix} \begin{bmatrix} x_1 \\ \text{---} \\ x_2 \end{bmatrix} + \begin{bmatrix} B_1 & B_2 \\ \text{---} & \text{---} \\ B_3 & B_4 \end{bmatrix} \begin{bmatrix} \mu_1 \\ \text{---} \\ \mu_2 \end{bmatrix} \quad (15)$$

where the longitudinal outputs  $F_1$  are

$$F_1 = \left[ \frac{\Delta F_X}{m}, \frac{\Delta F_Z}{m}, \frac{\Delta M_Y}{I_{YY}} \right]^T \quad (16)$$

and the lateral outputs  $F_2$  are

$$F_2 = \left[ \frac{\Delta F_Y}{m}, \frac{\Delta M_X}{I_{XX}}, \frac{\Delta M_Z}{I_{ZZ}} \right]^T \quad (17)$$

The A matrix terms  $A_1$ ,  $A_2$ ,  $A_3$ , and  $A_4$  represent the stability derivatives, the B matrix terms  $B_1$ ,  $B_2$ ,  $B_3$ , and  $B_4$  represent the control derivatives, the state perturbations are represented by  $x_1$  and  $x_2$ , and the control stick perturbations are represented by  $\mu_1$  and  $\mu_2$ . Equation (15) is shown in the most general form with coupled aerodynamics. Generally, the coupling derivatives  $A_2$ ,  $A_3$ ,  $B_2$ , and  $B_3$  have negligible effect relative to the uncoupled derivatives  $A_1$ ,  $A_4$ ,  $B_1$ , and  $B_4$ . Only the uncoupled derivatives are being used in the present model to save storage and computing time. All of the derivatives are available (ref. 2) and can be incorporated. Derivatives for the complete flight regime are stored in a stability derivative module (STABDER), and the technique for updating the derivatives shown in equation (14) will be described. The derivatives are updated at regular intervals to maintain small perturbations from nominal trim conditions. One second appears to give sufficiently good results.

### Kinematic Equations of Motion

This section describes the kinematic-equations-of-motion submodule; 12 differential equations are included to describe the total forces and moments acting on the vehicle, Euler rates, and Earth referenced velocities. The equations are for a spherical, nonrotating Earth. The linear acceleration and velocity equations are referenced to a local level geographic coordinate frame whereas the angular acceleration and rate equations are referenced to the body axis.

The vehicle acceleration vector  $\dot{\vec{v}}^n$  relative to the local level geographic frame is

$$\dot{\vec{v}}^n = C_{fb}^n \dot{\vec{v}}^b + \vec{g}^n - \Omega_{en}^n \vec{v}^n \quad (18)$$

where  $C_b^n$  is a matrix transformation from body-axis frame to geographic frame,  $\vec{g}^n$  is the gravity vector in the geographic frame,  $\Omega_{en}^n$  is a skew symmetric matrix for the angular velocity of the geographic frame relative to the Earth

fixed frame,  $\vec{v}^n$  is the aircraft velocity vector in the geographic frame, and  $\vec{f}^b$  is the specific force vector in the body-axis frame (ref. 9). The specific force vector is calculated by summing the perturbation linear accelerations  $\Delta F_X/m$ ,  $\Delta F_Y/m$ , and  $\Delta F_Z/m$  generated in the force and moment submodule with the gravity dependent nominal trim terms; that is,

$$\vec{f}^b = \begin{bmatrix} \frac{\Delta F_X}{m} \\ \frac{\Delta F_Y}{m} \\ \frac{\Delta F_Z}{m} \end{bmatrix} - g \begin{bmatrix} -\sin \theta_N \\ \cos \theta_N \sin \phi_N \\ \cos \theta_N \cos \phi_N \end{bmatrix} \quad (19)$$

where  $\theta_N$  and  $\phi_N$  are the trim pitch and bank angles that are generated in submodule STABDER. The components of  $\vec{f}^b$  are fed back to the linear accelerometer module discussed in the IMU section.

The body-axis acceleration vector  $\dot{\omega}_{ib}^b$  relative to inertial space is approximated as

$$\left\{ \dot{\omega}_{ib}^b \right\} = \begin{bmatrix} I_1 & 0 & I_3 \\ 0 & 1 & 0 \\ I_2 & 0 & I_1 \end{bmatrix} \begin{bmatrix} \frac{\Delta M_X}{I_{XX}} \\ \frac{\Delta M_Y}{I_{YY}} \\ \frac{\Delta M_Z}{I_{ZZ}} \end{bmatrix} \quad (20)$$

The perturbation angular accelerations  $\Delta M_X/I_{XX}$ ,  $\Delta M_Y/I_{YY}$ , and  $\Delta M_Z/I_{ZZ}$  calculated in the forces and moments submodule are proportional to the total moments acting on the helicopter (ref. 2). The inertia products  $I_1$ ,  $I_2$ , and  $I_3$  are functions of the moments of inertia  $I_{XX}$ ,  $I_{YY}$ , and  $I_{ZZ}$  and product of inertia  $I_{XZ}$  in the helicopter XZ-plane as follows:

$$I_1 = \frac{I_{XX}I_{ZZ}}{I_{XX}I_{ZZ} - (I_{XZ})^2} \quad (21a)$$

$$I_2 = \frac{I_1 I_{XZ}}{I_{ZZ}} \quad (21b)$$

$$I_3 = \frac{I_1 I_{XZ}}{I_{XX}} \quad (21c)$$

The three integral components of equation (20) are fed back to the rate gyro module discussed in the IMU section.

The Euler angle rates, using a yaw  $\psi_H$ , pitch  $\theta_H$ , and roll  $\phi_H$  sequence order are

$$\begin{bmatrix} \dot{\psi}_H \\ \dot{\theta}_H \\ \dot{\phi}_H \end{bmatrix} = \begin{bmatrix} 0 & \frac{\sin \phi_H}{\cos \theta_H} & \frac{\cos \phi_H}{\cos \theta_H} \\ 0 & \cos \phi_H & -\sin \phi_H \\ 1 & \sin \phi_H \tan \theta_H & \cos \phi_H \tan \theta_H \end{bmatrix} \left\{ \vec{\omega}_{nb}^b \right\} \quad (22)$$

The three integral components of equation (22) represent the Euler angles that relate the body frame to the local level geographic frame and are fed back to the attitude module discussed in the IMU section. The angular velocity vector  $\vec{\omega}_{nb}^b$  of the helicopter relative to the local level geographic frame is related to the difference between the body rates relative to inertial space and the angular velocity of the local level frame relative to the Earth fixed frame; that is,

$$\vec{\omega}_{nb}^b = \vec{\omega}_{ib}^b - C_n^{b,n} \vec{\omega}_{en}^n \quad (23)$$

The rate of change of the helicopter latitude  $L_H$ , longitude  $\lambda_H$ , and altitude  $Z_H$  relative to the Earth is

$$\begin{bmatrix} \dot{L}_H \\ \dot{\lambda}_H \\ \dot{Z}_H \end{bmatrix} = \begin{bmatrix} \frac{\dot{X}_H}{R_H} \\ \frac{\dot{Y}_H}{R_H \cos L_H} \\ -\dot{R}_H \end{bmatrix} \quad (24)$$

where  $\dot{X}_H$  is the north component of groundspeed,  $\dot{Y}_H$  is the east component of groundspeed, and  $R_H$  (with  $\dot{R}_H$  its derivative) is the vehicle altitude relative to the mass center of the Earth. The altitude  $R_H$  is a function of Earth radius  $R_e$  and helicopter altitude  $Z_H$  above the Earth surface as

$$R_H = R_e - Z_H \quad (25)$$

The Earth radius is assumed to be a constant of 6378.1631 km (20 925 732 ft). The integral of equation (24) gives the Earth related coordinates. In this program, latitudes are positive in the northern hemisphere and longitudes are negative in the western hemisphere.

### Updates

In this section, all of the helicopter variables are updated for the succeeding time period and for distribution to all of the other modules in the simulation. Body referenced helicopter variables include both linear and angular accelerations, linear and angular velocities, attitudes, and angles of attack



and sideslip. Earth referenced helicopter variables include linear accelerations, velocities, and positions. Coefficients of a direction cosine matrix are also available for use in other modules.

### Helicopter Initialization

The input data for the helicopter module consist of the iteration frequency (default value is 32 iterations per second) and three sets of three initial errors (default values are zero). The first set is referenced to the first waypoint location and includes along-track position error, cross-track position error, and altitude error; the second set is referenced to the desired velocities and heading at the first waypoint and includes groundspeed error, ground-track error, and altitude rate error; and the third set is body referenced and includes pitch attitude error, bank attitude error, and heading error.

The airspeed vector is calculated by summing the groundspeed and wind vectors. The airspeed magnitude  $\dot{X}_{a,V,H}$  is

$$\dot{X}_{a,V,H} = k \left[ (\dot{X}_{V,H})^2 + W^2 + 2\dot{X}_{V,H}W \cos (\eta - \psi_0) \right]^{1/2} \quad (26)$$

where

$$k = \begin{cases} +1, & (\dot{X}_{V,H} + W \cos (\eta - \psi_0)) > 0 \\ -1, & (\dot{X}_{V,H} + W \cos (\eta - \psi_0)) < 0 \end{cases}$$

and the heading of the airspeed vector  $\psi_H$  relative to north is

$$\psi_H = \psi_0 + \sin^{-1} \left[ \frac{W}{\dot{X}_{a,V,H}} \sin (\eta - \psi_0) \right] \quad (27)$$

where  $\dot{X}_{V,H}$  is the groundspeed,  $W$  is the wind magnitude,  $\psi_0$  is the initial ground track, and  $\eta$  is the wind direction relative to north. Any initial heading error is added to equation (27).

The airspeed  $\dot{X}_{a,V,H}$  calculated in equation (26) is used in the stability derivative and trim lookup table to find the equilibrium pitch and bank attitudes. Initial pitch and bank attitude errors are then added to the equilibrium values to determine the initial helicopter attitudes. A complete set of derivatives and trims is also determined at this time. Other calculations made in this module include the three specific forces and the three body rates relative to inertial space. Finally, the 12 differential equations described in the equations-of-motion section are initialized. These equations include the three groundspeed components, three body rates, three attitudes, and three position components.

## Stability Derivatives (STABDER)

The purpose of STABDER is to determine the stability and control derivatives and trim nominals during flight, at any operating point in the flight regime. The module is essentially composed of two parts; the first part determines the operating points at which to calculate the derivatives and trims, and the second part is the calculation of the derivatives and trims. The flight regime of the CH-47B helicopter includes airspeeds between -40 and 160 knots and rates of descent between  $\pm 10.16$  m/sec ( $\pm 2000$  ft/min). The nominal operating point is a function of both the commanded helicopter velocities (airspeed and rate of descent) and actual helicopter velocities and is always maintained within predefined limits of the actual helicopter velocities. When the difference between the actual and commanded velocities is less than the predefined limits, the commanded velocities are chosen for the nominal operating point. The assumption is that for a stable vehicle the helicopter velocity will converge to the commanded value. When the difference between the actual and commanded velocities is greater than the predefined limits, the nominal operating point is chosen as the sum of the actual velocity and the predefined limit. The airspeed limit is chosen as 9.6 knots and the rate of descent limit is 1.5 m/sec (300 ft/min). These limits are defined in DATA statements and can easily be changed. The operating point is determined every second, and is also set in a DATA statement. If the nominal operating point does not change during the 1-sec interval, the previously calculated derivatives and trims are used.

For the uncoupled system, there are 48 derivatives and trims to calculate each time the nominal operating point is changed. Furthermore, for each derivative and trim there are 99 data points; this includes 11 airspeeds at 9 rates of descent. In order to save storage, third-order curves are fitted through each of the 11 airspeed data points. This approach allows the storage to be reduced from  $4752_{10}$  to  $1728_{10}$  locations. For each derivative and trim, the third-order algebraic equation is first calculated as a function of airspeed; linear interpolation is then applied, using rate of descent as the independent variable. Initial condition and step input responses of the open-loop helicopter model show that the third-order curves have approximately the same dynamic responses as those obtained by using all of the derivatives.

## PREPROCESSOR

The simulation program has the capability of selecting various combinations of models for any run from a choice of many different model types, thereby minimizing the in-core storage and computation time. The feature that allows this capability is the preprocessor. In general, the preprocessor is a FORTRAN program that performs modifications on the main program prior to compilation and execution of the main program. The most obvious function of the preprocessor is to decide upon the appropriate models to be used and to write the correct CALL statements into the computer program. Model selection is accomplished by a combination of defaults and input data cards. Specific functions include: (1) writing subroutine calls within the mainline and within other subroutines, (2) choosing appropriate initialization models and writing the correct CALL statements within the initialization routine, (3) writing the necessary COMMON blocks for on-line plotting, (4) calculating the total number of differential

equations to be solved in the simulation and writing the COMMON blocks with proper array dimensions, (5) writing specific CALL statements to the real-time system for pseudo real-time runs, and (6) printing an alphabetical table of all subroutines used in a specific run and the total number of differential equations in the simulation.

The means for determining the correct subroutine CALL statement is through a module integer number (KEY 1) and a model integer number (KEY 2). Every major selectable module and submodule in the simulation program have an assigned KEY 1 integer (10, 20, 30, etc.). The model choices within a module have assigned one-digit numbers (0, . . . , 9), allowing a possibility of up to 10 model choices. The sum of KEY 1 and KEY 2 relates to an array SNAM and defines the particular subroutine to be called.

A general flow chart is illustrated in figure 4. Default values are first stored in all arrays and then are overridden by user input data cards. Input

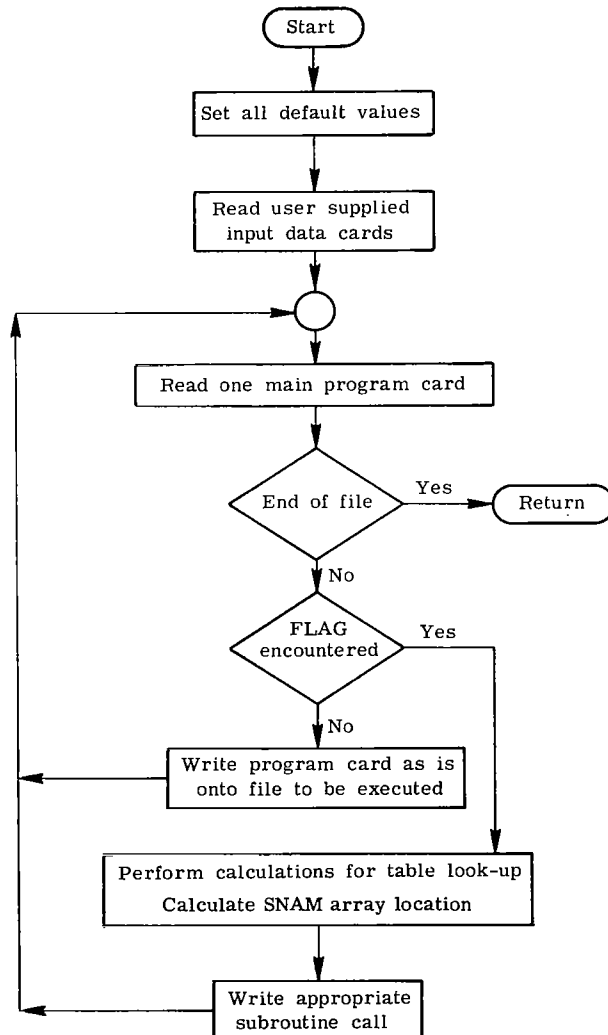


Figure 4.- General flow chart for preprocessor.

data cards are only required if optional subroutines are to be chosen. The unprocessed main program has the word "FLAG" in columns 1 to 4, wherever a selectable subroutine is to be called. The preprocessor checks each statement in the main program. If the word "FLAG" or one of the other special indicators (not defined here) for function 3, 4, or 5 defined previously is not encountered, the statement is written onto the file to be executed. If the word "FLAG" is encountered, the preprocessor checks the very next statement for an index integer number. The index number corresponds to the correct module (or submodule) to be called. Using this index number along with KEY 1 and KEY 2 data, the SNAM array location, corresponding to the correct subroutine to be called, is calculated. The preprocessor then writes this subroutine call onto the file to be executed. This process is continued until an end of file is detected.

Input data cards are only used when the default cases are not desired. Each data card includes three numbers: the first corresponds to the KEY 1 integer corresponding to the module being selected, the second corresponds to the KEY 2 integer corresponding to the model being selected, and the third integer is the number of differential equations in the model. For example, a data card with integers 200, 1, and 7 indicates a local level navigator module, model number 2, with 7 differential equations. Two problem examples for the preprocessor are illustrated in the next section of this paper. Presently, 39 modules have been defined for the batch program and 44 modules, for the real-time version.

The preprocessor has the capability of writing subroutine calls within subroutines as well as the mainline. The process is the same as previously described. However, there are additional considerations involved. For example, a subroutine called within another subroutine need not be added to the subroutine library if it has already been called in the mainline or in a previous subroutine. The preprocessor insures that a subroutine will only be added to the final library once, regardless of the number of times it has been called. The preprocessor will handle a subroutine call depth of up to four levels. A tree structure is used to keep track of the call depth levels and to determine which subroutines have been called and added to the library.

## PROGRAM ORGANIZATION

### Deck Structure

The deck configuration consists of five records plus the system control cards, as shown in figure 5. The first three records are used for modifications to key parts of the overall program; the first record is for modifications to the stability derivative curve coefficients, the second record is to modify the mainline and subroutine library, and the third record is to modify the preprocessor. The preprocessor input data cards, discussed in the preprocessor section and illustrated in the problem examples section, are located in the fourth record. The fifth record is for the main program input data cards, which are illustrated in the problem examples section.

System control cards

EOR

Mods for stability derivative curve coefficients

EOR

Mods for mainline and subroutine library

EOR

Mods for preprocessor

EOR

Preprocessor data cards

EOR

Main program data cards

EOF

Figure 5.- Control card deck structure. (EOR is end of record and EOF is end of file.)

### Problem Examples

This section contains some examples illustrating the input data cards for the preprocessor and the main program. The preprocessor data are field oriented, whereas all main-program input data are in the form of NAMELIST statements. The only requirement is that all NAMELIST cards be present in the data deck, even though all of the FORTRAN variables defined under a NAMELIST statement have defaulted values. Trajectory type data are the only input data that must be submitted.

The first preprocessor example illustrates the default option and shows the minimum input data required. The example case is to use all default modules. The preprocessor data deck consists of one card with a "0" in column 5, indicating that no other cards are required.

The second preprocessor example represents a more typical case. It is desired to have error models for the navigator (LLNAV2), rate gyros (RATEG2), and linear accelerometers (LINAC2) and to use the second wind model (EXTWIN2). The subroutine call names are in parentheses. All other modules are defaulted





1. Define the nominal trajectory:

Waypoint	Latitude, deg	Longitude, deg	Altitude, m (ft)	Velocity, m/sec (ft/sec)
1	40.64	-74.067	-426.7 (-1400)	30.48 (100)
2	40.64	-74.04	-335.3 (-1100)	24.38 (80)
3	40.66	-74.025	-243.8 (-800)	19.8 (65)

2. Assume initial helicopter errors:

Along track error, m (ft)	-30.48 (-100)
Altitude error, m (ft)	3.040 (10)
Velocity error, m/sec (ft/sec)	0.9144 (3)

3. Assume a linear accelerometer bias error of 0.003048 m/sec<sup>2</sup> (0.01 ft/sec<sup>2</sup>).

4. Assume a rate gyro scale factor error of 2 percent.

5. Assume the navigator has a north platform misalignment angle of 1  $\mu$ rad.

6. Assume a wind shear with two data points:

Point	Altitude, m (ft)	Wind direction, deg	Wind velocity, m/sec (ft/sec)
1	0 (0)	180	0 (0)
2	304.8 (1000)	235	6.1 (20)

7. Assume all default frequencies except:

Accelerometer module, per sec	32
Navigator module, per sec	32
Printout, per sec	0.1
Maximum run time, sec	20

Obviously, the run time for this example will complete only a very minor portion of the desired trajectory.

8. Print the following data:

- All initialization data
- Rate gyro outputs
- Linear accelerometer outputs
- Navigator outputs (option 2)
- Guidance commands (option 1)
- Controller outputs
- All helicopter outputs

Figure 7 illustrates the input data cards for this example. Eighteen NAMELIST data cards are shown; the real-time version has three additional input







cards. The NAMELIST card DECNTS describes the number of differential equations for each module in the simulation run, and the NAMELIST card PVAR describes the print variable options for each module. All other data in figure 7 can be correlated with numbers in the example. The U.S. Customary Units are used in the computer program. A computer printout for the second preprocessor example and the main program example is shown in the appendix.

The computer example was run on a Control Data Corporation (CDC) Cyber 175 computer, with a CDC FORTRAN FTN compiler (version 4.6). Version 1.1 of the NOS operating system was in use. The actual central processor times for compiling, loading, and executing the various sections of the computer program are summarized as follows:

Compile preprocessor, sec . . . . .	2.043
Load and execute preprocessor, sec . . . . .	2.951
Compile main program, sec . . . . .	1.086
Compile subroutine library, sec . . . . .	7.959
Load and execute complete program, sec . . . . .	3.819

The actual storage needed for this example is 50 600 octal words.

#### CONCLUDING REMARKS

A user-oriented computer-simulation program has been developed to allow for the evaluation of individual avionics systems in conjunction with a VTOL aircraft model and other types of VTOL avionics systems. After a model has been developed by any user, the subroutine is incorporated into the system library for use by all of the researchers. The input data for the main program are defined in NAMELIST statements. A series of defaults has been incorporated to minimize the amount of input data needed for any simulation run.

The modular approach allows many different models to be developed and installed into the subroutine library, with minimum impact on the total system. Individual modules for all of the basic avionics sensors allow a large choice of measurements and flexibility.

A preprocessor has been developed to allow a large selection of models. For any module, a possibility of up to 10 models exists. Default models allow a minimum of input data cards for the preprocessor.

A real-time program is maintained as nearly similar as possible with the batch program. By using the preprocessor capability, a dummy model with zero input frequency can be selected by input data cards; thereby, the software is eliminated from storage. The software can be replaced by actual hardware that operates through the electrical input/output channels.

Langley Research Center  
National Aeronautics and Space Administration  
Hampton, VA 23665  
February 9, 1977

## APPENDIX

## COMPUTER PRINTOUT FOR PROBLEM EXAMPLE

This appendix contains the computer printout for the problem example. The first page of the printout lists all of the subroutines selected by the preprocessor and indicates the total number of differential equations for the example. All other pages are related to the main program. Initialization data are included on the first 3-1/2 pages; this includes the iteration frequencies of all modules, trajectory data, helicopter data, trim data, and all error model initialization data. The completion of the printout contains data from the modules selected at the iteration frequency.

## VALT/FTD BATCH SIMULATION

THE FOLLOWING SUBROUTINES WERE USED IN THIS RUN:

ABSALT	ACTOYN	ACTPRT	ATPDATA	AIRSPD	ALTHOLD	AMOVE	ANGAC	ATCMOD	BODYFOS
CHSUB	CLIMB	COLWIND	CONTROL	ETRAN	EXTWIN2	FANDM	GCOUPLE	GENEQS	GSLAW1
GSPEED	GUIDPRT	GUPDATE	HEDEFF	HELCON	HELICOP	HYSTER	IARSAL1	IAIRDA1	IANGAC1
IATCMO1	ICNTR1	IEXWIN2	IGCOUP1	IGIDPT1	IGSLAW1	IGSPED1	IHDREF1	IHELICP	IKALMN1
ILINAC2	ILLNAV2	INTER	INTTRAJ1	IPILDT1	IRATEG2	IRIGYD1	IXNAV1	IXNAV2	IXNAV3
IXNAV4	IXNAV5	IXNAV6	IXNAV7	IXNAV8	IXNAV9	KALMAN1	LATTURN	LEGSWCH	LINAC2
LLNAV2	MEANWIN	NAVFOR	NMTRAJ1	NRAN2	ONPLOT	OUTPUT	PILOT	PLOTIT1	PLTSAV1
POSEST	PSLIM	PSITION	PATFG2	PATELIM	RDINPT	RIGYR01	ROTDYN	SPESTRY	SPEXIT
SPGUID	SPTRAL1	SPTRGID	SPWTND	STABDER	SVARDES	TIMVARS	TRKDIST	UPDATE	VELCOM
XNAV1	XNAV2	XNAV3	XNAV4	XNAV5	XNAV6	XNAV7	XNAVR	XNAV9	ZHELICP
ZSTAR									

THE TOTAL NUMBER OF DIFFERENTIAL EQUATIONS IN THIS RUN IS: 19

INITIALIZATION SECTION

HELICOPTER INERTIAS (SLUG-FEET SQ.)

IXX .3716300E+05	IYY .2017500E+06	IZZ .1900590E+06	JXZ .1463200E+05
---------------------	---------------------	---------------------	---------------------

FREQUENCIES IN ITERATIONS PER SECOND

HELICOPTER	.32000E+02
RATE INTEGRATING GYRO	.32000E+02
RATE GYRO	.32000E+02
LINEAR ACCELEROMETER	.32000E+02
ANGULAR ACCELEROMETER	0.
AIR DATA	0.
GROUNDSPEED	0.
HEADING REFERENCE	0.
ABSOLUTE ALTITUDE	0.
EXTERNAL NAV. AID 1	0.
EXTERNAL NAV. AID 2	0.
EXTERNAL NAV. AID 3	0.
EXTERNAL NAV. AID 4	0.
EXTERNAL NAV. AID 5	0.
EXTERNAL NAV. AID 6	0.
EXTERNAL NAV. AID 7	0.
EXTERNAL NAV. AID 8	0.
EXTERNAL NAV. AID 9	0.
LOCAL LEVEL NAV.	.32000E+02
KALMAN FILTER	.16000E+02
NOMINAL TRAJECTORY	.20000E+01
GUIDANCE STEERING LAW	.20000E+01
AIR TRAFFIC CONTROL	0.
PILOT TRAJECTORY	.20000E+01
GUIDANCE PERTURBATIONS	.16000E+02
GUIDANCE COUPLER	.16000E+02
CONTROLLER	.16000E+02
PRINTOUT	.10000E+00

MAXIMUM RUN TIME = .20000E+02 SECONDS

INTEGRATION TIME INTERVAL IS .31250E-01 SECONDS

RANDOM NUMBER GENERATOR SEED = 59.000

APPENDIX

WAYPOINT	MEAN HORIZONTAL WIND ALTITUDE (FT.)		WINDS		SPEED (FT./SEC.)	TRACK ANGLE (DEG)	DISTANCE (FT)
	LAT. (DEG)	VEL. (FT/SEC)	DIRECTION (DEG.)	NOM. (DEG)			
DESIPED					0. .2000000E+02		
1	.4064000E+02	.1000000E+03	-.7406700E+02	0. -.1400000E+04	0. .9000000E+02	0.	.7482707E+04
2	.4064000E+02	.8000000E+02	-.7404000E+02	0. -.1100000E+04	0. .2964106E+02	0.	.8404230E+04
3	.4066000E+02	.6500000E+02	-.7402500E+02	0. -.8000000E+03	0.	I 0.	

HELICOPTER DATA

GEOGRAPHIC FRAME VARIABLES

LATITUDE (DEG)	LONGITUDE (DEG)	ALTITUDE (FT)		
.4064000E+02	-.7406736E+02	-.1390000E+04		
N.VEL.(FT/SEC)	E.VEL.(FT/SEC)	TOTAL VEL.(FT/SEC)	GROUND TRACK (DEG)	DESCENT RATE (FT/SEC)
-.1766578E-12	.1030000E+03	.1030000E+03	.9000000E+02	0.

BODY AXIS VARIABLES

LONG.VEL.(FT/SEC)	LATERAL VEL.(FT/SEC)	VERT.VEL.(FT/SEC)
.1018998E+03	-.1356007E+02	.6445725E+01
PITCH ANGLE (DEG)	ROLL ANGLE (DEG)	YAW ANGLE (DEG)
.3662560E+01	-.3257091E+00	.9754433E+02
ANGLE OF ATTACK (DEG) =	.3662501E+01	
ANGLE OF SIDESLIP (DEG) =	-.2084877E-01	

TRIM DATA

GROUNDSPEEDS

U NOM(FT/SEC)	V NOM(FT/SEC)	W NOM(FT/SEC)	UA NOM(FT/SEC)	VA NOM(FT/SEC)	WA NOM(FT/SEC)
.9892233E+02	-.1356027E+02	.6416447E+01	.8421890E+02	-.3174507E-01	.5528836E+01

AIRSPEDS

ATTITUDES

THETA NOM (DEG)	PHI NOM (DEG)
.3756048E+01	-.3289732E+00

RATE GYROS

	PITCH	ROLL	YAW
BIAS (RAD./SEC.)	0.	0.	0.
RANDOM (RAD./SEC.)	0.	0.	0.
SCALE FACTOR	.1000396E+01	.1018096E+01	.9587966E+00

APPENDIX

LINEAR ACCELEROMETERS

---

	LONG.	LATERAL	VERTICAL
BIAS (FT./SEC.SQ.)	-.1169067E-01	-.1213770E-01	.2172295E-02
RANDOM (FT./SEC.SQ.)	0.	0.	0.

LOCAL LEVEL NAVIGATOR

---

MISALIGNMENT ERRORS

NORTH (RAD)	EAST (RAD)	DOWN (RAD)
.10000E-05	0.	0.

POSITION ERRORS

LAT (RAD)	LONG (RAD)	ALT (FT)
0.	0.	0.

VELOCITY ERRORS

LAT RATE (RAD/SEC)	LONG RATE (RAD/SEC)	VERTICAL (FT/SEC)
0.	0.	0.

RATE GYRO OUTPUTS

PI = -.37517E-06  
 OI = -.49428E-05  
 PI = -.41083E-05

TIME = 0.  
 .....

LINEAR ACCELEROMETER OUTPUTS

UDDTI = .20977E+01  
 VDDTI = .17235E+00  
 WDDTI = -.32128E+02

TIME = 0.  
 .....



LOCAL LEVEL NAVIGATOR

MISALIGNMENT ERRORS

NORTH (RAD) EAST (RAD)

DOWN (RAD)

POSITION ERRORS

LAT (RAD) LONG (RAD)

VELOCITY ERRORS

ALT (FT)

LAT RATE (RAD/SEC)

LONG RATE (RAD/SEC)

VERTICAL (FT/SEC)

.10000E-05 0. 0. 0. 0. I 0. 0. I

FINAL GUIDANCE COMMANDS

TIME = 0. SEC  
 XDOTGSC (FT/SEC) = .99500E+02 PSIHRC (DEG) = .90000E+02 ZDOTGSC (FT/SEC) = .86250E+00  
 ZGSC (FT) = .13997E+04  
 PHIGSC (DEG) = 0. PSIGSC (DEG) = .90000E+02

CONTROLLER OUTPUTS (INCHES)

DCC(1) DCC(2) DCC(3) DCC(4)  
 .58327E+00 -.23786E+00 .10400E+00 -.22499E+01

TIME = 0.

APPENDIX

HELICOPTER VARIABLES AT TIME = 0.

POSITIONS

XH = 0.                      YH = 0.                      ZH = -.139000E+04                      FEET  
 LATH = .406400E+02                      LBNH = -.740674E+02                      DEGREES

NAV. GROUND SPEEDS (FT/SEC)

XDOTH = -.176658E-12                      YDOTH = .103000E+03                      ZDOTH = 0.                      XDOTVH = .103000E+03

BODY VELOCITIES (FT/SEC)

UH = .101900E+03                      VH = -.135601E+02                      WH = .644573E+01

LINEAR ACCELERATIONS (FT/SFC.SQ.)

BUDDTH = 0.                      VDDTH = 0.                      DWDDTH = 0.

EULER ANGLES (DEG)

THEH = .366256E+01                      PHIH = -.325708E+00                      PSIH = .975443E+02

BODY RATES (RADIAN/SEC)

PH = -.375024E-06                      OH = -.485496E-05                      RH = -.428481E-05

ANGULAR ACCELERATIONS (RADIAN/SFC.SQ.)

PDDTH = 0.                      ODDTH = 0.                      RDDTH = 0.

ANGLES OF ATTACK, SIDESLIP (RADIAN)

ALPHAH = .639227E-01                      BETAH = -.363880E-03

RATE GYRO OUTPUTS

PI = -.32159E-03  
 OI = .24612E-02  
 RI = .85570E-03

TIME = .10000E+02

LINEAR ACCELEROMETER OUTPUTS

UDDTI = .23562E+01  
 VDDTI = -.59122E+00  
 WDDTI = -.32061E+02

TIME = .10000E+02

LOCAL LEVEL NAVIGATOR

MISALIGNMENT ERRORS

NORTH (RAD)      EAST (RAD)

-.70694E-03

.45655E-03

DOWN (RAD)

.55168E-02

POSITION ERRORS

LAT (RAD)

.42041E-07

LONG (RAD)

.42910E-07

ALT (FT)

VELOCITY ERRORS

LAT RATE (RAD/SEC)

I .90960E-08

LONG RATE (RAD/SEC)

.91608E-08

VERTICAL (FT/SEC)

I

FINAL GUIDANCE COMMANDS

TIME = .10000E+02 SEC

XDOTGSC (FT/SEC) = .89500E+02

PSIHRC (DEG) = .90000E+02

ZDOTGSC (FT/SEC) = .98849E+01

ZGSC (FT) = -.13150E+04

PHIGSC (DFG) = 0.

PSIGSC (DEG) = .90000E+02

CONTROLLER OUTPUTS (INCHES)

DCC(1)

.29038E-02

DCC(2)

-.65522E-01

DCC(3)

.12433E+00

DCC(4)

-.17019E-01

TIME = .10000E+02

HELICOPTER VARIABLES AT TIME = .10000000E+02

POSITIONS

XH = .164645F+02 YH = .969728E+03 ZH = -.131212E+04 FEET  
 LATH = .406400F+02 LONH = -.740639E+02 DEGREES

NAV. GROUND SPEEDS (FT/SEC)

XDOTH = .205792F+01 YDOTH = .912002E+02 ZDOTH = .969572E+01 XDOTVH = .912234E+02

BODY VELOCITIES (FT/SEC)

UH = .897999F+02 VH = -.129147E+01 WH = .187092E+02

LINEAR ACCELERATIONS (FT/SEC.SQ.)

DUDOTH = -.871640F+00 VDOTH = -.762282E-01 DWDOTH = .187697E+00

EULER ANGLES (DEG)

THEH = .568452F+01 PHTH = .105311E+01 PSIH = .897344E+02

BODY RATES (RADIAN/SEC)

PH = -.321459E-03 OH = .241743E-02 RH = .892471E-03

ANGULAR ACCELERATIONS (RADIAN/SEC.SQ.)

PDOTH = .246221F-03 ODOTH = -.100041E-01 RDOTH = -.246543E-03

ANGLES OF ATTACK, SIDESLIP (RADIAN)

ALPHAH = .225824E+00 BETAH = .136284E+00

RATE GYRO OUTPUTS

PI = -.22453E-04  
 OI = -.66496E-05  
 RI = .11453E-03

TIME = .20000E+02

LINEAR ACCELEROMETER OUTPUTS

UDOTI = .25119E+01  
 VDOTI = -.57249E+00  
 WDOTI = -.32019E+02

TIME = .20000E+02

LOCAL LEVEL NAVIGATOR

MISALIGNMENT ERRORS

NORTH (RAD)      EAST (RAD)

-.89806E-03      .44328E-03

DOWN (RAD)

.53459E-02

POSITION ERRORS

LAT (RAD)

.18388E-06

LONG (RAD)

.19259E-06

ALT (FT)

VELOCITY ERRORS

LAT RATE (RAD/SEC)

I .19219E-07

LONG RATE (RAD/SEC)

.19229E-07

VERTICAL (FT/SEC)

I

FINAL GUIDANCE COMMANDS

TIME = .20000E+02 SEC

XDOTGSC (FT/SEC) = .80000E+02

PSIHRC (DEG) = .90000E+02

ZDOTGSC (FT/SEC) = .99927E+01

ZGSC (FT) = -.12150E+04

PHIGSC (DEG) = 0.

PSIGSC (DEG) = .90000E+02

CONTROLLER OUTPUTS (INCHES)

DCC(1)

-.97428E-01

DCC(2)

-.85710E-01

DCC(3)

.12272E+00

DCC(4)

-.16618E-01

TIME = .20000E+02

APPENDIX

HELICOPTER VARIABLES AT TIME = .20000000E+02

POSITIONS

XH = .366779F+02 YH = .183286E+04 ZH = -.121482E+04  
 LATH = .406401F+02 LONH = -.740607F+02

FEET  
DEGREES

NAV. GROUND SPEEDS (FT/SEC)

XDOOTH = .200644F+01 YDOOTH = .813050E+02 ZDOOTH = .979245E+01 XDOOTVH = .813297E+02

BODY VELOCITIES (FT/SEC)

UH = .797482F+02 VH = -.162865F+01 WH = .186543E+02

LINEAR ACCELERATIONS (FT/SEC.SQ.)

DUOOTH = -.998102E+00 VDOOTH = -.118879E-01 DWDOOTH = -.198721E-01

EULER ANGLES (DEG)

THEH = .627673F+01 PTHH = .999730E+00 PSIH = .899630E+02

BODY RATES (RADIAN/SEC)

PH = -.224436E-04 OH = -.653138E-05 RH = .119457E-03

ANGULAR ACCELERATIONS (RADIAN/SEC.SQ.)

PDOOTH = .931936E-04 ODOOTH = -.113031E-01 RDOOTH = -.341376E-04

ANGLES OF ATTACK, SIDESLIP (RADIAN)

ALPHAH = .256789F+00 BETAH = .153215E+00

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