A Mathematical Model of a Single Main Rotor Helicopter for Piloted Simulation

Peter D. Talbot, Bruce E. Tinling, William A. Decker, and Robert T. N. Chen

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A MATHEMATICAL MODEL OF A SINGLE MAIN ROTOR HELICOPTER
FOR PILOTED SIMULATION

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SUMMARY

This report documents a helicopter mathematical model suitable for piloted simulation of flying qualities. The mathematical model is a nonlinear, total force and moment model of a single main rotor helicopter. The model has ten degrees of freedom: six rigid-body, three rotor-flapping, and the rotor rotational degrees of freedom. The rotor model assumes rigid blades with rotor forces and moments radially integrated and summed about the azimuth. The fuselage aerodynamic model uses a detailed representation over a nominal angle of attack and sideslip range of ±15°, and it uses a simplified curve fit at large angles of attack or sideslip. Stabilizing surface aerodynamics are modeled with a lift curve slope between stall limits and a general curve fit for large angles of attack. A generalized stability and control augmentation system is described. Additional computer subroutines provide options for a simplified engine/governor model, atmospheric turbulence, and a linearized six-degree-of-freedom dynamic model for stability and control analysis.

INTRODUCTION

An expanded flying-qualities data base is needed for use in developing design criteria for future helicopters. A safe and cost-effective way to establish such a data base is to conduct exploratory investigations using piloted ground-based simulators, and then to substantiate the results in flight using variable stability research helicopters.

A mathematical model suitable for real-time piloted simulation of single rotor helicopters has been implemented at Ames Research Center. As described in reference 1, simulation models used at Ames Research Center consist of a common core of rigid-body equations and an aerodynamic model that provides the aerodynamic forces and moments. This report documents the equations used in the aerodynamic model.

The report consists of a brief description of the overall model and its components and appendices that detail the equations used in the model and the parameters required to describe a helicopter configuration.

SIMULATION MODEL

The overall arrangement of the simulation model is shown in figure 1. The principal assumptions and considerations employed in developing each element of the model are given in the main body of the report; detailed equations for the forces and moments are given in the appendices. The model elements, denoted Ti in figure 1, are required to achieve transfer of velocities, forces, and moments from
Figure 1.— Block diagram showing principal elements of single rotor helicopter model.
one axis system to another; in some instances, to account for aerodynamic interference effects between model components. These elements are described in the appendices where required.

The notation employed is listed in appendix A, and a description of the various axis systems is given in appendix B. The variable names in the FORTRAN coding for the computer program have been chosen to be easily identifiable from the notation used herein. Through this mnemonic device, equations in the appendix can be identified in the computer program listings.

Main Rotor

The development of the equations describing the dynamics and the forces and moments acting on the main rotor are given in detail in references 2 and 3. This mathematical representation explicitly accounts for the dynamic effect of rotor modes, such as rotor-blade flapping, which can be in a frequency range which is important in studies of flying qualities. For the rotor model described in this report, the flapping dynamics were approximated using a tip-path plane representation.

The flapping equation of motion of the rotor blade was first developed using the following assumptions. The assumptions are similar to those used for the "classical" equations (refs. 4 and 5).

1. The rotor blade was rigid in bending and torsion, and the twist of the blade was linear.

2. The flapping angle and inflow angle were assumed to be small and the analysis utilized a simple strip theory.

3. The effects of the aircraft motion on the blade flapping were limited to those due to the angular acceleration \( \dot{\phi} \) and \( \dot{\theta} \), the angular rate \( \dot{\phi} \) and \( \dot{\theta} \), and the normal acceleration.

4. The reversed flow region was ignored and the compressibility and stall effects disregarded.

5. The inflow was assumed to be uniform and no inflow dynamics were used.

6. The tip-loss factor was assumed to be 1.

The flapping equations of motion explicitly contain the primary design parameters, namely: flapping hinge restraint, hinge offset, blade Lock number, and pitch-flap coupling. The blade flapping in those equations was then approximated by the first harmonic terms with time-varying coefficients, that is,

\[ \beta(t) = a_0(t) - a_1(t) \cos \psi - b_1(t) \sin \psi. \]

In the development of the equations for forces and moments, the same set of basic assumptions (1 through 7 above), discussed in conjunction with the development of the tip-path plane dynamic equations, was utilized. Thus, aerodynamically, momentum theory was used in conjunction with the uniform inflow; simple strip theory was utilized and the blade forces were analytically integrated over the radius. Because the reversed flow region and the stall and compressibility effects were ignored, the total rotor forces and moments were again analytically obtained by
summing the contributions, to each blade, that were analytical functions of the azimuth. Because of these assumptions and simplifications, the results of the analysis are valid only for a limited range of flight conditions. Nevertheless, a previous study (ref. 5) has shown that this type of analysis is valid for stability and control investigations of the rotorcraft up to an advance ratio of about 0.3. Also, similar to the development of the tip-path plane dynamic equations, these rotor forces and moments were first obtained in the wind-hub coordinate system. They were then transformed into the hub-body system (see appendix C).

The forces and moments thus developed contain periodic terms; the highest harmonic terms correspond directly to the number of rotor blades. For example, for a three-bladed rotor, the force and moment equations contain only three/revolution harmonic terms, and for a four-bladed rotor, four/revolution harmonic terms. The frequency of these harmonic terms is sufficiently high to be of no interest to handling quality investigations. These terms have therefore been deleted. The resulting force and moment expressions are given in appendix C.

A development to modify these forces and moment equations to include the effects of nonuniform inflow, similar to the development for the flapping equations in references 6 and 7, is in progress. These equations, along with the modified tip-path plane representation given in reference 7 will later supersede those shown in appendix C.

Tail Rotor

The tail rotor was modeled as a teetering rotor without cyclic pitch. For this case, the forces in the wind-hub system may be obtained from the expressions derived for the main rotor by setting the lateral and longitudinal cyclic pitch terms \( A_{1c} \) and \( B_{1c} \) equal to zero. Further, since the tail rotor flapping frequency is much higher than that of the main rotor system, the tip-path plane dynamics may be neglected. Thus, for tail rotor applications, the first and second derivatives of the blade-flapping nonrotating coordinates are set equal to zero in the force equations \( \ddot{\alpha}_0 = \ddot{\alpha}_1 = \ddot{\beta}_1 = 0 \) and \( \dot{\alpha}_0 = \dot{\alpha}_1 = \dot{\beta}_1 = 0 \). The result is a set of basic quasi-static force expressions similar to those in classical work (refs. 3 and 4).

The local flow at the tail rotor includes the effect of downwash from the main rotor system. The method employed to estimate this downwash and the equations for tail rotor forces are given in appendix D.

Empennage

The lift and drag forces on the vertical fin and horizontal tail are approximated for all angle of attack and sideslip, including rearward flight. Provision is made for the addition of terms due to main rotor-induced velocities at the horizontal tail, and tail rotor velocities at the vertical fin.

The principal assumptions made in developing the expressions for the forces and moments due to the vertical fin and horizontal tail are as follows:

1. The lift and drag forces are applied at the quarter chord of each surface at the spanwise location of the center of area.
2. The airfoil profiles are symmetrical.

3. The lift curve slope prior to stall is given by simple lifting-line theory assuming an elliptical lift distribution with uniform downwash. Corrections are applied for sideslip and for sweep of the vertical fin.

4. Maximum lift coefficient is specified; however, if the lift curve slope is such that this value is not reached at an angle of attack of $\pi/4$, $C_{l_{\text{max}}}$ is assumed to occur at this angle of attack.

5. Post stall variation of lift coefficient is based on $C_{l_{\text{max}}}$ decreasing by 20% as the angle of attack is increased by 20%, and following a particular variation thereafter to reach zero lift at an angle of attack of $\pi/2$.

6. Lift coefficient in rearward flight is 80% of that in forward flight.

7. The profile drag coefficient varies with angle of attack and reaches a value of 1 when $\alpha = \pm \pi/2$.

8. The induced drag coefficient varies as the square of the calculated lift coefficient.

A typical variation of the lift and drag coefficients for an empennage surface is shown in figure 2. Expressions for calculating the empennage forces and the required transformations for velocities and forces are given in appendix E.

![Figure 2](image)

**Figure 2.** - Typical variation of empennage lift and drag coefficients.

Fuselage

The aerodynamic model of the fuselage must fulfill two requirements. The first is to provide an estimate of the forces and moments at small angles of attack and
sideslip that will be encountered at substantial forward speeds. This provides a representation of the important effects of fuselage aerodynamics on performance and stability at these speeds. The second requirement is to provide a continuous variation of forces and moments over the entire range of angle of attack and sideslip (0° to ±180°) that can be encountered in approach to hovering flight or in hover. Continuity is required to avoid sudden unrealistic linear or angular accelerations in response to a small change in attitude. Accuracy of the model at extreme attitudes is considered to be of secondary importance, since the fuselage forces at these speeds are very small compared to the rotor forces.

A technique has been developed to provide a continuous model by fitting typical variations of the forces and moments through data points obtained at specific widely separated angles of attack and sideslip in a wind tunnel (see ref. 8). However, even this sparse level of test data for the fuselage is typically unavailable and an alternative technique must be employed.

The model employed herein relies on separate representations for angles of attack and sideslip in the range from -15° to 15° and from ±30° to ±180°. Continuity is provided by a linear interpolation for forces and moments in the angle range not covered. The forces and moments for the lower angle range are obtained from test data or from estimates based on data from similar fuselages. The data for the high angle of attack and sideslip range are based on the estimated magnitude and location of the drag force vector when the fuselage is in a 90° cross flow, and on an approximation to its observed variations with attitude from wind-tunnel tests of bodies of revolution.

Details of the procedure for estimating fuselage forces and moments are given in appendix F.

Rotor Rotational Degree of Freedom and RPM Governing

An option is available which provides for a rotational degree of freedom for the rotor. When this option is used, the main rotor and the tail rotor rotational speeds vary according to the current torque requirements and the engine power available. The initial trim conditions establish baseline values of rotor speeds and engine torque. Deviations from these baseline values change the rotor torque requirements and, hence, rotor speed. These changes in speed cause the rpm governor to vary the fuel flow to the engine to change the power to maintain the desired angular rate. A block diagram of the rpm governor is shown in figure 3. Further details of the dynamic model for this degree of freedom and the rpm governor are given in appendix G.

![Figure 3.- Block diagram of rpm governor.](image-url)
Control Systems

The helicopter model has a generalized control system that accepts inputs from the pilot, facilitates control augmentation and stability augmentation, and provides for mechanical control mixing or phasing of the cyclic inputs. A block diagram of this system is shown in figure 4. The control augmentation system employs a structure that permits implementation of crossfeeds from each of the four cockpit controls; namely, longitudinal and lateral cyclic stick, collective stick, and directional pedals. The feedback gain structure of the stability augmentation system permits feedback proportional to any element of the state vector to each of the four controls. The entire control structure also facilitates gain scheduling as functions of the flight parameters, such as airspeed.

Details concerning the zero position and sign convention of the cockpit controls and the mechanical cyclic control phasing logic are given in appendix H.

Atmospheric Turbulence

The representation of atmospheric turbulence is based on the Dryden model and is described in MIL-F-8785B (ref. 9). The inputs required for this model are the aircraft velocity relative to the air mass, the turbulence scale lengths, and the rms
gust levels. For this representation, scale length and the wind velocity can be specified as functions of altitude. The rms gust levels are dependent upon wind velocity.

**Linearized Six-Degree-of-Freedom Model**

A computer subroutine is available that generates the coefficients of a linear, first-order set of differential equations that represents the rigid body dynamics of the helicopter for small perturbations from a fixed operating point. The principal assumption necessary to generate this linear set of equations is that the helicopter initial angular rates are zero. The differential equations are of the form:

$$\dot{x} = [F]x + [G]\delta$$

where \(x\) represents perturbations from trim of the state variables \(u, w, q, \theta, v, p, \phi, \) and \(r\); and \(\delta\) represents the control displacements from trim \(\Delta\delta_e, \Delta\delta_c, \Delta\delta_a,\) and \(\Delta\delta_p\). The generation of the \(F\) and \(G\) matrices and a description of each element is given in appendix I.
APPENDIX A

NOTATION

\( a \) blade lift-curve slope, per rad

\( a_0 \) blade coning angle measured from hub plane in the hub-wind axes system, rad

\( a_1 \) longitudinal first-harmonic flapping coefficient measured from the hub plane in the wind-hub axes system, rad

\( a_{1s} \) longitudinal first-harmonic flapping coefficient measured from the hub plane in the hub-body axes system, rad

\( A_{1c} \) lateral cyclic pitch measured from hub plane in the wind-wub axes system, rad

\( A_{1s} \) lateral cyclic pitch measured from hub plane in the hub-body axes system, rad

\( AR \) aspect ratio, \( \text{span}^2 / \text{area} \)

\( b_1 \) lateral first-harmonic flapping coefficient measured from hub plane in the wind-hub axes system, rad

\( b_{1s} \) lateral first-harmonic flapping coefficient measured from hub plane in the hub-body axes system, rad

\( BL \) lateral distance (buttline) in the fuselage axes system, m (ft)

\( B_{1c} \) longitudinal cyclic pitch measured from hub plane in the wind-hub axes system, rad

\( B_{1s} \) longitudinal cyclic pitch measured from hub plane in the hub-body axes system, rad

\( c \) blade chord, m (ft)

\( c_n \) control gearing constants, \( n = 1 \) to 8

\( C_{A1s} \) cyclic control rigging constants, rad

\( C_{B1s} \) cyclic control rigging constants, rad

\( C_D \) drag coefficient

\( C_L \) lift coefficient

\( C_{Lm} \) maximum lift coefficient

\( C_T \) rotor thrust coefficient, \( T/\rho(\pi R^2)\omega R^2 \)

\( D \) drag force, N (lb)
\( \hat{D} \) damping matrix in flapping differential equations

e flapping hinge offset, \( m(\text{ft}) \)

g gravitational acceleration, \( m/\text{sec}^2 \) (ft/sec^2)

H rotor force normal to shaft, positive downwind, \( N \) (lb)

i_F incidence of vertical fin, positive for leading edge to the left, rad

i_{HS} incidence of horizontal stabilizer, positive for leading edge up, rad

i_S forward tilt of rotor shaft w.r.t. fuselage, positive forward, rad

I_\beta rotor blade moment of inertia about flapping hinge, \( \text{kg-m}^2 \) (slug-ft^2)

J rotor moment of inertia about shaft

k_{v_{TR}} factor to account for fraction of vertical tail in tail rotor wake

K_B flapping spring constant, \( N-m/\text{rad} \) (lb-ft/rad)

K_1 pitch-flap coupling ratio, \( \triangleq \tan \delta \)

\( \ddot{k} \) spring matrix in flapping differential equations

\( \ell \) fuselage rolling moment, \( n-m \) (ft-lb)

L fuselage lift, \( N \) (lb)

L) rolling moment, pitching moment, and yawing moment, respectively, \( N-m \) (ft-lb)

M) rolling moment, pitching moment, and yawing moment, respectively, \( N-m \) (ft-lb)

N rotor blade mass, kg (slugs)

M_\beta blade weight moment about flapping hinge, \( N-m \) (lb-ft)

n_b number of blades

p roll, pitch, and yaw rates in the body-c.g. axes system, rad/sec

q roll, pitch, and yaw rates in the body-c.g. axes system relative to the air mass, rad/sec

r roll, pitch, and yaw rates in the hub-body axes system, rad/sec

P ratio of flapping frequency to rotor system angular velocity

q dynamic pressure, \( \frac{1}{2} \rho V^2 \), N/m^2 (lb/ft^2)
Q  torque, N·m (ft-lb)
R  rotor radius, m (ft)
s  Laplace variable
S  area of stabilizing surface, m² (ft²)
STA  longitudinal location in the fuselage axes system, m (ft)
T  thrust, N (lb)

\{u, v, w\}  longitudinal, lateral, and vertical velocities in the body-c.g. system of axes, m/sec (ft/sec)

\{u_B, v_B, w_B\}  longitudinal, lateral, and vertical velocities relative to the air mass in the body-c.g. system of axes, m/sec (ft/sec)

\{u_H, v_H, w_H\}  longitudinal, lateral, and vertical velocities relative to the air mass in the air mass in the hub-body system of axes, m/sec (ft/sec)

\nu_i\}  main rotor induced velocity at rotor disk, m/sec (ft/sec)
\omega_f\}  fuel flow rate, N/hr (lb/hr)
WL  vertical location in the fuselage axes system, m (ft)

\{X, Y, Z\}  longitudinal, lateral, and vertical forces in the body-c.g. axes system, N (lb)

\alpha\}  stabilizing surface angle of attack, rad
\alpha_s\}  angle of attack for stall, rad
\beta_w\}  rotor sideslip angle, rad
\gamma\}  blade Lock number, \(\rho a c R^4/I\_B\)
\delta\}  equivalent rotor blade profile drag coefficient

\delta_a\}  lateral cyclic stick movement, positive to right, cm (in.)
\delta_c\}  collective control input, positive up, cm (in.)
\delta_e\}  longitudinal cyclic stick movement, positive aft, cm (in.)
\delta_p\}  pedal movement, positive right, cm (in.)
\epsilon\}  hinge offset ratio, \(e/R\)
\eta\}  stabilizing surface lift efficiency factor
\( \theta \) Euler pitch angle, rad

\( \theta_0 \) blade root collective pitch, rad

\( \theta_t \) total blade twist (root minus tip incidence), rad

\( \lambda \) inflow ratio \( \frac{\Delta \omega_H}{\Omega} - \frac{C_T}{2(\mu^2 + \lambda^2)^{1/2}} \)

\( \Lambda \) sweepback angle of fin, rad

\( \mu \) rotor advance ratio, \( \frac{\sqrt{u_H^2 + v_H^2}}{\Omega R} \)

\( \rho \) air density, kg/m\(^3\) (slugs/ft\(^3\))

\( \sigma \) rotor solidity ratio, blade area/disk area

\( \tau \) time constant

\( \phi \) Euler roll angle, rad

\( \Omega \) rotor angular velocity, rad/sec

Subscripts:

B body-c.g. axes system relative to air mass

c.g. center of gravity

E engine

f fuselage

F vertical fin

H hub-body axes system, hub location

HS horizontal stabilizer

i induced

MR main rotor

p pilot input

t throttle

TR tail rotor

W hub-wind system of axes
APPENDIX B

AXIS SYSTEMS

Hub Wind

The hub-wind axis system is used in the calculation of rotor forces and moments. The origin of the system is the rotor hub, and the T (thrust) axis is aligned with the rotor shaft. The $H_w$ (horizontal) axis is aligned with the component of relative wind normal to the shaft axis, and the $Y_w$ (side force) axis completes the right-handed orthogonal set. This axis system is shown in figure B-1 along with the components of the relative wind.

![Hub Wind Diagram](image)

Figure B-1.- The hub-wind axis system with main rotor force, moment, and velocity components defined.

Hub Body

The hub-body system coincides with the hub-wind system when the sideslip angle $\beta_w$ is zero. Thus, the T axis is aligned with the shaft axis, and the force $H_R$ lies in the $x_B - z_B$ plane (see fig. B-2).

Body Center of Gravity

All forces and moments are expressed relative to the body-c.g. system for use in the six-degrees-of-freedom rigid body equations of motion. This axis system has its origin at the center of gravity with the x axis aligned with the longitudinal axis of the helicopter, and the z axis lying on the plane of symmetry (see fig. B-2).
Aircraft Reference

The aircraft reference axes are used to locate all force and moment generating components and the center of gravity. The aircraft reference axes are parallel to the body-c.g. axes. The axis origin is typically located ahead and below the aircraft at some arbitrary point within the plane of symmetry. Stations are measured positive aft along the longitudinal axis. Buttlines are lateral distances, positive to the pilot's right. Waterlines are measured vertically, positive upward (see fig. B-2).

Local Wind

The local-wind axes systems are employed for calculation of lift and drag forces on the empennage and on the fuselage. For each empennage surface, the origin is at the quarter point of the mean aerodynamic chord, and the lift force is normal to the relative wind and to a spanwise line passing through the quarter chord point.
The derivation of the tip-path plane dynamic equation is given in detail in reference 2. For the nonteetering N-bladed rotor, the tip-path plane dynamic equations are as follows:
\[ \ddot{\alpha} + \ddot{\beta} + \ddot{\gamma} = \ddot{r} \]

\[
\ddot{\beta} = \Omega \begin{bmatrix}
\frac{\gamma}{2} \left( \frac{1}{4} - \frac{2}{3} \epsilon + \frac{\epsilon^2}{2} \right) & 0 & - \frac{\gamma u}{4} \left( \frac{1}{3} - \epsilon + \frac{\epsilon^2}{2} \right) \\
0 & \frac{\gamma}{2} \left( \frac{1}{4} - \frac{2}{3} \epsilon + \frac{\epsilon^2}{2} \right) & 2 \\
- \frac{\gamma u}{2} \left( \frac{1}{3} - \epsilon + \frac{\epsilon^2}{2} \right) & -2 & \frac{\gamma}{2} \left( \frac{1}{4} - \frac{2}{3} \epsilon + \frac{\epsilon^2}{2} \right)
\end{bmatrix}
\]

\[ \ddot{\gamma} = \Omega^2 \begin{bmatrix}
p^2 + \frac{\gamma K \mu^2}{4} \left( \frac{1}{2} - \epsilon + \frac{\epsilon^2}{2} \right) & - \frac{\gamma u}{4} \left( \frac{1}{2} - \epsilon + \frac{\epsilon^2}{2} \right) & - \frac{\gamma K \mu^2}{4} \left( \frac{1}{2} - \epsilon + \frac{\epsilon^2}{2} \right) \\
- \frac{\gamma u}{2} \left( \frac{1}{2} - \epsilon + \frac{\epsilon^2}{2} \right) & p^2 - 1 + \frac{\gamma K \mu^2}{8} \left( \frac{1}{4} - \frac{2}{3} \epsilon + \frac{\epsilon^2}{2} \right) & \frac{\gamma}{2} \left( \frac{1}{4} - \frac{2}{3} \epsilon + \frac{\epsilon^2}{2} \right) + \frac{\gamma u^2}{8} \left( \frac{1}{2} - \epsilon + \frac{\epsilon^2}{2} \right) \\
- \frac{\gamma K \mu^2}{2} \left( \frac{1}{2} - \epsilon + \frac{\epsilon^2}{2} \right) & - \frac{\gamma}{2} \left( \frac{1}{4} - \frac{2}{3} \epsilon + \frac{\epsilon^2}{2} \right) + \frac{\gamma u^2}{8} \left( \frac{1}{2} - \epsilon + \frac{\epsilon^2}{2} \right) & p^2 - 1 + \frac{3}{8} \gamma K \mu^2 \left( \frac{1}{2} - \epsilon + \frac{\epsilon^2}{2} \right)
\end{bmatrix}
\]

\[ f = \Omega^2 \begin{bmatrix}
- \frac{M \beta}{16} + \frac{\gamma}{2} \left[ \left( \frac{1}{4} - \frac{\epsilon}{3} \right) + \frac{u^2}{2} \left( \frac{1}{2} - \epsilon + \frac{\epsilon^2}{2} \right) \right] \theta_0 - \frac{\gamma}{2} \left[ \mu \left( \frac{1}{3} - \frac{\epsilon}{2} \right) - \frac{\gamma}{2} \mu \left( \frac{1}{3} - \frac{\epsilon}{2} \right) \right] B \frac{1}{16} + \frac{\gamma}{2} \left[ \left( \frac{1}{3} - \frac{\epsilon}{2} \right) + \frac{u^2}{2} \left( \frac{1}{3} - \frac{\epsilon}{2} \right) \right] \theta_t + \frac{\gamma}{2} \left( \frac{1}{3} - \frac{\epsilon}{2} \right) \\
- 2 \left( 1 + \frac{e \mu B}{2 \beta} \right) \left( \frac{P H}{\omega} \cos \beta_w + \frac{Q H}{\omega} \sin \beta_w \right) + \frac{\gamma}{8} \left( \frac{2}{3} - \epsilon \right) \left( \frac{P H}{\omega} \cos \beta_w + \frac{Q H}{\omega} \sin \beta_w \right) \\
- 2 \left( 1 + \frac{e \mu B}{2 \beta} \right) \left( \frac{P H}{\omega} \sin \beta_w - \frac{Q H}{\omega} \cos \beta_w \right) + \frac{\gamma}{4} \left( \frac{1}{3} - \frac{\epsilon}{3} \right) \left( \frac{P H}{\omega} \sin \beta_w - \frac{Q H}{\omega} \cos \beta_w \right) - \frac{\gamma}{2} \left( \frac{1}{3} - \frac{\epsilon}{3} \right) \theta_0 - \frac{\gamma u}{2} \left( \frac{1}{2} - \epsilon \right) \theta_t \\
+ \frac{\gamma}{2} \left[ \left( \frac{1}{4} - \frac{\epsilon}{3} \right) + \frac{u^2}{4} \left( \frac{1}{2} - \epsilon + \frac{\epsilon^2}{2} \right) \right] B \frac{1}{16} - \frac{\gamma u}{2} \left( \frac{1}{2} - \epsilon + \frac{\epsilon^2}{2} \right) \Lambda - \frac{\gamma}{2} \left( \frac{1}{4} - \frac{\epsilon}{3} \right) \left( \frac{P H}{\omega} \cos \beta_w + \frac{Q H}{\omega} \sin \beta_w \right)
\end{bmatrix}
\]
where

\[ p^2 = 1 + \frac{K_B}{I_B \Omega^2} + \frac{eM_p}{g_2} + \frac{\gamma K_1}{8} \left( 1 - \frac{4}{3} \varepsilon \right) \]

and

\[ \tilde{a} = (a_1, a_1, b_1)^T \]

For a two-bladed teetering rotor, the tip-path plane representation loses its physical meaning. However, if the approximation for blade flapping is employed, that is,

\[ \beta(t) = a_0(t) - a_1(t) \cos \psi - b_1(t) \sin \psi \]

then \( a_0 \) is treated as a preset constant. The coefficients \( a_1(t) \) and \( b_1(t) \) can then be solved for by setting \( \varepsilon = \dot{a}_0 = \ddot{a}_0 = 0 \) in equation (1).

The force and moment expressions, with the harmonic terms dropped that correspond to the number of rotor blades are given below. A derivation of these equations is given in reference 2.

\[ T_{MR} = \frac{n_b}{2} \rho c R (\Omega R)^2 \left[ \frac{1}{2} (1 - \varepsilon^2) \lambda + \dot{\theta}_0 \left[ \frac{1}{3} + \frac{u^2}{2} (1 - \varepsilon) \right] + \ddot{\theta}_0 \left[ \frac{1}{4} + \frac{u^2}{4} (1 - \varepsilon^2) \right] \right] \]

\[ - \frac{u}{2} (1 - \varepsilon^2) (B_{1c} - K_1 b_1) - a_0 \left[ \frac{1}{3} + \frac{u^2}{2} (1 - \varepsilon) \right] K_1 + a_1 \left[ \frac{u}{2} \varepsilon (1 - \varepsilon) \right] - \frac{\dot{a}_0}{\Omega} \left( \frac{1}{3} - \frac{\varepsilon}{2} \right) \]

\[ + \frac{b_1}{\Omega} \left[ \frac{u}{4} (1 - \varepsilon^2) \right] + \frac{u}{4} (1 - \varepsilon^2) \left( \frac{P_H}{\Omega} \cos \beta w + \frac{q_H}{\Omega} \sin \beta w \right) - \frac{n_b M_B}{g} \ddot{a}_0 \]
\[ H_w = \frac{n_b}{2} \rho a c R(\Omega R)^2 \left( \frac{\delta u}{2a} (1 - \varepsilon^2) - \frac{1}{4} (\theta_0 - K_1 a_0) \left[ 2\lambda u(1 - \varepsilon) - u(1 - \varepsilon)^2 \frac{b_0}{\Omega} \right] 
- \left( \varepsilon - \frac{2}{3} \right) \left( \frac{b_1}{\Omega} - a_1 \right) - \frac{2}{3} a_1 + \frac{2}{3} \left( \frac{p_H}{\Omega} \cos \beta_w + \frac{q_H}{\Omega} \sin \beta_w \right) \right) 
- \frac{\theta t}{4} \left[ \mu \lambda (1 - \varepsilon^2) + \frac{b_0}{\Omega} \mu (1 - \varepsilon^2) - 2 \left( \frac{\varepsilon}{3} - \frac{1}{4} \right) \left( \frac{b_1}{\Omega} - a_1 \right) - \frac{a_1}{2} \right] 
+ \frac{1}{2} \left( \frac{p_H}{\Omega} \cos \beta_w + \frac{q_H}{\Omega} \sin \beta_w \right) \right] + \frac{1}{4} (A_{1c} - K_1 a_1) \left[ - \frac{b_1}{4} (1 - \varepsilon^2) \right] 
+ \frac{1}{4} u(1 - \varepsilon)^2 \left( \frac{b_1}{\Omega} + b_1 \right) + \frac{2}{3} a_0 + \frac{\mu}{4} (1 - \varepsilon^2) \left( - \frac{p_H}{\Omega} \sin \beta_w + \frac{q_H}{\Omega} \cos \beta_w \right) \right) 
+ \frac{1}{4} (A_{1c} - K_1 b_1) \left[ \frac{3}{4} \mu (1 - \varepsilon)^2 \left( \frac{b_1}{\Omega} - a_1 \right) + (1 - \varepsilon^2) \left( \lambda - \frac{a_1}{4} \right) + \left( \varepsilon - \frac{2}{3} \right) \frac{b_0}{\Omega} \right] 
+ \frac{3u}{4} (1 - \varepsilon^2) \left( \frac{p_H}{\Omega} \cos \beta_w + \frac{q_H}{\Omega} \sin \beta_w \right) \right] + \frac{1}{4} \left( \varepsilon (1 - \varepsilon) \right) \left( \frac{b_1}{\Omega} - a_1 \right) \frac{4\lambda}{\Omega} 
- (1 - \varepsilon^2) \left[ \frac{2\lambda}{\Omega} \left( \frac{b_1}{\Omega} - a_1 \right) - \frac{2}{3} \left( 1 - \varepsilon^2 \right) \left( a_1 \frac{b_0}{\Omega} + a_0 \left( \frac{b_1}{\Omega} + b_1 \right) \right) \right] 
- \frac{2a_0}{3} \left( - \frac{p_H}{\Omega} \sin \beta_w + \frac{q_H}{\Omega} \cos \beta_w \right) - \left[ 2(1 - \varepsilon^2) \lambda - 4 \left( \frac{1}{3} - \frac{3}{2} \right) \frac{b_0}{\Omega} \right] \left( \frac{p_H}{\Omega} \cos \beta_w \right) 
+ \frac{q_H}{\Omega} \sin \beta_w \right] + \frac{4}{3} \left( \frac{b_1}{\Omega} - a_1 \right) \left( \frac{1}{3} - \varepsilon + \varepsilon^2 \right) \right] + \frac{\mu}{4} \left( \varepsilon (1 - \varepsilon) \right) \left[ a_1 \left( \frac{b_1}{\Omega} - a_1 \right) + b_1 \left( \frac{b_1}{\Omega} + b_1 \right) \right] 
+ \frac{b_1}{4} (1 - \varepsilon)^2 \left[ a_1 \left( \frac{b_1}{\Omega} - a_1 \right) + b_1 \left( \frac{b_1}{\Omega} + b_1 \right) - 2a_0^2 - \frac{b_1^2}{2} - 3a_1^2 \right] 
- \frac{a_1}{4} (1 - \varepsilon^2) \left( \frac{p_H}{\Omega} \cos \beta_w + \frac{q_H}{\Omega} \sin \beta_w \right) - \frac{b_1}{4} (1 - \varepsilon^2) \left( - \frac{p_H}{\Omega} \sin \beta_w + \frac{q_H}{\Omega} \cos \beta_w \right) \right] \]
\[
Y_W = \frac{n_b}{2} \rho a c R(\Omega R)^2 \left[ -\frac{1}{4} \left( a_0 - K_1 a_1 \right) \left[ \left( e - \frac{2}{3} \right) \left( \frac{\dot{a}_1}{\Omega} + b_1 \right) - \frac{2}{3} a_1 \right] + 3a_0 \left( 1 - e^2 \right) \mu \right.
\]
\[
- 2b_1 (1 - e) \mu^2 - \frac{2}{3} \left( \frac{p_H}{\Omega} \sin \beta_w + \frac{q_H}{\Omega} \cos \beta_w \right) \left[ \frac{\dot{a}_1}{\Omega} + b_1 \right] + \frac{\theta \epsilon}{4} \left[ \left( \frac{2e - 1}{3} \frac{p_H}{\Omega} \sin \beta_w + \frac{q_H}{\Omega} \cos \beta_w \right) \right]
\]
\[
+ 2a_0 \mu - b_1 (1 - e^2) \mu^2 - \frac{1}{2} \left[ \frac{p_H}{\Omega} \sin \beta_w + \frac{q_H}{\Omega} \cos \beta_w \right]
\]
\[
- \frac{1}{4} (A_1 c - K_1 a_1) \left[ \left( e - \frac{2}{3} \right) \frac{\dot{a}_0}{\Omega} + \lambda (1 - e^2) \right] + \mu \left[ \frac{5a_1}{4} (1 - e^2) \right.
\]
\[
+ \frac{1}{4} (1 - e^2) \left( \frac{b_1}{\Omega} - a_1 \right) \left[ + \frac{\mu}{4} (1 - e^2) \right. \left( \frac{p_H}{\Omega} \cos \beta_w + \frac{q_H}{\Omega} \sin \beta_w \right)
\]
\[
- \frac{1}{4} (B_1 c - K_1 b_1) \left[ - \frac{2}{3} a_0 + \mu \left[ \frac{7}{4} b_1 (1 - e^2) + \frac{1}{4} (1 - e^2) \left( \frac{\dot{a}_1}{\Omega} + b_1 \right)
\right.
\]
\[
+ \frac{1}{4} \left( \frac{p_H}{\Omega} \sin \beta_w + \frac{q_H}{\Omega} \cos \beta_w \right) \right] \left[ - \frac{\mu}{4} \left[ 2a_0 (1 - e) \right]
\right.
\]
\[
- \frac{1}{4} \left( 4 \left( \frac{1}{3} - e + e^2 \right) \frac{\dot{a}_0}{\Omega} \left( \frac{\dot{a}_1}{\Omega} + b_1 \right) - 2\lambda (1 - e)^2 \left( \frac{\dot{a}_1}{\Omega} + b_1 \right)
\right.
\]
\[
+ \frac{2a_0}{3} \left( \frac{p_H}{\Omega} \cos \beta_w + \frac{q_H}{\Omega} \sin \beta_w \right) + 2a_0 \left( \frac{1}{3} - \frac{e}{2} \right) \left( \frac{b_1}{\Omega} - a_1 \right) - 2b_1 \left[ \frac{\lambda}{2} (1 - e^2)
\right.
\]
\[
- \frac{\dot{a}_0}{\Omega} \left( \frac{1}{3} - \frac{e}{2} \right) \left[ + 4 \left( \frac{1}{3} - \frac{e}{2} \right) \frac{\dot{a}_0}{\Omega} - 2(1 - e^2) \lambda \right] \left( \frac{p_H}{\Omega} \sin \beta_w + \frac{q_H}{\Omega} \cos \beta_w \right)
\right.
\]
\[
- \frac{\mu}{4} \left[ 6a_0 \lambda (1 - e) - \frac{a_1 b_1}{2} (1 - e^2) - 3(1 - e)^2 a_0 \frac{\dot{a}_0}{\Omega}
\right.
\]
\[
- \frac{7}{4} (1 - e^2) a_0 \left( \frac{\dot{a}_1}{\Omega} + b_1 \right) - \frac{5}{4} b_1 (1 - e^2) \left( \frac{p_H}{\Omega} \cos \beta_w + \frac{q_H}{\Omega} \sin \beta_w \right)
\right.
\]
\[
- \frac{7}{4} a_1 (1 - e^2) \left( \frac{p_H}{\Omega} \sin \beta_w + \frac{q_H}{\Omega} \cos \beta_w \right) - \frac{5}{4} (1 - e)^2 b_1 \left( \frac{b_1}{\Omega} - a_1 \right)
\right.
\]
\[
\left. - \mu^2 [a_0 a_1 (1 - e)] \right]
\]
\[
M_W = \frac{n_b}{2} \left[ K_\beta a_1 - \frac{e M_c}{8} (\dot{a}_1 + 2b_1 \Omega - a_1 \Omega^2) \right] - \frac{n_b}{2} I \beta \Omega^2 \gamma e \left[ - \frac{1}{6} + \frac{\mu^2}{8} (1 - e) \right] (A_1 c - K_1 a_1)
\]
\[
- \frac{\mu}{4} (1 - e^2) a_0 + \frac{\mu^2}{6} (1 - e) b_1 + \left[ \frac{1}{6} - \frac{e}{4} \right] \left( \frac{\dot{a}_1}{\Omega} + b_1 \right) + \frac{1}{6} \left( - \frac{p_H}{\Omega} \sin \beta_w + \frac{q_H}{\Omega} \cos \beta_w \right)
\]

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\[ I_w = \frac{n_b}{2} \left[ K \beta b_1 - \frac{e M_3}{g} (b_1 - 2 \dot{a}_1 \Omega - b_1 \Omega^2) \right] - \frac{n_b}{2} I_\beta \Omega^2 \gamma e \left( \frac{\mu}{2} (1 - \epsilon^2)(\theta_0 - K_1 \alpha_0) \right. \\
\left. - \left[ \frac{1}{6} + \frac{3}{8} \mu^2 (1 - \epsilon) \right] (B_{1c} - K_1 b_1) + \frac{\mu}{3} \theta_t + \frac{\mu}{2} (1 - \epsilon) \lambda + \frac{\mu^2}{8} (1 - \epsilon) a_1 \right. \\
- \frac{\mu}{4} (1 - \epsilon)^2 \left. \left( \frac{\dot{a}_0}{\Omega} + \left( \frac{1}{6} - \frac{\epsilon}{4} \right) \left( \frac{b_1}{\Omega} - a_1 \right) + \frac{1}{6} \left( \frac{p_H}{\Omega} \cos \beta_w + \frac{q_H}{\Omega} \sin \beta_w \right) \right) \right] \\
Q = \frac{n_b}{2} \rho ac R^2 (\Omega R)^2 \left( \frac{b_1}{4a} \left[ 1 + (1 - \epsilon^2) \mu^2 \right] - (\theta_0 - K_1 \alpha_0) \right) \left[ \frac{\lambda}{3} + \left( \frac{\epsilon}{3} - \frac{1}{4} \right) \frac{\dot{a}_0}{\Omega} + \frac{\mu}{6} \frac{b_1}{\Omega} \right. \\
- \frac{\mu \epsilon}{4} \left( \frac{b_1}{\Omega} - a_1 \right) + \frac{\mu}{6} \left( \frac{p_H}{\Omega} \cos \beta_w + \frac{q_H}{\Omega} \sin \beta_w \right) \left. + (A_{1c} - K_1 a_1) \right] \left[ \left( \frac{1}{8} - \frac{\epsilon}{6} \right) \left( \frac{\dot{a}_1}{\Omega} + b_1 \right) \right. \\
- \frac{\mu}{6} a_0 + \frac{b_1}{16} (1 - \epsilon^2) \mu^2 + \frac{1}{8} \left( - \frac{p_H}{\Omega} \sin \beta_w + \frac{q_H}{\Omega} \cos \beta_w \right) \right]\]

For the case of a two-bladed teetering rotor, the forces and moments can be obtained by setting \( \epsilon = 0, \dot{a}_0 = \ddot{a}_0 = 0 \) in the above equations.

The rotor profile drag coefficient \( \delta \) is required in the computation of \( H_w \) and \( Q \). An expression for this coefficient was employed which provides a reasonable match with measured section characteristics as follows:

\[ \delta = 0.009 + 0.3 \left( \frac{6C_T}{a_0} \right)^2 \]
where \((6\frac{C_T}{\alpha})\) is approximately equal to an averaged equivalent rotor blade angle of attack.

The rotor inflow ratio, defined as

\[
\lambda = \frac{w_H}{\Omega R} - \frac{C_T}{2(\mu^2 + \lambda^2)^{1/2}}
\]

is required in the computation of thrust. This implicit relationship is solved in the computer program through a Newton-Ralphson iterative technique.

The main rotor-induced velocity at the disk is required in subsequent calculations of the rotor-induced velocities on the helicopter components. The expression for this velocity is as follows:

\[
v_i = \left(\frac{w_H}{\Omega R} - \lambda\right) \Omega R
\]

Transformation \(T_1\)

The calculation of the rotor flapping dynamics and the rotor forces and moments requires the angular velocities and accelerations expressed in the hub-body system of axes and the angle of sideslip at the hub. In addition, the cyclic pitch must be expressed in the hub-wind system. The required velocities and accelerations are obtained from those expressed in the body-c.g. system as follows:

\[
p_H = p \cos i_s + r \sin i_s
\]

\[
q_H = q
\]

\[
\dot{p}_H = \dot{p} \cos i_s + \dot{r} \sin i_s
\]

\[
\dot{q}_H = \dot{q}
\]

\[
u_H = [u_B - r_B(BL_{c.g.}) - q_B(WL_H - WL_{c.g.})] \cos i_s + [w_B + p_B(BL_{c.g.}) - q_B(STA_{c.g.} - STA_H)] \sin i_s
\]

\[
v_H = v_B + p_B(WL_H - WL_{c.g.}) + r_B(STA_{c.g.} - STA_H)
\]

\[
w_H = [w_B + p_B(BL_{c.g.}) - q_B(STA_{c.g.} - STA_H)] \cos i_s + [-u_B + r_B(BL_{c.g.}) + q_B(WL_H - WL_{c.g.})] \sin i_s
\]

where \(i_s\) is the forward tilt of the shaft axis relative to the body-c.g. Z-axis (see fig. B-2) and \(STA, WL, \) and \(BL\) are coordinates in the aircraft reference system of axes. It is assumed that the rotor hub lies in the \(BL = 0\) plane.

The sideslip of the rotor is then defined as:
\[ \beta_w = \sin^{-1} \frac{v_H}{(v_H^2 + u_H^2)^{1/2}} \]

(\(\beta_w\) is defined as zero if \(v_H = u_H = 0\).) Finally the cyclic pitch must be expressed in the hub-wind axis system. The expressions are:

\[
A_{1c} = A_{1s} \cos \beta_w - B_{1s} \sin \beta_w
\]

\[
B_{1c} = A_{1s} \sin \beta_w + B_{1s} \cos \beta_w
\]

**Transformation \( T_2 \)**

The rotor forces and moments calculated in the hub-wind axes system must be expressed in the body-c.g. system as inputs to the six-degree-of-freedom rigid-body equations of motion. To accomplish this, the forces and moments are first expressed in the hub-body system of axes and then transferred to the body-c.g. system.

The forces and moments in the x-y plane are modified by the first transformation as follows:

\[
H_h = H_w \cos \beta_w + Y_w \sin \beta_w
\]

\[
Y_h = -H_w \sin \beta_w + Y_w \cos \beta_w
\]

\[
M_h = M_w \cos \beta_w + L_w \sin \beta_w
\]

\[
L_h = -M_w \sin \beta_w + L_w \cos \beta_w
\]

The transformation to the body-c.g. system accounts for the shaft-tilt and the moment about the c.g. caused by the rotor forces. The forces expressed in the body-c.g. system are:

\[
X_{MR} = T \sin i_s - H_h \cos i_s
\]

\[
Y_{MR} = Y_h
\]

\[
Z_{MR} = -T \cos i_s - H_h \sin i_s
\]

The moments in the body-c.g. system are:

\[
L_{MR} = L_h \cos i_s - Q \sin i_s + Y_{MR}(W_LH - W_L \text{c.g.}) - Z_{MR}(B_L \text{c.g.})
\]

\[
M_{MR} = M_h - Z_{MR}(S_T a \text{c.g.} - S_T H) - X_{MR}(W_LH - W_L \text{c.g.})
\]

\[
N_{MR} = Q \cos i_s + L_h \sin i_s + X_{MR}(B_L \text{c.g.}) + Y_{MR}(S_T \text{c.g.} - S_T H)
\]
APPENDIX D

TAIL ROTOR FLAPPING AND FORCE CALCULATION

The tail rotor is assumed to be teetering with a constant built-in coning $a_0$. The angular rate of the tail rotor is sufficiently high so that the tip-path dynamics may be ignored. Under these conditions, $a_1$ and $b_1$ in the flapping equations are assumed to be constant, and a steady-state solution of the equations is obtained to yield these quantities as follows:

$$a_{1TR} = \frac{1}{\Delta_{TR}} \left[ k_{1TR} \left( 1 + \frac{3}{2} \mu_{TR}^2 \right) f_{1TR} - \left( 1 + \frac{\mu_{TR}^2}{2} \right) f_{2TR} \right]$$

$$b_{1TR} = \frac{1}{\Delta_{TR}} \left[ \left( 1 - \frac{\mu_{TR}^2}{2} \right) f_{1TR} + k_{1TR} \left( 1 + \frac{\mu_{TR}^2}{2} \right) f_{2TR} \right]$$

where

$$\Delta_{TR} = 1 - \frac{\mu_{TR}^4}{4} + \kappa_{TR}^2 \left( 1 + \frac{\mu_{TR}^2}{2} \right) \left( 1 + \frac{3}{2} \mu_{TR}^2 \right)$$

$$f_{1TR} = \frac{4}{3} \nu_{TR}^2 a_{TR}^0 - \frac{16 p_{TR}}{\gamma_{TR} \Omega_{TR}} - \frac{q_{TR}}{\Omega_{TR}}$$

$$f_{2TR} = \frac{8}{3} k_{1TR} \nu_{TR}^2 a_{TR}^0 + \frac{16 q_{TR}}{\gamma_{TR} \Omega_{TR}}$$

$$- \mu_{TR} \left( \frac{8}{3} \theta_{TR}^0 + 2 \theta_{TR} \right) + \frac{p_{TR}}{\Omega_{TR}}$$

The forces on the tail rotor are then calculated as follows:

$$T_{TR} = \rho a_{TR} c_{TR} R_{TR} (\Omega_{TR} R_{TR})^2 \left[ \frac{\lambda_{TR}}{2} + \theta_{TR}^0 \left( \frac{1}{3} + \frac{\mu_{TR}^2}{2} \right) + \frac{\theta_{TR}}{4} \left( 1 + \frac{\mu_{TR}^2}{2} \right) + \frac{\mu_{TR} k_{1TR}}{2} b_{1TR} \right.$$

$$- a_{TR} \left( \frac{1}{3} + \frac{\mu_{TR}^2}{2} \right) k_{1TR} + \frac{\mu_{TR} p_{TR}}{4} \Omega_{TR} \left]$$

$$- a_{TR} \left( \frac{1}{3} + \frac{\mu_{TR}^2}{2} \right) k_{1TR} + \frac{\mu_{TR} p_{TR}}{4} \Omega_{TR} \right]$$

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The positive directions for tail rotor torque and rotation are shown on figure D-1.
The rotor blade profile drag coefficient, \( \delta \), is required in the computation of \( H_{TR} \) and \( Q_{TR} \). This is calculated similarly to the case of the main rotor as follows:

\[
\delta_{TR} = 0.009 + 0.3 \left( \frac{6 C_{TR}}{\sigma_{TR} a_{TR}} \right)^2
\]

The inflow ratio, as in the case of the main rotor, must be calculated from the following relationship:

\[
\lambda_{TR} = \frac{-V_{TR}}{\Omega_{TR} R_{TR}} - \frac{C_{TR}}{2V_{TR}^2 + \lambda_{TR}^2}
\]
This implicit relationship is solved through an iteration procedure.

Main Rotor Interference and Transformation $T_3$

The calculation of the tail rotor forces and moments requires the local flow velocity components in the hub-wind axes system for the tail rotor. The velocity at the tail rotor includes provision for the downwash contributed by the flow field of the main rotor. For these calculations, the tail rotor is assumed to lie in the $x$-$z$ plane.

The velocities at the rotor hub in an axis system co-directional with the body-c.g. system are:

$$u_{TR} = u_B$$
$$v_{TR} = v_B + p_B(WL_{TR} - WL_{c.g.}) - r_B(STA_{TR} - STA_{c.g.})$$
$$w_{TR} = w_B + q_B(STA_{TR} - STA_{c.g.}) + w_{iTR}$$

The quantity $w_{iTR}$ is the downwash velocity due to the main rotor. The downwash velocity is represented in the modes as a function of the wake angle $\chi$ and sideslip. The variation of the downwash varies nonlinearly with location in the wake. Therefore, the downwash must be determined uniquely for each location from data such as those published in reference 10. The representation in the model consists of the downwash expressed as a function of a power series in the wake angle at angles of sideslip of $0^\circ$, $\pm 90^\circ$, and $180^\circ$ as follows:

$$\left(\frac{w_{iTR}}{v_1}\right) = \sum_{n=0}^{b} k_n \chi^n$$

where $v_1$ is the momentum-theory value of the rotor-induced velocity. The coordinate system employed in reference 10 is centered at the hub, and the $X$ and $Y$ axes lie in the tip-path plane. Accordingly, the tail rotor location in this coordinate system varies with the value of $\alpha$, the tilt of the tip-path plane. In view of the approximate nature of the wake estimate, which can vary widely with disk load distribution, inclusion of this level of detail is unwarranted. Accordingly, the downwash is estimated for a constant location referenced to the hub-body system and the wake angle referred to that system. Hence,

$$\chi = \tan^{-1}\left(\frac{\mu}{\lambda}\right)$$

Interpolation for the value of the downwash at arbitrary sideslip angles is accomplished as in the following example, where $\beta$ has been determined to be between $\pm 90^\circ$:

$$w_{iTR} = \left[\left(\frac{w_{iTR}}{v_1}\right)_{\beta=0}\cos \beta + \left(\frac{w_{iTR}}{v_1}\right)_{\beta=90^\circ}\sin \beta\right]v_1$$

Similarly, when $90^\circ < \beta < 270^\circ$, the downwash is calculated as follows:
\[
\omega_{TR} = \left( \frac{\omega_{TR}}{v_1} \right)_{\beta=18^0} |\cos \beta| + \left( \frac{\omega_{TR}}{v_1} \right)_{\beta=90^0} |\sin \beta| \right) v_1
\]

The advance ratio for the tail rotor, which is assumed to be in the \( x-z \) plane, is then:

\[
u_{TR} = \sqrt{\frac{u_{TR}^2 + w_{TR}^2}{\Omega_{TR}}}
\]

and the rotor sideslip angle is defined for the tail rotor as:

\[
\beta_{TR} = \tan^{-1} \left( \frac{w_{TR}}{u_{TR}} \right)
\]

Then, in the hub-wind axis system defined for the tail rotor, the angular velocities are:

\[
p_{TR} = p_B \cos \beta_{TR} + r_B \sin \beta_{TR}
\]

\[
q_{TR} = r_B \cos \beta_{TR} - p_B \sin \beta_{TR}
\]

Transformation \( T_\omega \)

The transformation to express the tail rotor forces and moments in the body-c.g. system of axes is as follows:

\[
X_{TR} = -(Y_{TR})_W \sin \beta_{TR} - (H_{TR})_W \cos \beta_{TR}
\]

\[
Y_{TR} = T_{TR}
\]

\[
Z_{TR} = (Y_{TR})_\cos \beta_{TR} - (H_{TR})_W \sin \beta_{TR}
\]

\[
L_{TR} = Y_{TR}(WL_{TR} - WL_{c.g.})
\]

\[
M_{TR} = -Q_{TR} + Z_{TR}(ST_{TR} - STA_{c.g.}) - X_{TR}(WL_{TR} - WL_{c.g.})
\]

\[
N_{TR} = -Y_{TR}(STA_{TR} - STA_{c.g.})
\]
APPENDIX E

EMPENNAGE FORCES AND MOMENTS

The expressions for the lift and drag coefficients for the vertical fin and the horizontal stabilizer are identical when referenced to a local wind-axes system for the particular surface. (See appendix B.)

The conditions at the stall of each surface are defined as follows:

\[
\begin{align*}
\alpha_s &= \frac{C_{LM}}{a} & \alpha_s &\leq \frac{\pi}{4} \\
\alpha_s &= \frac{\pi}{4} & \alpha_s &> \frac{\pi}{4} \\
\alpha_1 &= 1.2\alpha_s \\
C_{LM} &= C_{LM} & \alpha_s &\leq \frac{\pi}{4} \\
C_{LM} &= a\frac{\pi}{4} & \alpha_s &> \frac{\pi}{4}
\end{align*}
\]

The angle of attack for input into the expressions for the lift and drag coefficient is:

\[
\begin{align*}
\alpha_1 &= \alpha & 0 \leq \alpha &< \frac{\pi}{2} \\
\alpha_1 &= -\alpha & -\frac{\pi}{2} \leq \alpha &< 0 \\
\alpha_1 &= \pi - \alpha & \frac{\pi}{2} \leq \alpha &< \pi \\
\alpha_1 &= \pi + \alpha & -\pi \leq \alpha &< -\frac{\pi}{2}
\end{align*}
\]

where \(\alpha_1\) represents the angle-of-attack input to the lift and drag equations.

The lift and drag coefficients are calculated using the following expressions:

\[
\begin{align*}
C_{L0} &= a\alpha_1 & 0 \leq \alpha_1 &< \alpha_s \\
C_{L0} &= C_{LM} - a(\alpha_1 - \alpha_s) & \alpha_s &< \alpha_1 < \alpha_1 \\
C_{L0} &= 0.8C_{LM}\left[1 - \left(\frac{\alpha_1 - \alpha_s}{\pi/2 - \alpha_1}\right)^2\right] & \alpha_1 &\leq \alpha_1 \leq \frac{\pi}{2} \\
C_{Dp} &= 0.009 + 0.11\alpha_1^2 & 0 \leq \alpha_1 &\leq 0.35 \\
C_{Dp} &= -0.1254 + 0.9415\alpha_1 + 0.977525\sin^2\alpha_1 & 0.35 \leq \alpha_1 < \frac{\pi}{2}
\end{align*}
\]
where $C_{DP}$ is the profile drag coefficient.

$$C_D = C_{DP} + \frac{C_L^2}{0.8\pi AR}$$

Depending on the quadrant in which the angle of attack lies, the lift coefficient for the surface is:

- $C_L = C_{L0}$ for $0 \leq \alpha < \frac{\pi}{2}$
- $C_L = -C_{L0}$ for $\frac{\pi}{2} \leq \alpha < \pi$
- $C_L = -0.8C_{L0}$ for $\frac{\pi}{2} \leq \alpha < \pi$
- $C_L = 0.8C_{L0}$ for $-\pi \leq \alpha < -\frac{\pi}{2}$

**Horizontal Tail**

The lift curve slope of the horizontal tail is calculated from elementary lifting line theory and is corrected for sideslip as follows:

$$\alpha_{HS} = \left(\frac{2\pi}{I + 2/AR_{HS}}\right) \cos^2 \beta_{HS}$$

**Main rotor interference and transformation $T_5$** - The velocity at the horizontal tail includes a factor to account for rotor downwash, $w_{1HT}$. This factor is calculated similarly to that calculated for the tail rotor. In calculating the velocities at the horizontal tail, the relatively small contribution of roll rate is ignored. The expressions for the velocity components at the horizontal tail are as follows:

$$u_{HS} = u_B$$
$$v_{HS} = v_B - r_B(STA_{HS} - STA_{c.g.})$$
$$w_{HS} = w_B + q_B(STA_{HS} - STA_{c.g.}) + w_{1HT}$$

The angle of attack and angle of sideslip of the horizontal tail are then:

$$\alpha_{HS} = \tan^{-1}\left(\frac{w_{HS}}{u_{HS}}\right) + i_{HS}$$

$$\beta_{HS} = \sin^{-1}\left(\frac{v_{HS}}{V_{HS}}\right)$$

where

$$V_{HS} = \sqrt{u_{HS}^2 + v_{HS}^2 + w_{HS}^2}$$
The angle $\alpha_{HS}$ is the incidence of the horizontal tail. Note that the effect of a cambered surface may be accounted for by selecting this parameter to differ from the angle of the chord plane by an amount equal to the angle of attack for zero lift.

The factor employed to calculate lift and drag forces from the lift and drag coefficients is:

$$F_{HS} = \frac{1}{2} \rho V_H^2 S_{HS} n_{HS}$$

where the constant $n_{HS}$ is used to account for fuselage blockage or the horizontal tail area considered to be within the fuselage.

Transformation $T_6$- The forces in the body-c.g. system of axes attributed to the horizontal tail are:

$$X_{HS} = F_{HS} (-C_{D_{HS}} \cos \beta_{HS} \cos \alpha_{HS} + C_{L_{HS}} \sin \alpha_{HS})$$

$$Y_{HS} = F_{HS} (-C_{D_{HS}} \sin \beta_{HS})$$

$$Z_{HS} = F_{HS} (-C_{L_{HS}} \cos \alpha_{HS} - C_{D_{HS}} \cos \beta_{HS} \sin \alpha_{HS})$$

The forces are assumed to act at the quarter chord point of the horizontal tail at a spanwise location corresponding to the center of area. The moments are:

$$L_{HS} = Y_{HS} (W_{L_{HS}} - W_{L_{c.g.}}) + Z_{HS} (B_{L_{HS}})$$

$$M_{HS} = -X_{HS} (W_{L_{HS}} - W_{L_{c.g.}}) + Z_{HS} (S_{T_{A_{HS}}} - S_{T_{A_{c.g.}}})$$

$$N_{HS} = -X_{HS} (B_{L_{HS}}) - Y_{HS} (S_{T_{A_{HS}}} - S_{T_{A_{c.g.}}})$$

Vertical Fin

The estimate of the lift curve slope of the vertical fin includes the effect of sweep as well as of sideslip as follows:

$$a_F = \left( \frac{2\pi}{1 + 2/AR_F} \right) \cos^2 (\beta_F + \Lambda_F)$$

Transformation $T_7$- The computation of the velocity at the vertical fin ignores the small contribution from roll rate and includes a term in the sideslip due to the induced velocity of the tail rotor. The three components of velocity are as follows:

$$u_F = u_B$$

$$v_F = v_B - r_B (S_{T_{A_F}} - S_{T_{A_{c.g.}}}) + v_{iTR} k v_{TR}$$

$$w_F = w_B + q_B (S_{T_{A_F}} - S_{T_{A_{c.g.}}}) + w_{iF}$$
The constant $k v_{TR}$ can be adjusted to account for the fraction of the vertical tail immersed in the tail rotor wake. The main rotor interference factor $w_{IF}$ is computed similarly to the factor for downwash at the tail rotor.

The angles of attack and sideslip of the vertical fin in the local wind coordinate system are:

$$\alpha_F = \tan^{-1}\left( \frac{v_F}{u_F} \right) + i_F$$

$$\beta_F = \sin^{-1}\left( \frac{w_F}{V_F} \right)$$

where

$$V_F = \sqrt{u_F^2 + v_F^2 + w_F^2}$$

As in the case of the horizontal tail, the angle $i_F$ can be adjusted to account for camber of the vertical fin.
APPENDIX F

CALCULATION OF FUUSELAGE FORCES AND MOMENTS

In calculating the fuselage forces and moments, it is assumed that the longitudinal forces and moments are dependent on angle of attack and the lateral forces and moments are dependent on angle of sideslip. The exception is the drag force, which is assumed to have a contribution from both angle of attack and sideslip. This representation is a gross simplification, and if detailed test data are available on which to base a more sophisticated representation, they should be substituted. (See, e.g., ref. 8.)

Low-angle approximation $\alpha, \beta = -15^\circ$ to $15^\circ$

\[
\begin{align*}
L &= q_f \left( \frac{L_0}{q} + \frac{3L/q}{\alpha} \alpha \right) \\
Y &= q_f \left( \frac{3Y/q}{\beta} \beta \right) \\
D_\alpha &= q_f \left( \frac{D_0}{q} + \frac{3D/q}{\alpha} \alpha + \frac{3^2D}{\partial^2\alpha} \alpha^2 \right) \\
\zeta &= q_f \left( \frac{3Z/q}{\beta} \beta \right) \\
M &= q_f \left( \frac{M_0}{q} + \frac{3M/q}{\alpha} \alpha \right) \\
N &= q_f \left( \frac{3N/q}{\beta} \beta \right) \\
D &= D_\alpha + D_\beta \\
D_\beta &= q_f \left( \frac{3^2D/q}{\beta^2} \beta^2 \right)
\end{align*}
\]

High-angle approximation

\[
\begin{align*}
L &= q_f \left( \frac{D_{\alpha=90^\circ}}{q_f} |\sin \alpha| \sin \alpha \cos \alpha \right) \\
Y &= q_f \left( \frac{-D_{\beta=90^\circ}}{q_f} |\sin \beta| \sin \beta \cos \beta \right) \\
D_\alpha &= q_f \left( \frac{D_{\alpha=90^\circ}}{q_f} |\sin \alpha| \sin^2 \alpha \right) \\
\zeta &= q_f \left( \frac{D_{\beta=90^\circ}}{q_f} |\sin \beta| \sin \beta \right) \\
M &= q_f \left( M_{\alpha=90^\circ} |\sin \alpha| \sin \alpha \right) \\
N &= q_f \left( N_{\beta=90^\circ} |\sin \beta| \sin \beta \right) \\
D &= D_\alpha + D_\beta \\
D_\beta &= q_f \left( D_{\beta=90^\circ} |\sin \beta| \sin \beta^2 \right)
\end{align*}
\]

Phasing between the low-angle approximation and the high-angle approximation is based on a complex phase angle and linear interpolation. The complex phase angle is given by:

\[
\pi_{FUS} = \cos^{-1} \left( \frac{u_f}{V_f} \right)
\]

Linear interpolation based on this phase angle is employed over the range of $15^\circ < |\pi_{FUS}| < 30^\circ$. 

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Main rotor interference and transformation $T_g$— The fuselage angles of attack, angle of sideslip, and dynamic pressure include an averaged effect of rotor downwash. The expression for downwash was obtained for typical single rotor helicopters from an empirical fit to data presented in reference 10. The velocities on the fuselage are as follows:

$$u_f = u_B$$
$$v_f = v_B$$
$$w_f = w_B + w_{if}$$

Where $w_{if}$ is the induced velocity due to the rotor and is defined as:

$$w_{if} = \left( \frac{w_{if}}{v_i} \right) v_i$$

$$\left( \frac{w_{if}}{v_i} \right) = 1.299 + 0.671\chi - 1.172\chi^2 + 0.351\chi^3$$

The rotor wake angle $\chi$ is defined as:

$$\chi = \tan^{-1} \left( \frac{\mu}{-\lambda} \right)$$

The angles of attack and sideslip are then:

$$\alpha_f = \tan^{-1} \frac{w_f}{u_f}$$
$$\beta_f = \sin^{-1} \frac{v_f}{v_f}$$

where

$$v_f = \sqrt{u_f^2 + v_f^2 + w_f^2}$$

The dynamic pressure on the fuselage is based on the velocity, including the effect of rotor downwash.

$$q_f = \frac{1}{2} \rho v_f^2$$

Transformation $T_{fg}$— The fuselage forces and moments are calculated in the wind system of axes about the fuselage reference point. The transformation must transfer the moment to the center of gravity and express all forces and moments in the body system of axes.
The forces and moments in the body-c.g. system of axes are given by the following equations:

\[ X_f = -D \cos \beta \cos \alpha - Y \sin \beta \sin \alpha + L \sin \alpha \]

\[ Y_f = Y \cos \beta - D \sin \beta \]

\[ Z_f = -L \cos \alpha - D \cos \beta \sin \alpha - Y \sin \beta \sin \alpha \]

\[ L_f = \ell \cos \beta \cos \alpha - M \sin \beta \cos \alpha - N \sin \alpha + Y_f(WL_{ACF} - WL_{C.g.}) - Z_fBL_{C.g.} \]

\[ M_f = M \cos \beta + \ell \sin \beta - X_f(WL_{ACF} - WL_{C.g.}) + Z_f(STA_{ACF} - STA_{C.g.}) \]

\[ N_f = N \cos \alpha + \ell \cos \beta \cos \alpha - M \sin \beta \sin \alpha + Y_f(STA_{C.g.} - STA_{ACF}) + X_f(BL_{C.g.}) \]
The rpm governor model provides for an rpm degree of freedom with simple engine and governor dynamics. When the rpm governor option is used, the main rotor and tail rotor speeds are changed based on current torque requirements and engine power available. Trim initial conditions establish baseline values of rotor speed and engine torque. Flight variations from that trim condition result in changing torque requirements which cause rotor speed variations which feed through the governor control laws to provide fuel flow changes. The fuel flow changes provide engine power changes which are converted to engine torque available.

The rpm degree of freedom may be described by the equation:

\[ Q_E - Q_R = J \dot{\Omega} \]

where:

- \( Q_E \) = engine torque
- \( Q_R \) = required rotor torque
- \( J \) = rotor rotational inertia
- \( \dot{\Omega} \) = rotor speed rate of change

For low flapping hinge offsets, the rotational inertia, \( J \), may be approximated from the blade flapping inertia, \( I_B \):

\[ J = N I_B \]

where \( N \) is the number of blades.

The main rotor torque required is a complicated function of many variables including blade pitch settings, airspeed, inflow velocity, flapping angles, and rotor speed. The torque equation provides the necessary calculation. The rotor speed governing of this model uses linear control theory based on an operating point. Thus, the rotor speed, \( \Omega \), takes the form:

\[ \Omega = \Omega_0 + \Delta \Omega \]

where \( \Omega_0 \) is the trim rotor speed and \( \Delta \Omega \) is the rotor-speed variation.

A detailed representation of engine torque requires a complicated nonlinear function of many variables including operating power setting, ambient pressure and temperature, and fuel flow, \( W_f \). For this simplified model, rpm governing acts on fuel flow to control engine torque. A gas turbine engine produces power which must be converted to torque based on the current rotor speed.

\[ Q_E = \frac{550}{\Omega} \text{ HP} \]
It is important to note that the current rotor speed is used, not the operating point speed, $\Omega_0$. Thus, the power-to-torque conversion factor is always changing.

The engine power response to fuel flow is simply modeled as a first order lag:

$$\Delta HP = \frac{K_E}{1 + \tau_E s} \Delta \omega_f$$

The time constant, $\tau_E$, and gain, $K_E$, are selected based on engine characteristics and operating point.

The complete rpm degree of freedom and engine-governor model is shown in figure 3.

Note that a throttle on/off switch has been added to allow turning the power on or off. A separate time constant for the throttle is provided to allow for a large power change time constant for the throttle and a small power change time constant for normal engine operation.

The governor control law is given by:

$$\Delta \omega_f = -\left(K_{g1} + \frac{K_{g2}}{s} + K_{g3}s\right)\Delta \Omega$$

which provides proportional, integral, and rate feedback.
Cyclic, collective, and pedal controls may be specified from a zero position either centered or full left or down. The cyclic control position must be consistent between longitudinal and lateral cyclic, that is, centered or full left and forward. The zero position of the collective or pedals may be specified independently. The table below lists the zero position, positive direction, and sign of the moment or force produced in the body system of axes. Alternate control zero position conventions are listed in parentheses.

<table>
<thead>
<tr>
<th>Control</th>
<th>Zero position</th>
<th>Positive direction</th>
<th>Moment/force</th>
</tr>
</thead>
<tbody>
<tr>
<td>Longitudinal cyclic, $\delta_e$</td>
<td>Centered (full forward)</td>
<td>Aft</td>
<td>$+M$</td>
</tr>
<tr>
<td>Lateral cyclic, $\delta_a$</td>
<td>Centered (full left)</td>
<td>Right</td>
<td>$+L$</td>
</tr>
<tr>
<td>Collective, $\delta_c$</td>
<td>Full down</td>
<td>Up</td>
<td>$-Z$</td>
</tr>
<tr>
<td>Pedals, $\delta_p$</td>
<td>Centered (full left) forward</td>
<td></td>
<td>$+N$</td>
</tr>
</tbody>
</table>

The control gearing, rigging, and cyclic control phasing are governed by the following equations:

$$A_{1S} = C_1 \delta_a + C_3 \delta_e + C_{A_{1S}}$$
$$B_{1S} = C_2 \delta_a + C_4 \delta_e + C_{B_{1S}}$$
$$\theta_0 = C_6 \delta_c + C_5$$
$$\theta_{0TR} = C_8 \delta_p + C_7$$

The terms $C_{A_{1S}}$, $C_{B_{1S}}$, $C_5$, and $C_7$ are the constant or rigging terms for each control. The terms $C_1$ through $C_4$ can be used to adjust for the phase angle between the cyclic control input and the resulting flapping. These phase angle terms may be specified in two ways. First, a cyclic control phase relationship may be specified on the basis of main rotor dynamic properties:

$$PP = \frac{K_\beta}{I_\beta \Omega^2} + \frac{cRM_\beta}{gI_\beta} + \frac{\gamma k_1}{8} \left(1 - \frac{4}{3} \epsilon\right)$$

$$C_1 = \frac{1 - (8/3)\epsilon + 2\epsilon^2}{1 - (4/3)\epsilon} \quad (CK2)$$

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The terms CK1 and CK2 are the longitudinal and lateral cyclic control sensitivities, respectively. As an alternative, a control phase angle, \( \psi_0 \), may be specified, which provides the following relationships:

\[
C_1 = (CK2) \cos \psi_0 \\
C_2 = (CK2) \sin \psi_0 \\
C_3 = C_2 \left( \frac{CK1}{CK2} \right) \\
C_4 = -C_1 \left( \frac{CK1}{CK2} \right)
\]
APPENDIX I

LINEARIZED SIX-DEGREE-OF-FREEDOM REPRESENTATION OF HELICOPTER DYNAMICS

The linear, first-order set of differential equations describing the rigid body motion of the helicopter are of the form:

\[
\dot{x} = [F]x + [G]\delta
\]

where \( x \) represents the perturbations from trim of the state variables \( u_B, w_B, q_B, \theta, v_B, p_B, \phi, \) and \( r_B \); and \( \delta \) represents the deviations from the trim control positions of \( \delta_e, \delta_c, \delta_a, \) and \( \delta_p \). The linear representation is valid only if the initial angular velocities \( p_B, q_B, \) and \( r_B \) are zero.

The elements of the \( F \) and \( G \) matrices are of two types. The first type consists of inertial and gravitational terms that can be obtained analytically from the equations of motion. The second type consists of partial derivatives arising from aero-dynamic forces and moments. The force and moment derivatives are obtained by considering both position and negative perturbations from trim. For example:

\[
X_u = \frac{\partial X}{\partial u} = \frac{X(u_0 + \Delta u) - X(u_0 - \Delta u)}{2 \Delta u}
\]

The elements of the \( F \) and \( G \) matrices and the state variable vectors are given in table I-1.
TABLE I-1.- ELEMENTS OF THE LINEARIZED EQUATIONS OF MOTION

\[ X = (Δu, Δw, Δq, Δδ, Δv, Δp, Δφ, Δr)^T \]

\[ δ = (Δδ_ε, Δδ_c, Δδ_a, Δδ_p)^T \]

\[
\begin{bmatrix}
X_u & X_w & X_q & -g \cos \phi_0 & X_v & -g \cos \phi_0 \sin \theta_0 & X_p & 0 & X_r + v_0 \\
\frac{Z_u}{m} & \frac{Z_w}{m} & \frac{Z_q + u_0}{m} & \frac{Z_v}{m} & \frac{Z_p - v_0}{m} & -g \sin \phi_0 \cos \theta_0 & 0 & 0 & \frac{Z_r}{m} \\
M_u & M_w & M_q & M_v & M_p & 0 & 0 & 0 & 0 \\
\frac{I_y}{m} & \frac{I_y}{m} & 0 & \frac{I_y}{m} & \frac{I_y}{m} & 0 & 0 & 0 & \frac{I_y}{m} \\
0 & 0 & \cos \phi_0 & 0 & \cos \phi_0 & 0 & 0 & -\sin \phi_0 & \frac{1}{I_c} \ \\
\frac{(I_x L_u + I_x N_u)}{I_c} & \frac{(I_x L_w + I_x N_w)}{I_c} & \frac{(I_x L_q + I_x N_q)}{I_c} & \frac{(I_x L_v + I_x N_v)}{I_c} & 0 & \frac{(I_x L_p + I_x N_p)}{I_c} & 0 & 0 & \frac{(I_x L_r + I_x N_r)}{I_c} \\
\frac{(I_x L_e + I_x N_e)}{I_c} & \frac{(I_x L_c + I_x N_c)}{I_c} & \frac{(I_x L_a + I_x N_a)}{I_c} & \frac{(I_x L_p + I_x N_p)}{I_c} & 0 & \frac{(I_x L_p + I_x N_p)}{I_c} & 0 & 0 & \frac{(I_x L_r + I_x N_r)}{I_c} \\
\frac{(I_x L_e + I_x N_e)}{I_c} & \frac{(I_x L_c + I_x N_c)}{I_c} & \frac{(I_x L_a + I_x N_a)}{I_c} & \frac{(I_x L_p + I_x N_p)}{I_c} & 0 & \frac{(I_x L_p + I_x N_p)}{I_c} & 0 & 0 & \frac{(I_x L_r + I_x N_r)}{I_c} \\
\end{bmatrix}
\]

\[ I_c = (I_x L_x - I_x^2) \]

\[
\begin{bmatrix}
X_e & X_c & X_a & X_p \\
\frac{Z_e}{m} & \frac{Z_c}{m} & \frac{Z_a}{m} & \frac{Z_p}{m} \\
M_e & M_c & M_a & M_p \\
\frac{I_y}{m} & \frac{I_y}{m} & \frac{I_y}{m} & \frac{I_y}{m} \\
0 & 0 & 0 & 0 \\
\frac{(I_x L_e + I_x N_e)}{I_c} & \frac{(I_x L_c + I_x N_c)}{I_c} & \frac{(I_x L_a + I_x N_a)}{I_c} & \frac{(I_x L_p + I_x N_p)}{I_c} \\
0 & 0 & 0 & 0 \\
\frac{(I_x L_e + I_x N_e)}{I_c} & \frac{(I_x L_c + I_x N_c)}{I_c} & \frac{(I_x L_a + I_x N_a)}{I_c} & \frac{(I_x L_p + I_x N_p)}{I_c} \\
\end{bmatrix}
\]

\[ G = \begin{bmatrix}
\frac{(I_x L_e + I_x N_e)}{I_c} & \frac{(I_x L_c + I_x N_c)}{I_c} & \frac{(I_x L_a + I_x N_a)}{I_c} & \frac{(I_x L_p + I_x N_p)}{I_c} \\
\frac{(I_x L_e + I_x N_e)}{I_c} & \frac{(I_x L_c + I_x N_c)}{I_c} & \frac{(I_x L_a + I_x N_a)}{I_c} & \frac{(I_x L_p + I_x N_p)}{I_c} \\
\frac{(I_x L_e + I_x N_e)}{I_c} & \frac{(I_x L_c + I_x N_c)}{I_c} & \frac{(I_x L_a + I_x N_a)}{I_c} & \frac{(I_x L_p + I_x N_p)}{I_c} \\
\end{bmatrix}\]
APPENDIX J

CONFIGURATION DESCRIPTION REQUIREMENTS

Table J-1 lists the parameters required to describe a helicopter configuration for use in the computer simulation. Listed are the parameter name, algebraic symbol, computer mnemonic, and units for each parameter. In addition, an example value based on an AH-1G is provided for each parameter. AH-1G parameters are based on values in reference 8.
<table>
<thead>
<tr>
<th>Name</th>
<th>Algebraic symbol</th>
<th>Computer mnemonic</th>
<th>Units</th>
<th>Example value</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Main rotor (MR) group</strong></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>MR rotor radius</td>
<td>$R_{MR}$</td>
<td>ROTOR</td>
<td>ft</td>
<td>22</td>
</tr>
<tr>
<td>MR chord</td>
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<td>Number of blades</td>
<td>$n_b$</td>
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<td>$\gamma_{MR}$</td>
<td>GAMMA</td>
<td>N-D</td>
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<td>MR hinge offset</td>
<td>$\epsilon$</td>
<td>EPSLN</td>
<td>percent/100</td>
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<tr>
<td>MR flapping spring constant</td>
<td>$K_\beta$</td>
<td>AKBETA</td>
<td>lb-ft/rad</td>
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<td>MR pitch-flap coupling tangent of $\delta_3$</td>
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<tr>
<td>MR blade twist</td>
<td>$\alpha_{MR}$</td>
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<td>MR precone angle (required for teetering rotor)</td>
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<td>MR solidity</td>
<td>$\sigma_{MR}$</td>
<td>SIGMA</td>
<td>N-D</td>
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<td>$C_{T_{max}}$</td>
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<td><strong>Tail rotor (TR) group</strong></td>
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<td>TR radius</td>
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<td>Units</td>
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<td>Aircraft mass and inertia</td>
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<td>Aircraft weight</td>
<td>( W_{ic} )</td>
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<td>Fus drag, ( \alpha = \beta = 0 )</td>
<td>( \frac{D}{q} )</td>
<td>D1</td>
<td>ft^2</td>
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<td>Fus drag, variation with ( \alpha )</td>
<td>( \frac{2(D/q)}{\partial \alpha} )</td>
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<td>ft^2/rad</td>
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<td>Fus drag, variation with ( \beta )</td>
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<td>Name</td>
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<td>Units</td>
<td>Example value</td>
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<td>Fus pitch moment, $\alpha = \beta = 0$</td>
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<td><strong>Vertical fin (VF)</strong></td>
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<td>in.</td>
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<td>Vertical fin waterline</td>
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<td>AIFF</td>
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<td>VF sweep angle</td>
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<td>Tail rotor induced velocity effect at VF</td>
<td>$k_{VTR}$</td>
<td>XKVTR</td>
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TABLE J-1.- Concluded.

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<th>Name</th>
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<th>Units</th>
<th>Example value</th>
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<tbody>
<tr>
<td>Controls</td>
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<tr>
<td>Swashplate lateral cyclic pitch for zero lateral cyclic stick</td>
<td>$C_{A1S}$</td>
<td>CAIS</td>
<td>rad</td>
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<tr>
<td>Swashplate longitudinal cyclic pitch for zero longitudinal cyclic stick</td>
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<td>CBIS</td>
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<td>Longitudinal cyclic control sensitivity</td>
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REFERENCES


This report documents a helicopter mathematical model suitable for piloted simulation of flying qualities. The mathematical model is a non-linear, total force and moment model of a single main rotor helicopter. The model has ten degrees of freedom: six rigid-body, three rotor-flapping, and the rotor rotational degrees of freedom. The rotor model assumes rigid blades with rotor forces and moments radially integrated and summed about the azimuth. The fuselage aerodynamic model uses a detailed representation over a nominal angle of attack and sideslip range of ±15°, and it uses a simplified curve fit at large angles of attack or sideslip. Stabilizing surface aerodynamics are modeled with a lift curve slope between stall limits and a general curve fit for large angles of attack. A generalized stability and control augmentation system is described. Additional computer subroutines provide options for a simplified engine/governor model, atmospheric turbulence, and a linearized six-degree-of-freedom dynamic model for stability and control analysis.