Tilt-Rotor Flutter Control in Cruise Flight

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SUMMARY

Tilt-rotor flutter control under cruising operation is analyzed. The rotor model consists of a straight fixed wing, a pylon attached to the wingtip, and a three-bladed rotor. The wing is cantilevered to the fuselage and is allowed to bend forward and upward. It also has a torsional degree of freedom about the elastic axis. Each rotor blade has two bending degrees of freedom. Feedback of wingtip velocity and acceleration to cyclic pitch is investigated for flutter control, using strip theory and linearized equations of motion. To determine the feedback gain, an eigenvalue analysis is performed. A second, independent, timewise calculation is conducted to evaluate the control law while employing more sophisticated aerodynamics. The effectiveness of flutter control by cyclic pitch change was confirmed.

NOMENCLATURE

<table>
<thead>
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<th>Symbol</th>
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<tr>
<td>$A$</td>
<td>blade aspect ratio</td>
</tr>
<tr>
<td>$A_{kj}$</td>
<td>defined in equations (55) through (61) for $k = 1, \ldots, 7$</td>
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<tr>
<td>$A_w$</td>
<td>wing aspect ratio</td>
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<tr>
<td>$a_e$</td>
<td>position of elastic axis from leading edge, nondimensionalized by wing chord</td>
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<tr>
<td>$a_i, a_j$</td>
<td>$i$th, $j$th generalized coordinate for wing deformation</td>
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<tr>
<td>$a_{jk}$</td>
<td>$k$th generalize coordinate for the wing's $j$th deformation mode:</td>
</tr>
<tr>
<td>$a_w$</td>
<td>position of pitch axis from leading edge, nondimensionalized by wing chord</td>
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* National Research Council Research Associate
\[ B^c \] matrix, containing virtual work on blade caused by lateral cyclic pitch

\[ B^* \] matrix, containing virtual work on blade caused by longitudinal cyclic pitch

\[ C \] Theodorsen function

\[ C_d \] sectional drag coefficient

\[ C_{l\alpha} \] sectional lift slope

\[ C_m \] sectional pitching moment coefficient

\[ c_g \] position of center of gravity of wing from leading edge, nondimensionalized by wing chord

\[ c \] blade chord

\[ c_w \] wing chord

\[ D \] physical state equation matrix, defined in equations (112) and (114), and in appendix B

\[ d \] drag per unit length

\[ d_w \] wing drag per unit length

\[ E^n \] physical equation matrix for the \( n \)th order time derivative of state vector \( \mathbf{X} \); defined in equations (112) and (113), and in appendix B

\[ EI_B \] blade bending stiffness perpendicular to chordline

\[ EI_C \] blade bending stiffness along chordline

\[ EI_{BW} \] wing bending stiffness perpendicular to chordline

\[ EI_{CW} \] wing bending stiffness along chordline

\[ F_i \] virtual work on wing caused by blade loading through the \( i \)th deformation mode

\[ f_m \] wing torsional moment around the elastic axis (positive nose up)

\[ f_z \] wing loading in x-direction per unit length (positive upward)

\[ f_z \] wing loading in z-direction per unit length (positive forward)

\[ G_i \] virtual work on wing caused by wing loading through the \( i \)th deformation mode

\[ GJ_W \] wing torsional rigidity

\[ G_{W1} \] constant virtual work component on the wing, defined by equation (105)

\[ G_{W_{kj}} \] virtual work component on the wing caused by the wing's \( j \)th deformation mode in term of generalized coordinates \( a_j \) and \( \dot{a}_j \) for \( k = 2, 3 \), respectively; defined by equations (106) and (107)

\[ h \] mast height; distance between rotor and the wing elastic axis

\[ I_9 \] unit matrix of order nine

\[ I_B \] blade flapping mass moment of inertia
constant (independent of generalized coordinates) virtual work component on blade caused by the blade’s jth deformation mode; defined in equation (87)

virtual work components on blade caused by the blade’s jth deformation mode in terms of generalized coordinates \(a_jk\); defined in equations (93) through (97) for \(k = 1, \ldots, 5\)

virtual work component on blade defined in equation (98)

virtual work component on blade defined in equation (99)

virtual work components on blade caused by the wing’s jth deformation mode in terms of generalized coordinates \(q_jk\); defined in equations (88) through (92) for \(k = 1, \ldots, 5\)

pylon pitching mass moment of inertia about the intersection of pylon and wing elastic axis

pylon rolling mass moment of inertia about the intersection of pylon and wing elastic axis

pylon yawing mass moment of inertia about the intersection of pylon and wing elastic axis

wing mass moment of inertia around elastic axis per unit length

virtual work component on wing caused by the blade’s ith deformation mode in terms of generalized coordinates \(q_jk\); defined in appendix A for \(k = 1, \ldots, 5\)

virtual work component on wing caused by cyclic pitch (cosine term); defined in appendix A

virtual work component on wing caused by cyclic pith (sine term); defined in appendix A

virtual work component on wing caused by the wing’s ith deformation mode in terms of generalized coordinates \(q_jk\); defined in appendix A for \(k = 1, \ldots, 5\)

constant virtual work component defined in appendix A

constants, associated with first, second order time derivatives of wing deformation variables - \((u_w, w_w, p_w)\), for lateral and longitudinal cyclic pitch; equations (116), (117)

wing semispan

lift per unit length

blade lift per unit length

harmonic component of lift

steady component of lift

virtual mass component of lift

wing lift per unit length
\( M \)  
\( M_B \)  
\( M_P \)  
\( m \)  
\( m_e \)  
\( m_k \)  
\( m_s \)  
\( m_v \)  
\( m_w \)  
\( N \)  
\( N_{pi} \)  
\( N_{ri} \)  
\( N_{yi} \)  
\( P_i \)  
\( P_W \)  
\( P_{W_i} \)  
\( p_w \)  
\( p_{0n} \)  
\( p_{zn} \)  
\( Q_{kj} \)  
\( q_{jk} \)  
\( q_{ni} \)  
\( R \)  
\( R_C \)  
\( r \)  
\( r_z \)  
\( r_x \)  
\( S \)  
\( S_{Bkj} \)  
\( S_w \)  
\( S_{Wkj} \)
$S_\alpha$ static mass moment of wing around elastic axis per unit length, $S_\alpha = m_w e_w (\alpha_e - c_p)$

$T$ transformation matrix defined in equation (30)

$t$ time

$U$ air velocity at blade section

$U_{Wi}$ ith mode of wing deformation $u_w$

$u_s$ induced velocity

$u_t$ blade tip velocity

$u_w$ wing upward deformation

$V$ wingtip deformation matrix, containing $V_{Wi}$ terms, equation (124)

$V$ airplane cruise velocity

$V_i$ ith mode of blade deformation $v_n$

$V_R$ air velocity at rotor blade section

$V_{Wi}$ ith mode of wing deformation $v_w$

$V_{0i}$ defined in equation (7)

$V_{1i}$ defined in equation (8)

$v_d$ downwash caused by wing

$v_{in}$ in plane air-velocity component relative to blade caused by wing deformation

$v_n$ nth blade inplane deformation (positive in rotational direction)

$v_{out}$ out-of-plane air-velocity component relative to blade caused by wing deformation

$\dot{v}_p$ blade deformation velocity perpendicular to $\vec{r} \times \vec{\Omega} + \vec{V}$

$v_w$ wing deformation in x-direction (positive upward)

$W$ wingtip deformation matrix, containing $W_{Wi}$ terms, equation (125)

$W_i$ ith mode of blade deformation $w$

$W_{Wi}$ ith mode of wing deformation $w_w$

$W_{0i}$ defined in equation (9)

$W_{1i}$ defined in equation (10)

$w_n$ nth blade out-of-plane deformation (positive forward)

$w_w$ wing deformation in z-direction (positive forward)

$X$ state vector, defined in equation (115)

$X_i$ wingtip x-displacement caused by ith mode; equation (41)

$y$ wing spanwise position from aircraft symmetry plane

$Z_i$ wingtip z-displacement caused by ith mode; equation (42)

$\alpha$ angle of attack

$\alpha_G$ geometric angle of attack

$\Delta v$ induced velocity

$\Delta \gamma$ circulation
\( \eta \) correction factor, given by \( \eta_1 \eta_2 \)
\( \eta_1 \) aspect ratio correction factor
\( \eta_2 \) correction factor for spanwise location of vortex trailing
\( \theta \) blade pitch angle
\( \theta_c \) lateral cyclic pitch
\( \theta_n \) pitch control input (collective and cyclic) for \( n \)th blade
\( \theta_s \) longitudinal cyclic pitch
\( \theta_w \) wing pitch angle
\( \theta_0 \) collective pitch
\( \rho \) air density
\( \nu_p \) pylon pitching angle (positive nose up)
\( \nu_r \) pylon rolling angle (positive clockwise)
\( \nu_y \) pylon yawing angle (positive counterclockwise)
\( \phi \) inflow angle
\( \phi_w \) inclination of wing lift
\( \psi_n \) azimuth angle of \( n \)th blade
\( \Omega \) rotor rotational velocity
\( \omega_i \) rotor \( i \)th modal natural frequency
\( \omega_{wi} \) wing \( i \)th modal natural frequency
\( (\cdot)^t \) transposed matrix
\( (\cdot) \) complex variable
\( (\cdot), (\cdot) \) time derivative \( \frac{\partial}{\partial t}, \frac{\partial^2}{\partial t^2} \)
\( (\cdot)', (\cdot)'' \) spanwise derivative \( \frac{\partial}{\partial y}, \frac{\partial^2}{\partial y^2} \)
\( (\cdot)'' \) or radius-wise derivative \( \frac{\partial}{\partial r}, \frac{\partial^2}{\partial r^2}, \frac{\partial^3}{\partial r^3} \)
\( (\cdot)\cos, (\cdot)\sin \) \( \cos(\psi_n) \) term; cyclic pitch
\( (\cdot)_{i}, (\cdot)_{j} \) deformation mode \( i, j \)
\( (\cdot)_{n} \) \( n \)th rotor blade
\( (\cdot)_{0} \) constant term, i.e., independent of \( \psi_n \); collective pitch
\( (\cdot)_{\sin}, (\cdot)_{\cos} \) \( \sin(\psi_n) \) term; cyclic pitch
\( (\cdot)_{tip} \) wing tip
\( \theta_0 \) zero square matrix of order nine

Superscript
\( c \) lateral cyclic pitch
\( I \) inverse matrix
\( s \) longitudinal cyclic pitch
\( 0 \) collective pitch

6
INTRODUCTION

The tilt-rotor aircraft is one of the promising transportation systems of the near future. It realizes a unique flight capability with its configuration. In the current trend toward using active control systems to widen the allowable flight conditions in fixed-wing aircraft or in conventional rotorcraft, it is of interest to consider active controls for the tilt-rotor aircraft to expand its flight envelope, maneuverability and versatility. Active control systems for rotorcraft have been studied for years, but until recently almost all such studies dealt with active control systems to reduce the vehicle's gust response. It may be said that those studies have been aimed at coping with unusual phenomena which occur under normal operating conditions. In contrast, the objective of the flutter control studies is to enable the aircraft or rotorcraft to fly at higher speeds, to dive more steeply, or to allow a reduction of structural weight. In this report, tilt-rotor flutter control using rotor cyclic pitch is treated using the governing equations of reference 1. Determination of the control law is based on a harmonic method using wing tip deflection as the feedback. The control law determined in this way is further examined by timewise calculation using the local circulation method, where more precise aerodynamic theories are involved (references 2 through 5).

Subscripts

B, b blade

c lateral cyclic pitch

d indicates constant associated with first order time derivative

dd indicates constant associated with second order time derivative

i, j, k indices

p wing torsional deformation

s longitudinal cyclic pitch

tip wingtip

u wing upward deformation

W, w wing

w in K-constants: wing chordwise deformation

x in x-direction

z in z-direction
TILT-ROTOR MODEL AND EQUATIONS OF MOTION

Figure 1 illustrates the tilt-rotor model considered here, which consists of a cantilevered wing, a pylon, and rotor blades. This model is for airplane cruise mode, which is the major concern in this report. Operation in helicopter mode or transition mode is not considered. The wing has one torsional and two bending degrees of freedom about the elastic axis. Wing dihedral angle and sweep angle are not considered. The pylon is rigidly attached at the wingtip, its direction is assumed to be parallel to the cruising velocity, and at the end of the pylon is a hingeless three-bladed rotor. The rotor blade also has a straight elastic axis, with elastic lead-lag and flapping degrees of freedom. In both the wing and the rotor blade, one of the principal elastic axes for bending coincides with the zero-lift- or chord-line. Aircraft motion is not considered. Since the wing is cantilevered to the fuselage, wing antisymmetric motion is also neglected (reference 6).

One coordinate system is assumed fixed to the aircraft (figure 2); consequently, it is an inertia system. The X-axis is directed upward. The Y-axis coincides with the wing elastic axis. The Z-axis is directed forward. The aircraft cruising velocity $V$ is in the Z direction, but the wing-elastic principal axes do not necessarily coincide with the X- and Z-axes. The origin of this coordinate system is located at the intersection of wing elastic axis and aircraft-symmetry plane. The rotor is located at a distance $h$ forward from the wing elastic axis at the wingtip.

Figure 3 shows the wing and blade deflections. The wing deflection is expressed in the X- and Z-direction, $u_w$ and $w_w$ (positive upward and forward, respectively), and torsion about the elastic axis, $p_w$ (positive nose up). The deflected position of the pylon root is described as $(u_{w,\text{tip}}, L, w_{w,\text{tip}})$ or $(r_z, L, r_x)$. The pylon direction is given by yawing, pitching, and rolling angles that are denoted by $\nu_y$, $\nu_p$, and $\nu_r$, respectively, which are equal to $w'_{w,\text{tip}}$, $p_{w,\text{tip}}$, and $-w'_{w,\text{tip}}$, respectively. The rotor rotates clockwise for an observer looking forward and the nth rotor blade azimuth angle $\psi_n$ is measured from the vertical position. This reference direction moves with the pylon rolling motion, so the blade azimuth angle in reference to the inertia system is given by $\psi_n + \nu_r$ (figure 4). Blade deflection is expressed in terms of in-the-rotational-plane $v_n$ (positive in the rotational direction) and out-of-the-rotational-plane $w_n$ (positive forward) bending.

The blade and wing deflections are described in series of natural modes (reference 7).

$$v_n = \sum_i q_{ni}(t)V_i(r)$$  \hspace{1cm} (1)

$$w_n = \sum_i q_{ni}(t)W_i(r)$$  \hspace{1cm} (2)
The equations of motion for the blade and wing generalized coordinates $q_{ni}$ and $a_i$ are taken from reference 1.

The equations for the deformation of the $n$th rotor blade are

$$
\ddot{q}_{ni} + \omega_i^2 q_{ni} + V_{1i} \ddot{\nu}_r - V_{0i} \left( \frac{\ddot{r}_z}{R} + \frac{h}{R} \ddot{\nu}_p \right) \sin \psi_n + W_{0i} \ddot{r}_z - \frac{h}{R} V_{0i} \ddot{\nu}_y \cos \psi_n + W_{1i} \ddot{\nu}_y \sin \psi_n - W_{1i} \ddot{\nu}_p \cos \psi_n + 2\Omega W_{1i} \ddot{\nu}_y \cos \psi_n + 2\Omega W_{1i} \ddot{\nu}_p \sin \psi_n = P^i_n
$$

(6)

$$
V_{0i} = R \int_0^R mV_i \, dr
$$

(7)

$$
V_{1i} = \int_0^R mV_i \, dr
$$

(8)

$$
W_{0i} = R \int_0^R mW_i \, dr
$$

(9)

$$
W_{1i} = \int_0^R mW_i \, dr
$$

(10)

where $P^i_n$ is virtual work done by the forces on the $n$th blade through the $i$th deformation mode.

For wing deformation the equation is

$$
\ddot{a}_i + \omega_{wi}^2 a_i + I_B N \Omega (W'_{W_i,tip} \ddot{\nu}_p - P_{W_i,tip} \ddot{\nu}_y)
$$

$$
+ \sum_{n=1}^N \sum_j \left\{ -V_{1j} U_{W_j,tip} + W_{0j} \frac{W_{W_j,tip}}{R} \right\} \ddot{q}_{nj}
$$

$$
+ \left[ -V_{0j} \left( \frac{U_{W_j,tip}}{R} + \frac{h}{R} P_{W_j,tip} \right) + W_{1j} W'_{W_j,tip} \right] \ddot{q}_{nj} \sin \psi_n
$$

9
\[-(V_{ij} \frac{h}{R} + W_{1j} P_{W_j,tip}) \dot{q}_{nj} \cos \psi_n \]
\[-2\Omega V_{ij} \left( \frac{U_{W_j,tip}}{R} + \frac{h}{R} P_{W_j,tip} \right) \dot{q}_{nj} \cos \psi_n \]
\[+ 2\Omega \left( \frac{h}{R} V_{ij} W_{W_j,tip}' - W_{1j} P_{W_j,tip} \right) \dot{q}_{nj} \sin \psi_n \]
\[+ \Omega^2 \left( \frac{h}{R} V_{ij} W_{W_j,tip}' - W_{1j} P_{W_j,tip} \right) q_{nj} \cos \psi_n \]
\[+ \Omega^2 \left( V_{ij} \left( \frac{U_{W_j,tip}}{R} + \frac{h}{R} P_{W_j,tip} \right) + W_{1j} W_{W_j,tip}' \right) q_{nj} \sin \psi_n \right\} \]
\[= F_i + G_i \quad (11) \]

where \( F_i \) and \( G_i \) are the virtual work done by forces on blade and wing, respectively, through the \( i \)th deformation mode. Natural modes \( V_i, W_i, U_{W_i}, W_{W_i}, \) and \( P_{W_i} \), and corresponding frequencies \( \omega_i \) and \( \omega_{wi} \) satisfy the following equations of motion and boundary and normalization conditions.

Equations of motion for the natural modes of vibration are

Blade:

\[\frac{d^2}{dr^2} \left[ \left( EI_C \sin^2 \theta + EI_B \cos^2 \theta \right) \frac{d^2 V_i}{dr^2} \right] \]
\[+ \frac{d^2}{dr^2} \left[ \left( EI_C - EI_B \right) \sin^2 \theta \cos^2 \theta \frac{d^2 V_i}{dr^2} \right] \]
\[- \frac{d}{dr} \left( \int_{r}^{R} m z \frac{dV_i}{dr} \right) - \omega_i^2 m W_i = 0 \quad (12) \]

\[\frac{d^2}{dr^2} \left[ \left( EI_C \cos^2 \theta + EI_B \sin^2 \theta \right) \frac{d^2 W_i}{dr^2} \right] \]
\[+ \frac{d^2}{dr^2} \left[ \left( EI_C - EI_B \right) \sin^2 \theta \cos^2 \theta \frac{d^2 W_i}{dr^2} \right] \]
\[- \frac{d}{dr} \left( \int_{r}^{R} m z \frac{dW_i}{dr} \right) - m(\omega_i^2 + \Omega^2) V_i = 0 \quad (13) \]

Wing:

\[\frac{d^2}{dy^2} \left[ \left( EI_{CW} \sin^2 \theta_w + EI_{BW} \cos^2 \theta_w \right) \frac{d^2 U_{Wi}}{dy^2} \right] \]
\[ \frac{d^2}{dy^2} \left[ (EI_{CW} - EI_{BW}) \sin^2 \theta_w \cos^2 \theta_w \frac{d^2 W_{Wi}}{dy^2} \right] - \omega_{wi}^2 (m_w - S_\alpha P_{Wi}) = 0 \]  
(14)

\[ \frac{d^2}{dy^2} \left[ (EI_{CW} \cos^2 \theta_w + EI_{BW} \sin^2 \theta_w) \frac{d^2 W_{Wi}}{dy^2} \right] + \frac{d^2}{dy^2} \left[ (EI_{CW} - EI_{BW}) \sin^2 \theta_w \cos^2 \theta_w \frac{d^2 U_{Wi}}{dy^2} \right] \]

\[- \omega_{wi}^2 m_w W_{Wi} = 0 \]
(15)

\[ \frac{d}{dy} \left( GJ_w \frac{dP_{Wi}}{dy} \right) - \omega_{wi}^2 (I_w P_{Wi} + S_\alpha U_{Wi}) = 0 \]
(16)

The equations for the boundary conditions are

At the bladetip:

\[ V_i'' = 0, \quad V_i''' = 0, \quad W_i'' = 0, \quad W_i''' = 0 \]
(17)

At the bladeroot:

\[ V_i = 0, \quad V_i' = 0, \quad W_i = 0, \quad W_i' = 0 \]
(18)

At the wingroot:

\[ U_{Wi} = 0, \quad U_{Wi}' = 0, \quad W_{Wi} = 0, \quad W_{Wi}' = 0, \quad P_{Wi} = 0 \]
(19)

At the wingtip

\[ M_P \ddot{r}_z - \frac{\partial}{\partial y} \left[ (EI_{CW} \sin^2 \theta_w + EI_{BW} \cos^2 \theta_w) \frac{\partial^2 U_{Wi}}{\partial y^2} \right] \]

\[ - \frac{\partial}{\partial y} \left[ (EI_{CW} - EI_{BW}) \sin \theta_w \cos \theta_w \frac{\partial^2 W_{Wi}}{\partial y^2} \right] \]

\[ + NM_B \ddot{r}_z + NM_B h \ddot{v}_p = 0 \]
(20)
The equations for the normalization conditions are

Blade:

$$- \frac{\partial}{\partial y} \left[ (EI_{CW} - EI_{BW}) \sin \theta_w \cos \theta_w \frac{\partial^2 W_{Wi}}{\partial y^2} \right] + N M_B \ddot{r}_z = 0 \quad (21)$$

$$I_p y \ddot{v}_y + (EI_{CW} \cos^2 \theta_w + EI_{BW} \sin^2 \theta_w) \frac{\partial^2 U_{Wi}}{\partial y^2}$$

$$+ (EI_{CW} - EI_{BW}) \sin \theta_w \cos \theta_w \frac{\partial^2 W_{Wi}}{\partial y^2}$$

$$+ N M_B h^2 \ddot{v}_y + \frac{N}{2} I_B \ddot{v}_y = 0 \quad (22)$$

$$I_p \ddot{v}_p + G J_W \frac{\partial P_{Wi}}{\partial y} + N M_B h^2 \ddot{v}_p + \frac{N}{2} I_B \ddot{v}_p = 0 \quad (23)$$

$$I_p r \ddot{v}_r - (EI_{CW} \sin^2 \theta_w + EI_{BW} \cos^2 \theta_w) \frac{\partial^2 U_{Wi}}{\partial y^2}$$

$$- (EI_{CW} - EI_{BW}) \sin \theta_w \cos \theta_w \frac{\partial^2 W_{Wi}}{\partial y^2} + N I_B \ddot{v}_r = 0 \quad (24)$$

Wing:

$$\int_0^R m (V_i^2 + W_i^2) \, dr = 1 \quad (25)$$

$$\int_0^L \left[ m_w \left( U_{Wi}^2 + W_{Wi}^2 \right) + I_w P_{Wi}^2 + 2 S_w U_{Wi} P_{Wi} \right] \, dy$$

$$+ \left[ M_p \left( U_{Wi}^2 + W_{Wi}^2 \right) + I_{py} (W_{Wi}')^2 + I_{pp} P_{Wi}^2 + I_{pr} (U_{Wi}')^2 \right]_{y=L}$$

$$+ N M_B \left[ (U_{Wi} + h P_{Wi})^2 + h^2 (W_{Wi}')^2 + W_{Wi}^2 \right]_{y=L}$$

$$+ \frac{N I_B}{2} \left[ 2 (U_{Wi}')^2 + (W_{Wi}')^2 + P_{Wi}^2 \right]_{y=L} = 1 \quad (26)$$

These equations do not have the expressions for the virtual work terms $P_n^i$, $F_i$ and $G_i$. These virtual work terms are directly related to the air loading and can be calculated under
various assumptions such as quasi-steady or unsteady aerodynamics, lifting line theory, or lifting surface theory. There are many possible options in calculating aerodynamic loading, and some of them restrict the validity of the final equations. For that reason, details of the air loading estimation are not given in this chapter and air loadings are assumed to be given. Methods to calculate the air loading to complete the equations and to enable it to be numerically calculated are given in the following chapters. In the actual aircraft, gravitational force also contributes, but this effect is neglected in this report. However, the gravitational force depends only on the mass distribution, so it could be easily included.

Since the blade is rigid in torsion, only lift and drag are considered in the blade equation of motions. The direction of lift and drag are determined by the direction of the inflow, including the rotor-induced velocity, which also depends upon the method of air loading estimation. To avoid additional assumptions, the blade air loading is given by the components in the rotational plane \( p_{\varphi n} \) (positive in the rotational direction) and out-of-rotational plane \( p_{zn} \) (positive forward), in the same directions as those used in defining the blade deformation. The forces acting in the radial direction are not considered since the blade is assumed to neither stretch nor shrink in that direction. Thus the virtual work in the equations of motion for the blade is given by

\[
P^i_n = \int_0^R (p_{\varphi n} V_i + p_{zn} W_i) \, dr \tag{27}
\]

As for the case of the blade, wing loading is also given without specifying any aerodynamic theory. It is given in cruising direction \( f_z \) (positive forward) and upward direction \( f_z \). The wing has a torsional degree of freedom and the torsional moment about the elastic axis is given by \( f_m \) (positive nose up). Then the virtual work \( G_i \) is

\[
G_i = \int_0^L (f_z U_{W_i} + f_z W_{W_i} + m_e P_{W_i}) \, dy \tag{28}
\]

The virtual work on the wing by the blade air loading \( F_i \) is given using the wingtip positions \( U_{W_i,tip} \) and \( W_{W_i,tip} \), and its rotation \( P_{W_i,tip} \).

\[
F_i = \sum_{n=1}^N \int_0^R \left\{ \left( \begin{array}{c} U_{W_i,tip} \\ W_{W_i,tip} \end{array} \right)^t (T) \left( \begin{array}{c} 0 \\ 0 \\ p_{\varphi n} \\ p_{zn} \end{array} \right) \right. \\
+ \left( \begin{array}{c} W'_{W_i,tip} \\ P_{W_i,tip} \\ -U'_{W_i,tip} \end{array} \right)^t (T) \left[ \begin{array}{c} r \\ v_n \\ h + w_n \end{array} \right] \left( \begin{array}{c} 0 \\ p_{\varphi n} \\ p_{zn} \end{array} \right) \left\} \, dr \tag{29}
\]
The variable \((T)\) is the transformation matrix from the rotating coordinate system fixed to the blade before deformation to the inertia coordinate system fixed to the aircraft and is given by
\[
(T) = \begin{pmatrix}
1 & -\nu_r & \nu_p \\
\nu_r & 1 & -\nu_y \\
-\nu_p & \nu_y & 1
\end{pmatrix}
\begin{pmatrix}
\cos \psi_n & -\sin \psi_n & 0 \\
\sin \psi_n & \cos \psi_n & 0 \\
0 & 0 & 1
\end{pmatrix}
\]

(30)

Blade and wing deformation are expressed by the summation of natural modes for each degree of freedom.

\[
w_n = \sum_{i=1}^{2} W_i(r)q_{n_i}(t)
\]

(31)

\[
v_n = \sum_{i=1}^{2} V_i(r)q_{n_i}(t)
\]

(32)

\[
u_w = \sum_{i=1}^{3} U_W(y) a_i(t)
\]

(33)

\[
u_w = \sum_{i=1}^{3} W_W(y) a_i(t)
\]

(34)

\[
u_w = \sum_{i=1}^{3} P_W(y) a_i(t)
\]

(35)

\[
u_w = \sum_{i=1}^{3} X_i a_i(t)
\]

(36)

\[
u_w = \sum_{i=1}^{3} Y_i a_i(t)
\]

(37)

\[
u_w = \sum_{i=1}^{3} N_y i a_i(t)
\]

(38)

\[
u_w = \sum_{i=1}^{3} N_p i a_i(t)
\]

(39)

\[
u_w = \sum_{i=1}^{3} N_r i a_i(t)
\]

(40)
Even when a simple model is assumed, such as a fully articulated, nonelastic blade, the following formulation does not become simple because of the high inflow ratio. Each blade is controlled by cyclic and collective pitch; thus \( \theta_n \) and \( q_{ni} \) are composed of three components

\[
\begin{align*}
\theta_n &= \theta_0(t) + \theta_s(t) \sin \psi_n + \theta_c(t) \cos \psi_n \\
q_{ni} &= q_{i0}(t) + q_{is}(t) \sin \psi_n + q_{ic}(t) \cos \psi_n
\end{align*}
\]

Substituting those relations into the equations of motion, the following relations are obtained

Blade:

\[
\begin{align*}
\ddot{q}_{i0} + \omega_i^2 q_{i0} \\
+ (\ddot{q}_{is} - 2\Omega \dot{q}_{ic} - \Omega^2 q_{is} + \omega_i^2 q_{is}) \sin \psi_n \\
+ (\ddot{q}_{ic} + 2\Omega \dot{q}_{is} - \Omega^2 q_{ic} + \omega_i^2 q_{ic}) \cos \psi_n \\
+ \sum_{j=1}^{3} [Q_{1j} \ddot{a}_j + (Q_{2j} \ddot{a}_j + Q_{3j} \dot{a}_j) \sin \psi_n + (Q_{4j} \ddot{a}_j + Q_{5j} \dot{a}_j) \cos \psi_n] = P_n^i
\end{align*}
\]
\[ \ddot{a}_t + \omega^2_{\psi t} a_t + I_B N \Omega \sum_{j=1}^{3} (N_{yi} N_{pj} - N_{pi} N_{yj}) \dot{a}_j \\
+ \sum_{n=1}^{N} \sum_{j=1}^{2} \left\{ A_{1j}^i \dot{q}_{j0} + \sin \psi_n [A_{1j}^i (\dot{q}_{is} - 2\Omega \dot{q}_{ic} - \Omega^2 q_{ic}) + A_{2j}^i \dot{q}_{i0} + A_{4j}^i \dot{q}_{i0} + A_{6j}^i q_{i0}] \\
+ \cos \psi_n [A_{1j}^i (\dot{q}_{ic} + 2\Omega \dot{q}_{is} - \Omega^2 q_{ic}) + A_{3j}^i \dot{q}_{i0} + A_{5j}^i \dot{q}_{i0} + A_{7j}^i q_{i0}] \\
+ \frac{1 - \cos 2\psi_n}{2} \left[A_{2j}^i (\dot{q}_{is} - 2\Omega \dot{q}_{ic} - \Omega^2 q_{ic}) + A_{4j}^i (\dot{q}_{is} - \Omega q_{ic}) + A_{6j}^i q_{is} \right] \\
+ \frac{1 + \cos 2\psi_n}{2} \left[A_{3j}^i (\dot{q}_{ic} + 2\Omega \dot{q}_{is} - \Omega^2 q_{ic}) + A_{5j}^i (\dot{q}_{ic} + \Omega q_{is}) + A_{7j}^i q_{ic} \right] \\
+ \frac{1}{2} \sin 2\psi_n \left[A_{2j}^i (\dot{q}_{ic} + 2\Omega \dot{q}_{is} - \Omega^2 q_{ic}) + A_{3j}^i (\dot{q}_{is} - 2\Omega \dot{q}_{ic} - \Omega^2 q_{is}) \\
+ A_{4j}^i (\dot{q}_{ic} + \Omega q_{ic}) + A_{5j}^i (\dot{q}_{is} - \Omega q_{ic}) + A_{6j}^i q_{ic} + A_{7j}^i q_{is} \right] \right\} \]
\[ = F_i + G_i \quad (54) \]

\[ A_{1j}^i = V_{1j} N_{rt} + W_{0j} \frac{Z_i}{R} \quad (55) \]
\[ A_{2j}^i = -\frac{V_{0j}}{R} (X_i + h N_{pi}) + W_{1j} N_{yi} \quad (56) \]
\[ A_{3j}^i = -\frac{V_{0j}}{R} h N_{yi} - W_{1j} N_{pi} \quad (57) \]
\[ A_{4j}^i = -2\Omega \frac{h}{R} V_{0j} N_{yi} \quad (58) \]
\[ A_{5j}^i = -2\Omega \frac{V_{0j}}{R} (X_i + h N_{pi}) \quad (59) \]
\[ A_{6j}^i = \Omega^2 \left[ \frac{V_{0j}}{R} (X_i + h N_{pi}) + W_{1j} N_{yi} \right] \quad (60) \]
\[ A_{7j}^i = \frac{1}{R} \Omega^2 (h V_{0j} N_{yi} - RW_{1j} N_{pi}) \quad (61) \]

When one compares the terms multiplying the constant (independent of $\psi_n$), $\sin \psi_n$ and $\cos \psi_n$ terms of equation (48) and the constant term of equation (54), the equations defining
\( q_{i0}, q_{is}, q_{ic}, \) and \( a_i \) are obtained
\[
\ddot{q}_{i0} + \omega^2 q_0 + \sum_{j=1}^{3} Q_{ij} \ddot{a}_j = (P^i_n)_0
\]  
(62)

\[ q_{is} - 2\Omega \dot{q}_{is} - (\Omega^2 - \omega^2)q_{is} + \sum_{j=1}^{3} (Q_{ij} \ddot{a}_j + Q_{3j} \dot{a}_j) = (P^i_n)_{\text{sin}} \]  
(63)

\[ q_{ic} + 2\Omega \dot{q}_{ic} - (\Omega^2 - \omega^2)q_{ic} + \sum_{j=1}^{3} (Q_{ij} \ddot{a}_j + Q_{5j} \dot{a}_j) = (P^i_n)_{\text{cos}} \]  
(64)

\[
\ddot{a}_i + \omega_{wi}^2 a_i + \sum_{j=1}^{3} I_B N \Omega (N y_i N p_j - N p_i N y_j) \ddot{a}_j \\
+ \sum_{j=1}^{2} \left[ NA^i_{1j} \ddot{q}_{j0} + \frac{NA^i_{2j}}{2} \ddot{q}_j + \frac{NA^i_{3j}}{2} \bar{q}_j \right] \\
+ \frac{N}{2} (A^i_{0j} + 2\Omega A^i_{3j}) \dot{q}_s + \frac{N}{2} (A^i_{5j} - 2\Omega A^i_{2j}) \dot{q}_{jc} \\
+ \frac{N}{2} (A^i_{0j} + \Omega A^i_{5j} - \Omega^2 A^i_{2j}) q_j \\
+ \frac{N}{2} (A^i_{4j} - \Omega A^i_{4j} - \Omega^2 A^i_{3j}) q_{jc} \\
= (F_i + G_i)_0
\]  
(65)

where subscripts \((\ )_0\), \((\ )_{\text{sin}}\), and \((\ )_{\text{cos}}\) indicate constant (independent of azimuth angle) terms, \( \sin \psi_n \), and \( \cos \psi_n \) terms, respectively, which are considered next. The number of blades \( N \) is assumed to be odd and greater than two in these equations, but the calculations for an even-bladed rotor can be done in a similar manner.

The virtual work terms on the right hand side of equations of motion (62) through (65), need to be determined next. The terms should be in a linear form to apply the harmonic method. The deflection itself, rather than the deflection from the equilibrium state, is used in the linearization. Thus, the coefficients of generalized coordinates (i.e., coefficients for the first order in Taylor expansion) are different from those at a deflected equilibrium point. However, the equilibrium value is not correct under the simplifications involved in
the air loading calculations, especially using a simple aerodynamic theory. This approach was chosen to avoid additional assumptions and to make the following calculations simple. Figure 5 gives schematic views of the air loading geometry on the blade and the wing. The blade aerodynamic force is calculated using quasi-steady theory. The Prandtl-Glauert rule is applied to correct for compressibility effects. The blade lift and drag per unit span, \( l \) and \( d \), respectively, and the in-plane and out-of-plane aerodynamic forces \( p_{\theta n} \) and \( p_{zn} \) are written as

\[
\begin{align*}
\theta_n &= -i \sin \phi - d \cos \phi \\

d_n &= l \cos \phi - d \sin \phi \\
l &= \frac{1}{2\sqrt{1-M^2}} \rho V_R^2 c C_{l\alpha} \alpha \\
d &= \frac{1}{2} \rho V_R^2 c C_d \\
V_R &= \sqrt{V^2 + (r\Omega)^2} \\
\phi &= \tan^{-1} \left( \frac{V}{r\Omega} \right)
\end{align*}
\]

The blade-angle-of-attack equation is

\[
\alpha = \theta - \phi + \frac{1}{V_R} (v_{in} \sin \phi - v_{out} \cos \phi)
\]

\[
v_{in} = \dot{v}_r + \dot{v}_x + (-\dot{r}_x - h \dot{v}_p + V \nu_x) \sin \psi_n + (-h \dot{v}_y + V \nu_y) \cos \psi_n
\]

\[
v_{out} = \dot{w}_r + \dot{w}_x + \dot{r}_x \sin \psi_n - \dot{r}_y \cos \psi_n
\]

where \( v_{in} \) and \( v_{out} \) are the air-velocity components relative to the blade caused by wing deformation, and \( \phi \) is the inflow angle. The induced velocity is neglected, since it is small in comparison with cruising velocity, and stability not static deflections is the major concern of this study. When \( v_{in} \) and \( v_{out} \) are written in terms of natural modes and Fourier-type generalized coordinates, the following relations are obtained

\[
\begin{align*}
v_{in} \sin \phi - v_{out} \cos \phi &= \sum_{j=1}^{2} S_{B1j} \dot{q}_{j0} + \sum_{j=1}^{3} S_{W1j} \dot{a}_j + \\
&\quad + \sin \psi_n \left( \sum_{j=1}^{2} S_{B2j} \dot{q}_{js} + \sum_{j=1}^{2} S_{B3j} \dot{q}_{jc} + \sum_{j=1}^{3} S_{W2j} \dot{a}_j + \sum_{j=1}^{3} S_{W3j} \dot{a}_j \right) + \\
&\quad + \cos \psi_n \left( \sum_{j=1}^{2} S_{B4j} \dot{q}_{jv} + \sum_{j=1}^{2} S_{B5j} \dot{q}_{jz} + \sum_{j=1}^{3} S_{W4j} \dot{a}_j + \sum_{j=1}^{3} S_{W4j} \dot{a}_j \right)
\end{align*}
\]
The relations (66) through (85) give blade and wing air loading in terms of the generalized coordinates. Substituting equations (66) and (67) into (27), the virtual work on the blade caused by the blade air loading is given as follows:

\[
P_n^i = I_{B0}^i + \sum_{j=1}^{2} I_{B1j}^i \dot{q}_{j0} + \sum_{j=1}^{3} I_{W1j}^i \dot{a}_{j} + \left( \sum_{j=1}^{2} I_{B2j}^i \dot{q}_{jc} + \sum_{j=1}^{2} I_{B3j}^i q_{jc} + \sum_{j=1}^{3} I_{W2j}^i \dot{a}_{j} - \sum_{j=1}^{3} I_{W3j}^i a_{j} + I_{S}^i \theta_{c} \right) \sin \psi_n + \left( \sum_{j=1}^{2} I_{B4j}^i \dot{q}_{jc} + \sum_{j=1}^{2} I_{B5j}^i q_{jc} + \sum_{j=1}^{3} I_{W4j}^i \dot{a}_{j} - \sum_{j=1}^{3} I_{W5j}^i a_{j} + I_{C}^i \theta_{c} \right) \cos \psi_n \tag{86}
\]

\[
I_{B0}^i = -\frac{1}{2} \rho C_{la} \int_{0}^{R} \frac{1}{\sqrt{1 - M^2}} V_{R}^2(\theta - \phi)(V_{i} \sin \phi - W_{i} \cos \phi) \, dr
- \frac{1}{2} \rho \int_{0}^{R} c_{d} V_{R}^2(V_{i} \cos \phi + W_{i} \sin \phi) \, dr \tag{87}
\]

\[
I_{B1j}^i = -\frac{1}{2} \rho C_{la} \int_{0}^{R} \frac{1}{\sqrt{1 - M^2}} V_{R} c S_{B1j}(V_{i} \sin \phi - W_{i} \cos \phi) \, dr \tag{88}
\]

\[
I_{B2j}^i = -\frac{1}{2} \rho C_{la} \int_{0}^{R} \frac{1}{\sqrt{1 - M^2}} V_{R} c S_{B2j}(V_{i} \sin \phi - W_{i} \cos \phi) \, dr \tag{89}
\]

\[
I_{B3j}^i = -\frac{1}{2} \rho C_{la} \int_{0}^{R} \frac{1}{\sqrt{1 - M^2}} V_{R} c S_{B3j}(V_{i} \sin \phi - W_{i} \cos \phi) \, dr \tag{90}
\]

\[
S_{B1j} = V_{j} \sin \phi - W_{j} \cos \phi \tag{76}
\]

\[
S_{B2j} = S_{B1j} \tag{77}
\]

\[
S_{B3j} = -\Omega S_{B1j} \tag{78}
\]

\[
S_{B4j} = S_{B1j} \tag{79}
\]

\[
S_{B5j} = \Omega S_{B1j} \tag{80}
\]

\[
S_{W1j} = r N_{rj} \sin \phi - Z_{j} \cos \phi \tag{81}
\]

\[
S_{W2j} = -(X_{j} + h N_{pj}) \sin \phi - r N_{yj} \cos \phi \tag{82}
\]

\[
S_{W3j} = V N_{pj} \sin \phi \tag{83}
\]

\[
S_{W4j} = -h N_{yj} \sin \phi + r N_{pj} \cos \phi \tag{84}
\]

\[
S_{W5j} = V N_{yj} \sin \phi \tag{85}
\]
\[ I_{B4j} = -\frac{1}{2} \rho C_{l\alpha} \int_{0}^{R} \frac{1}{\sqrt{1 - M^2}} V_{Rc} S_{B4j} (V_i \sin \phi - W_i \cos \phi) \, dr \]  
(91)

\[ I_{B5j} = -\frac{1}{2} \rho C_{l\alpha} \int_{0}^{R} \frac{1}{\sqrt{1 - M^2}} V_{Rc} S_{B5j} (V_i \sin \phi - W_i \cos \phi) \, dr \]  
(92)

\[ I_{W1j} = -\frac{1}{2} \rho C_{l\alpha} \int_{0}^{R} \frac{1}{\sqrt{1 - M^2}} V_{Rc} S_{W1j} (V_i \sin \phi - W_i \cos \phi) \, dr \]  
(93)

\[ I_{W2j} = -\frac{1}{2} \rho C_{l\alpha} \int_{0}^{R} \frac{1}{\sqrt{1 - M^2}} V_{Rc} S_{W2j} (V_i \sin \phi - W_i \cos \phi) \, dr \]  
(94)

\[ I_{W3j} = -\frac{1}{2} \rho C_{l\alpha} \int_{0}^{R} \frac{1}{\sqrt{1 - M^2}} V_{Rc} S_{W3j} (V_i \sin \phi - W_i \cos \phi) \, dr \]  
(95)

\[ I_{W4j} = -\frac{1}{2} \rho C_{l\alpha} \int_{0}^{R} \frac{1}{\sqrt{1 - M^2}} V_{Rc} S_{W4j} (V_i \sin \phi - W_i \cos \phi) \, dr \]  
(96)

\[ I_{W5j} = -\frac{1}{2} \rho C_{l\alpha} \int_{0}^{R} \frac{1}{\sqrt{1 - M^2}} V_{Rc} S_{W5j} (V_i \sin \phi - W_i \cos \phi) \, dr \]  
(97)

\[ I_{C} = -\frac{1}{2} \rho C_{l\alpha} \int_{0}^{R} \frac{1}{2} V_{R}^2 (V_i \sin \phi - W_i \cos \phi) \, dr \]  
(98)

\[ I_{S} = I_{C} \]  
(99)

The virtual work on the wing done by the rotor-blade air loading, given by equation (29), is

\[
F_i = \sum_{n=1}^{N} \int_{0}^{R} \left\{ \left( \begin{array}{c}
U_{W_i,tip} \\
W_{W_i,tip}
\end{array} \right)^t (T) \left( \begin{array}{c}
0 \\
p_{\theta n}
\end{array} \right)
\right.
\]

\[
+ \left( \begin{array}{c}
W_{W_i,tip}' \\
P_{W_i,tip}'
\end{array} \right)^t (T) \left[ \left( \begin{array}{c}
\rho \nabla \\
v_n
\end{array} \right) \times \left( \begin{array}{c}0 \\
p_{\theta n}
\end{array} \right) \right] \} \right) \, dr
\]

\[
= \sum_{j=1}^{3} K_{W1j} \dot{q}_{j} + \sum_{j=1}^{3} K_{W2j} a_{j}
\]

\[
+ \sum_{j=1}^{2} K_{B1j} \dot{q}_{j} + \sum_{j=1}^{2} K_{B2j} \dot{q}_{j}
\]

\[
+ \sum_{j=1}^{2} K_{B1j} \dot{q}_{j} + \sum_{j=1}^{2} K_{B2j} \dot{q}_{j}
\]

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Details of $K_{kkj}$, $K_{Bkj}$, $K_i$, $K_{i\alpha}$ and $K_i$ are given in appendix A.

The wing aerodynamics are more complicated because of the pitching motion. When some aerodynamics theories are referred to as quasi-steady, the reference often means the usage of wing velocity relative to air, neglecting the effect of shed vortices. When pitching motion is introduced, wing velocity relative to the air varies along the chord. Aerodynamic force becomes dependent upon the point used to represent the wing motion. It is not appropriate to take the elastic axis as a representative point, since the air forces are not directly related to blade elasticity. In steady or unsteady aerodynamics (such as lifting-surface theory or Theodorsen’s theory) the three-quarter chord line is an important point to represent the boundary condition. In this formulation the boundary condition for the angle of attack is at the three-quarter chord line, and the effect of shed vortices is disregarded. The wing aerodynamic force is given by lift $l_w$, drag $d_w$ and pitching moment around the elastic axis $m_e$

$$l_w = \frac{1}{2} \rho V^2 c_w C_{\alpha} [\theta_w + p_w - \frac{v_d + \dot{u}_w - (0.75 - a_e) c_w \dot{\theta}_w}{V + \dot{w}_w}]$$

$$m_e = \frac{1}{2} \rho V^2 c_w C_m + l(a_e - 0.25)c_w$$

where $C_m$ is the pitching-moment coefficient about the aerodynamic center (which is assumed to fall on the one-quarter chord line), and $a_e c_w$ is the distance between the leading edge and the wing elastic axis. Downwash caused by the wing itself is denoted by $v_d$ and is given using the geometrical angle of attack $\theta_w$ and the wing-aspect ratio $A_W$

$$v_d = \frac{2 - A_W + \sqrt{4 + A_W^2}}{2 + \sqrt{4 + A_W^2}} V \theta_w$$

The inclination of lift is expressed as $\phi_w = v_d/V$. When this inclination of lift is assumed unchanged to be, the aerodynamic force on the wing is given in the X and Y directions
as \((l_w \cos \phi_w - 0.5pV^2c_wC_d \sin \phi_w, -l_w \sin \phi_w + 0.5pV^2c_wC_d \cos \phi_w)\). The virtual work on the wing done by the wing aerodynamic forces is given by

\[
G_i = \int_0^L \left( \begin{array}{c}
l_w \cos \phi_w - 0.5pV^2c_wC_d \sin \phi_w \\
l_w \sin \phi_w + 0.5pV^2c_wC_d \cos \phi_w \\
m_c \\
W_{wi} \\
P_{wi}
\end{array} \right)^t \left( \begin{array}{c}
U_{wi} \\
W_{wi}\end{array} \right) dy
= G_{W1}^i + \sum_{j=1}^3 G_{W2j}^i \dot{a}_j + \sum_{j=1}^3 G_{W3j}^i \dot{a}_j
\]

(104)

\[
G_{W1}^i = \int_0^L \frac{1}{2} \rho v c_w C_{i\alpha} (V \dot{\theta}_w - v_d) \cdot [U_{wi} \cos \phi_w - W_{wi} \sin \phi_w + (a_e - 0.25)c_w P_{wi}] dy
\]

(105)

\[
G_{W2j}^i = \int_0^L \frac{1}{2} \rho c_w C_{i\alpha} V^2 P_{wj} \cdot [U_{wi} \cos \phi_w - W_{wi} \sin \phi_w + (a_e - 0.25)c_w P_{wi}] dy
\]

(106)

\[
G_{W3j}^i = \int_0^L \frac{1}{2} \rho c_w C_{i\alpha} \cdot [V U_{wj} + (0.75 - a_e) V c_w P_{wj} - v_dW_{wj}] \cdot [U_{wi} \cos \phi_w - W_{wi} \sin \phi_w + (a_e - 0.25)c_w P_{wi}] dr
\]

(107)

After the virtual work is written in terms of generalized coordinates, and when one compares the terms multiplying the constant (independent of azimuth angle \(\psi_n\)), \(\sin \psi_n\) and \(\cos \psi_n\) terms, the following equations are obtained for the blade equation of motion

\[
\ddot{q}_{i0} + \omega_i^2 q_{i0} + \sum_{j=1}^3 Q_{1ij}^i \ddot{a}_j - \sum_{j=1}^2 I_{B1j}^i \dot{q}_{j0} - \sum_{j=1}^3 I_{W1j}^i \dot{a}_j = I_{B0}^i
\]

(108)

\[
\ddot{q}_{is} + (\omega_i^2 - \Omega^2) q_{is} - 2\Omega \dot{q}_{ic} + \sum_{j=1}^3 Q_{2ij}^i \ddot{a}_j + \sum_{j=1}^3 Q_{3ij}^i \dot{a}_j
\]

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In the same way, by substituting equations (100) and (104) into equation (54), the following equation is obtained from the constant term of the wing equation of motion:

\[
\ddot{q}_{ic} + (\omega_i^2 - \Omega^2)q_{ic} + 2\Omega \dot{q}_{is} + \sum_{j=1}^{3} Q_{ij}^i \ddot{a}_j + \sum_{j=1}^{3} Q_{ij}^j \dot{a}_j
\]

\[
- \sum_{j=1}^{2} I_{B2j}^i \dot{q}_{js} - \sum_{j=1}^{2} I_{B3j}^i q_{jc} - \sum_{j=1}^{2} I_{B4j}^i \dot{q}_{jc} - \sum_{j=1}^{2} I_{B5j}^i q_{jc} - \sum_{j=1}^{3} I_{W2j}^i \ddot{a}_j - \sum_{j=1}^{3} I_{W3j}^i a_j = I_{S}^i \theta_s
\]

\[
- \sum_{j=1}^{3} I_{W4j}^i \ddot{a}_j - \sum_{j=1}^{3} I_{W5j}^i a_j = I_{C}^i \theta_c
\]

\[
\ddot{a}_i + \omega_i^2 a_i + \sum_{j=1}^{3} \left[ I_B N \Omega(N_{yi} N_{pj} - N_{pi} N_{yj}) - G_{W3j}^i - K_{W1j}^i \right] \ddot{a}_j
\]

\[- \sum_{j=1}^{3} (G_{W2j}^i + K_{W2j}^i) a_j
\]

\[+ \sum_{j=1}^{3} N \alpha_{1j}^i \ddot{q}_{j0} - \sum_{j=1}^{3} K_{B1j}^i \dot{q}_{j0} + \sum_{j=1}^{3} K_{B4j}^i q_{j0}
\]

\[+ \sum_{j=1}^{2} N \alpha_{2j}^i \ddot{q}_{j0} + \sum_{j=1}^{2} \left( \frac{N}{2} A_{4j}^i + N \Omega A_{3j}^i - K_{B2j}^i \right) \dot{q}_{j0}
\]

\[+ \sum_{j=1}^{2} \left( \frac{N}{2} A_{6j}^i + \frac{N}{2} \Omega A_{5j}^i - \frac{N}{2} \Omega^2 A_{2j}^i - K_{B5j}^i \right) q_{j0}
\]

\[+ \sum_{j=1}^{2} \frac{N}{2} A_{3j}^i \ddot{q}_{j0} + \sum_{j=1}^{2} \left( \frac{N}{2} A_{5j}^i - N \Omega A_{2j}^i - K_{B4j}^i \right) \dot{q}_{j0}
\]

\[+ \sum_{j=1}^{2} \left( \frac{N}{2} A_{7j}^i - \frac{N}{2} \Omega A_{4j}^i - \frac{N}{2} \Omega^2 A_{3j}^i - K_{B6j}^i \right) q_{jc}
\]
Final equations of motion for the generalized coordinates in equations (108) through (110) are written in the following form:

\[ E^2 \ddot{X} + E^1 \dot{X} + E^0 X = D (\theta_0 \quad \theta_s \quad \theta_c) \]  

(112)

\[ E^n = \begin{pmatrix} E_{1,1}^n & E_{1,2}^n & \cdots & E_{1,9}^n \\ E_{2,1}^n & E_{2,2}^n & \cdots & E_{2,9}^n \\ \vdots & \vdots & \ddots & \vdots \\ E_{9,1}^n & E_{9,2}^n & \cdots & E_{9,9}^n \end{pmatrix} \]  

(113)

\[ D = \begin{pmatrix} D_{1,1} & D_{1,2} & D_{1,3} \\ D_{2,1} & D_{2,2} & D_{2,3} \\ \vdots & \vdots & \vdots \\ D_{9,1} & D_{9,2} & D_{9,3} \end{pmatrix} \]  

(114)

\[ X = (q_{10}, q_{1s}, q_{1c}, q_{20}, q_{2s}, q_{2c}, a_1, a_2, a_3)^t \]  

(115)

Details of the elements \( E_{ij}^n \) and \( D_{ij} \) are given in appendix B.

DESIGN OF CONTROL SYSTEM

One approach to suppress coupled wing/rotor self-exited oscillation is active pitch control of the rotor blade (references 8-11). The tilt-rotor aircraft has other control surfaces, such as a flap. However, only the pitch control using a conventional swashplate is considered here. The control system is a simple feedback system which uses wing-deflection feedback. The possibility of flutter control is investigated using a harmonic balance method. For simplicity, air loading is calculated here using a two-dimensional strip theory for both the wing and the rotor blade. In the following analysis, the gravitational force is neglected. When blade pitch control is introduced, the natural frequencies and corresponding mode shape of the system vary as a function of time. This change is not considered and the natural frequency and mode shape are assumed to be valid for a specific flight condition.

This formulation uses generalized coordinates. What actually can be measured in flight is not a generalized coordinate but the actual deflection or its derivative, which is given by an infinite series of generalized coordinates. There are two choices in introducing a feedback control system: one uses a generalized coordinate as input for the control system, and the other uses an actual deflection or a rate of deflection. The latter is chosen here.
The wing deflection speed and the acceleration at the wingtip are used for the feedback input. Cyclic pitch control is given by

\[ \theta_s = K_{ud} \ddot{u}_{w,\text{tip}} + K_{uad} \dddot{u}_{w,\text{tip}} \\
+ K_{wad} \dot{u}_{w,\text{tip}} + K_{wad} \dddot{u}_{w,\text{tip}} \\
+ K_{psd} \ddot{p}_{w,\text{tip}} + K_{psd} \dddot{p}_{w,\text{tip}} \]

\[ \theta_c = K_{ucd} \ddot{u}_{w,\text{tip}} + K_{ucd} \dddot{u}_{w,\text{tip}} \\
+ K_{wcd} \dot{u}_{w,\text{tip}} + K_{wcd} \dddot{u}_{w,\text{tip}} \\
+ K_{pcd} \ddot{p}_{w,\text{tip}} + K_{pcd} \dddot{p}_{w,\text{tip}} \]

Using this control law, the final equation is modified to

\[ \mathbf{E}^{2c} \dddot{X} + \mathbf{E}^{1c} \dddot{X} + \mathbf{E}^{0c} \dddot{X} = \mathbf{D} \begin{pmatrix} \theta_0 & 0 & 0 \end{pmatrix}^T \]

\[ \mathbf{E}^{2c} = \mathbf{E}^2 - (K_{uad} \mathbf{B}^s + K_{ucd} \mathbf{B}^c) \mathbf{V} \\
- (K_{wad} \mathbf{B}^s + K_{wcd} \mathbf{B}^c) \mathbf{W} \\
- (K_{psd} \mathbf{B}^s + K_{pcd} \mathbf{B}^c) \mathbf{P} \]

\[ \mathbf{E}^{1c} = \mathbf{E}^1 - (K_{uad} \mathbf{B}^s + K_{ucd} \mathbf{B}^c) \mathbf{V} \\
- (K_{wad} \mathbf{B}^s + K_{wcd} \mathbf{B}^c) \mathbf{W} \\
- (K_{psd} \mathbf{B}^s + K_{pcd} \mathbf{B}^c) \mathbf{P} \]

\[ \mathbf{E}^{0c} = \mathbf{E}^0 \]

\[ \mathbf{B}^s = \begin{pmatrix} I^1_S & I^1_S & I^1_S \\
0 & 0 & 0 \\
0 & 0 & 0 \\
I^2_S & I^2_S & I^2_S \\
0 & 0 & 0 \\
0 & 0 & 0 \\
K^1_S & K^1_S & K^1_S \\
K^2_S & K^2_S & K^2_S \\
K^3_S & K^3_S & K^3_S \end{pmatrix} \]
The stability of the system, whose motion is given by equation (118), is defined by the eigenvalues of the following matrix S.

\[
\mathbf{S} = \begin{pmatrix}
I_C^1 & I_C^1 & I_C^1 \\
0 & 0 & 0 \\
0 & 0 & 0 \\
I_C^2 & I_C^2 & I_C^2 \\
0 & 0 & 0 \\
0 & 0 & 0 \\
K_C^1 & K_C^1 & K_C^1 \\
K_C^2 & K_C^2 & K_C^2 \\
K_C^3 & K_C^3 & K_C^3
\end{pmatrix}
\]  \hspace{1cm} (123)

\[
\mathbf{V} = \begin{pmatrix}
0 & 0 & 0 & 0 & 0 & 0 & V_{W1,tip} & V_{W1,tip} & V_{W1,tip} \\
0 & 0 & 0 & 0 & 0 & 0 & V_{W2,tip} & V_{W2,tip} & V_{W2,tip} \\
0 & 0 & 0 & 0 & 0 & 0 & V_{W3,tip} & V_{W3,tip} & V_{W3,tip}
\end{pmatrix}
\]  \hspace{1cm} (124)

\[
\mathbf{W} = \begin{pmatrix}
0 & 0 & 0 & 0 & 0 & 0 & W_{W1,tip} & W_{W1,tip} & W_{W1,tip} \\
0 & 0 & 0 & 0 & 0 & 0 & W_{W2,tip} & W_{W2,tip} & W_{W2,tip} \\
0 & 0 & 0 & 0 & 0 & 0 & W_{W3,tip} & W_{W3,tip} & W_{W3,tip}
\end{pmatrix}
\]  \hspace{1cm} (125)

\[
\mathbf{P} = \begin{pmatrix}
0 & 0 & 0 & 0 & 0 & 0 & P_{W1,tip} & P_{W1,tip} & P_{W1,tip} \\
0 & 0 & 0 & 0 & 0 & 0 & P_{W2,tip} & P_{W2,tip} & P_{W2,tip} \\
0 & 0 & 0 & 0 & 0 & 0 & P_{W3,tip} & P_{W3,tip} & P_{W3,tip}
\end{pmatrix}
\]  \hspace{1cm} (126)

The stability of the system, whose motion is given by equation (118), is defined by the eigenvalues of the following matrix S.

\[
\mathbf{S} = \begin{pmatrix}
-E^{2cI} & E^{4c} \\
-I_9 & -E^{2cI} \mathbf{I}_9
\end{pmatrix}
\]  \hspace{1cm} (127)

where $E^{2cI}$, $I_9$, and $0_9$ indicate the inverse matrix of $E^{2c}$, a unit matrix of order nine, and a zero square matrix of order nine, respectively.

**TIMEWISE CALCULATION**

To make a transient numerical calculation for the tilt-rotor's coupled-wing and blade deflections, and to evaluate the control law given in the previous section, a method is
given for independent timewise calculations. Determination of the control law is based on a rather simple theory. A more sophisticated aerodynamics theory and two or more natural oscillatory modes for each degree of freedom are employed in the transient analysis.

Most of the aeroelastic studies conducted previously have been using quasi-steady and strip theory. In this study the aerodynamic force is calculated on the basis of an unsteady theory. The calculation is based on Weissinger's concept through a vortex lattice method for the wing and on a local circulation method for the rotor blade (both with modifications to take the shed-vortex effect into account). Unsteady aerodynamics is a very complex problem even for a fixed wing with infinite aspect ratio. To get a precise solution requires large amounts of computer time. As for this study an efficient timewise calculation of tilt-rotor motion was desired, the following formulation aims at a rapid calculation rather than an exact solution. Arbitrary wing or rotor-blade motion is treated as a superposition of harmonic motions for a short interval to make the calculation simple. Then, for each harmonic motion, wing theory in harmonic motion is applied.

Wing lift $l_w$ and pitching moment $m_e$ consist of three components: steady components $l_s$ and $m_s$, harmonic components $l_k$ and $m_k$, and virtual-mass components $l_v$ and $m_v$:

\begin{align}
    l_w &= l_s + \sum_k l_k + l_v \tag{128} \\
    m_e &= m_s + \sum_k m_k + m_v. \tag{129}
\end{align}

As the shed vortex distribution depends upon the history of the wing motion the wing deformation and its derivatives are expressed as a series of harmonic components, using their previous values:

\begin{align}
    u_w &= u_{w0} + \sum_k \bar{u}_{wk} e^{i\omega_k t} \tag{130} \\
    \dot{u}_w &= \dot{u}_{w0} + \sum_k \bar{\dot{u}}_{wk} e^{i\omega_k t} \tag{131} \\
    w_w &= w_{w0} + \sum_k \bar{w}_{wk} e^{i\omega_k t} \tag{132} \\
    \dot{w}_w &= \dot{w}_{w0} + \sum_k \bar{\dot{w}}_{wk} e^{i\omega_k t} \tag{133} \\
    p_w &= p_{w0} + \sum_k \bar{p}_{wk} e^{i\omega_k t} \tag{134} \\
    \dot{p}_w &= \dot{p}_{w0} + \sum_k \bar{\dot{p}}_{wk} e^{i\omega_k t} \tag{135}
\end{align}
where variables with a bar represent complex numbers.

The boundary condition at the position $bc_w$ from the leading edge is

$$\dot{u} = -\dot{u}_w + \dot{p}_w (b - a_w) c_w + (V + \dot{w}_w)(\theta_w + p_w)$$  \hspace{1cm} (136)

When the deflection and its time derivative are assumed to be small, the equation is linearized as

$$\dot{u} = -\dot{u}_w + \dot{p}_w (b - a_w) c_w + V(\theta_w + p_w) + \dot{w}_w \theta_w$$  \hspace{1cm} (137)

The steady lift component is calculated through Weissinger's concept. At the three-quarter chord line, the downwash velocity is given by

$$\dot{u}_{0.75} = -\dot{u}_w + \dot{p}_w (0.75 - a_w) c_w + V(\theta_w + p_w) + \dot{w}_w \theta_w$$  \hspace{1cm} (138)

Since this value is dependent upon time, it is not directly used in the boundary condition for the air loading calculation. The steady component of the air loading is calculated using the constant term of this air velocity as the boundary condition in a vortex lattice method

$$\dot{u}_c = -\dot{u}_{w0} + \dot{p}_{w0} (0.75 - a_w) c_w + V(\theta_w + p_{w0}) + \dot{w}_{w0} \theta_w$$  \hspace{1cm} (139)

Steady pitching moment around the elastic axis $m_s$ is attributed to wing airfoil shape and is given using the pitching-moment coefficient for a steady airfoil

$$m_s = \frac{1}{2} \rho V^2 c_w C_m - l_s (0.25 - a_w) c_w$$  \hspace{1cm} (140)

The steady lift given in this way is accurate in the sense that the shed vortex effect does not exist and that the trailing vortex effect is included.

The unsteady components (harmonic components and virtual mass components) are more difficult to estimate. Even under the assumptions just given (sinusoidally oscillating air loading), there is not an easy and versatile calculation method which properly includes shed vortex and trailing vortex effects. As stated previously, this study aims at the efficient timewise calculation of tilt-rotor motion. Air loading $l_k$, caused by the $k$th harmonic motion is calculated using a two-dimensional theory. Aspect ratio and spanwise location are taken into account by multiplying $l_k$ by a factor $\eta$:

$$l_k = \eta C_{l\alpha} \rho V^2 c_w \left( \frac{w_k c_w}{2V} \right) (-\ddot{u}_{wk} + V\ddot{p}_{wk} + (0.75 - a_w)c_w \ddot{p}_{wk} + \ddot{w}_{wk} \theta_w)$$  \hspace{1cm} (141)

$$m_k = -l_k (0.25 - a_w) c_w$$  \hspace{1cm} (142)
where $C (\omega_k c_w/2V)$ is the Theodorsen function, and the factor $\eta$ is given by

$$\eta = \eta_1 \eta_2$$ (143)

$$\eta_1 = \frac{A_w}{2 + \sqrt{4 + A_w^2}}$$ (144)

$$\eta_2 = \frac{4}{\pi} \sqrt{1 - \left(\frac{y}{L}\right)^2}$$ (145)

$$A_w = \frac{(2L)^2}{S_w}$$ (146)

The variable $\eta_1$ is the correction for finite aspect ratio $A_w$, and $\eta_2$ is calculated under an assumption that this air loading is distributed elliptically along the span. The air loading caused by the virtual mass motion is also given by two-dimensional theory with the same correction:

$$l_v = \eta (l_2 + l_3)$$ (147)

$$m_v = \eta (m_2 + m_3 + m_4)$$ (148)

$$l_2 = \rho \pi \left(\frac{c_w}{2}\right)^2 (-\bar{u}_w + (0.5 - a_w)c_w \bar{p}_w)$$ (149)

$$l_3 = \rho \pi \left(\frac{c_w}{2}\right)^2 V \bar{p}_w$$ (150)

$$m_2 = -l_2 (0.25 - a_w) c_w$$ (151)

$$m_3 = -l_3 (0.75 - a_w) c_w$$ (152)

$$m_4 = -\frac{1}{8} \rho \pi \left(\frac{c_w}{2}\right)^4 \bar{p}_w.$$ (153)

Drag is calculated using a drag coefficient

$$d = \frac{1}{2} \rho V^2 c_w C_d$$ (154)

These equations give the lift, drag, and pitching moment, but not the directions of lift and drag. In the case of lifting-surface theory, the most accurate expression of lift and induced drag is the sum of integration of lift along the chord line and leading-edge suction. Because of the simplification, where the chordwise distribution of bound vortex is not taken into account, this expression is not available in the present method. The most
suitable form for this simple case is that the lift is inclined backward by an amount, equal to the angle between the free flow and the lifting line (the same expression as that in simple lifting-line theory). It is sometimes discussed in the term of induced drag, and it is known that the estimation of induced drag is difficult even in a complete lifting surface theory. The lift and drag values are converted into air-load components in $x$ and $z$ direction ($l_x$ and $l_z$)

$$l_x = l \cos \phi_w - d \sin \phi_w$$

$$l_z = -l \sin \phi_w - d \cos \phi_w$$

(155)  

(156)

The calculation of lift-inclination angle $\phi_w$ is given in appendix C with the formulation of a vortex lattice method.

The blade air loading calculation is similar to that of the wing, except that it is based on the local circulation method instead of a vortex lattice method and that it lacks the torsional degree of freedom. The details of the local circulation method are described in references 2 and 3. Here the method used in the unsteady modification is stated. Among two dimensional theories of harmonic air loading, Theodorsen's and Sears' theories are widely used. Blade deflection in two directions, in plane and out of plane, is treated just like the wing case. Apart from the deflection, the blade encounters the induced velocity field made by the returning vortex; hence the local circulation method is better explained by the Sears' theory. However, as stated previously, because this model does not have a pitching degree of freedom, and because the Sears function can be approximated by the Theodorsen function when reduced frequency is small, this remaining induced velocity is included in the boundary condition on the blade element. Harmonic air loading is calculated using Theodorsen's theory in this study (reference 12). The air loading caused by virtual-mass effects is added afterward, because the relation of induced velocity and circulation is given by the quasi-steady assumption; thus, it does not contain virtual mass or acceleration terms at all. In this calculation, variation of pylon direction is assumed to be small, so rotor-blade air loading is calculated using the theory for axial flight.

The components of blade lift are steady lift, harmonic lift, and virtual mass lift

$$l_b = l_s + \sum_k l_k + l_v$$

(157)

The blade motion relative to the air is composed of cruising velocity, rotating velocity, induced velocity, and blade- and wing-deformation velocity (figure 6). These velocities give the geometrical angle of attack at the elastic axis as

$$\alpha_G = \theta - \phi - \frac{\dot{\upsilon}_p + u_s \cos \phi}{U}$$

(158)
\[ \phi = \tan^{-1} \frac{V}{r\Omega} \]  
\[ U = \sqrt{(r\Omega)^2 + V^2} \]  
When \( \dot{\nu}_p \) is written in terms of in-plane and out-of-plane deformations equations (158) becomes
\[ \alpha_G = \theta - \phi - \frac{u_t}{U} \]  
\[ u_t = (u_s - \dot{w} - \dot{w}_w - r\dot{\nu}_y \sin \psi_n + r\dot{\nu}_p \cos \psi_n) \cos \phi 
+ [\dot{v} + r\dot{\nu}_r + (-\dot{w}_w - h\dot{\nu}_p + V\nu_p) \sin \psi_n + (-h\dot{\nu}_y + V\nu_y) \cos \psi_n] \sin \phi \]  
This geometrical angle of attack is written in Fourier series, based upon its past values
\[ \alpha_G = \alpha_{G0} + \sum_k \bar{\alpha}_{Gk} e^{i\omega_k t}. \]  
For the steady component \( \alpha_{G0} \), the conventional local-circulation method with the Prandtl-Glauert rule is applied, and the following equation is satisfied at each control point:
\[ l_s = \frac{1}{2\sqrt{1 - M^2}} \rho U^2 C_{l_a} \left( \alpha_{G0} - \frac{1}{U} \sum \Delta v \right) 
= \rho U \sum \Delta \gamma \]  
where \( \Delta v \) and \( \Delta \gamma \) are the induced velocity perpendicular to local airflow and the circulation of the hypothetical wing, respectively.

The sinusoidal component of the air loading is given by two-dimensional theory, without considering the effect of compressibility, but with corrections, similar to the wing-air loading case, for aspect ratio and spanwise location:
\[ \bar{I}_k = \frac{\eta}{2} \rho U^2 C_{l_a} C \left( \frac{\omega_k c w}{2V} \right) \bar{\alpha}_{Gk} e^{i\omega_k t} \]  
\[ \eta = \eta_1 \eta_2 \]  
\[ \eta_1 = \frac{A}{2 + \sqrt{4 + A^2}} \]  
\[ \eta_2 = \frac{4}{\pi} \sqrt{1 - \xi^2} \]  
\[ A = \frac{R - R_c}{c} \]  
\[ \xi = \frac{2r - (R + R_c)}{R - R_c} \]
Theodorsen's theory treats both the pitching motion and the heaving motion of a wing. The relative velocity of the blade to the air varies along the chordwise position because of pitching motion. Since the rotor blade model does not have a pitching degree of freedom, virtual-mass terms are assumed to be related to the acceleration of the blade elastic axis neglecting blade pitch motion.

\[ l_v = \rho \pi \left( \frac{c}{2} \right)^2 (\ddot{w} \cos \phi - \ddot{v} \sin \phi) \]  

This equation concludes the air loading calculation. The virtual work required to complete the equations of motion can now be calculated by equations (27) through (29). Timewise integration is carried out using a Runge-Kutta-Gill method. By conducting timewise calculations both with and without the feedback system, the effectiveness of the feedback system can be evaluated.

NUMERICAL RESULTS

The methods of control law determination and the timewise calculation just given are applied in this section. The specifications of the model aircraft used in numerical calculations are given in table 1 and in figures 7 and 8. Blade pitch angle is chosen to give the thrust of 7,000 N at the design advance ratio \( V/\Omega R = 0.7 \) and such that the thrust changes proportionally to the square of cruising speed. Trim thrust and pitch angle determined in this way are given in figure 9.

Figure 10 presents the change of the real part of the eigenvalues as a function of aircraft cruise velocity. This figure shows that this model remains stable for advance ratios up to 1.8.

To investigate the feedback-control efficiency, the stiffness of the considered tilt-rotor model is reduced. Because flutter peculiar to the tilt-rotor aircraft is the major concern in this report, the wing stiffness was reduced to one-eighth of the baseline model for all the degrees of freedom in the wing, while the rotor-blade stiffness is assumed to be unchanged. Natural frequencies and mode shapes employed in the eigenvalue analysis are given in Figures 11 through 15 for the new rotor/wing model. These modes are attributed to blade-in-plane, blade out-of-plane, wing-upward, wing-forward, and wing-torsional degrees of freedom, respectively. Figure 16 gives the real part of the eigenvalue as a function of advance ratio \( V/\Omega R \) for the reduced stiffness model. An eigenvalue will become unstable when the advance ratio exceeds about 1.5.
For further numerical results, an almost critical case in which the advanced ratio is equal to 1.5 is chosen and the design of a control law is investigated. In the previous section, input to the feedback system as a function of wingtip upward and forward bending and torsion was described. Using all of these parameters is not necessary in the present case. When selecting one or more of the parameters, the eigenvector and mode shape corresponding to the unstable mode is of greatest importance. The two eigenvectors which have the greatest real part at the advance ratio of 1.5 are given in figure 17. Among the elements of the eigenvector, only the elements which have absolute values of more than 5% of the largest element value are shown in this figure. The critical element is attributed to the wing's second lowest natural frequency and the other element to the the wing's third lowest frequency. Wing second and third modes correspond, in turn, to forward deflection and torsion, respectively, as seen in Figures 14 and 15. To stabilize or to augment the stability of these modes, feedback of dominant deformations are expected to be effective. As the simplest case, paying attention to the least damped mode only, velocity of the wing forward bending $\dot{w}_w$ is chosen as an input for the feedback system, which means all the feedback gains except for $K_{wds}$ and $K_{wcd}$ are equal to zero.

Figure 18 shows the effect of feedback on the real part of the systems eigenvalues. Only those eigenvalues that have a real part greater than $-1.0$ are shown among the eigenvalues, since highly damped modes need not be considered. The imaginary parts (frequency) of the eigenvalues change at most 2% when the feedback gain changes from 0.0 to $-0.05$. The eigenvalues without feedback are given along each line. The dominant, generalized coordinate corresponding to each eigenvector is also given along each line in parentheses. All the modes are simultaneously stabilized when feeding back the cosine term of cyclic pitch. Some modes are stabilized yet others are destabilized in case of feedback with the sine term of cyclic pitch. Feedback of the sine term of cyclic pitch works about as well as that of the cosine term as far as the critical mode ($a_2$, wing forward bending) is concerned. However, sine feedback is unfavorable, in that the two modes which have the largest eigenvalue real part cannot be stabilized simultaneously. From figure 18, feedback of the cosine term of cyclic pitch is expected to work better than the sine term of cyclic pitch does. Consequently the feedback gain to the cosine term $K_{wcd}$ is set to $-0.03$ for evaluation of this control system.

Timewise calculation are made to see whether the feedback system just discussed is effective. Calculations are carried out without introducing rotor-blade or wing elasticity for the first five revolutions of the rotor rotation to get the air loading in an almost equilibrium state. The full equations of motion are employed after five revolutions; thus, the initial conditions for these equations are the deviations from the deflected equilibrium state. Figures 19 through 24 show the results for all the degrees of freedom both without and with feedback control. The abscissa is the number of rotor revolutions after elasticity
is introduced. The results for 100 revolutions, or approximately 15 sec, are given. With no feedback control, the wing-upward deflection appears to be converging (figure 19) because the lowest damped modes are dominated by generalized coordinates $a_2$ and $a_3$ (wing-forward deflection and wing torsion). Wing-forward deflection (figure 20) and wing torsion (pylon-pitch angle; figure 21) show oscillation with almost constant amplitude with no feedback. Blade deformations do not appear to converge without control in spite of the fact that the lowest damped modes are not characterized by blade deflection.

Once feedback control is introduced ($\dot{w}_w$ feedback with $k_{wd} = -0.03$) wing-forward and wing-torsional deflections are stabilized (Figs. 20 and 21). Stabilization is expected because these deflections are characterized by the two lowest damped eigenvalues and the real parts of each value are significantly reduced (stabilized) with the introduction of feedback control. The wing-upward deflection does not show an improvement because the eigenvalue, characterizing this motion ($\omega = 6.0 \text{ rad/sec}$) is not affected by the introduction of feedback to the cosine term. The amplitude of blade-out-of-plane deflection becomes smaller since the eigenvalue for that mode ($\omega = 62.3 \text{ rad/sec}$) is changed by the control. The cyclic-pitch angle (figure 24) is sufficiently small so that there is additional control capability for stabilizing less stable flight conditions or for stabilizing oscillations of greater amplitude. From this time-history analysis, this control system has been proved to work as it should, and it would be expected to work as well under less stable operating conditions.

CONCLUDING REMARKS

Tilt rotor flutter control was investigated. The determination of the control law was based on a harmonic-balance method using a simple aerodynamic theory. It was confirmed that the stability of tilt-rotor motion is improved by using closed-loop, cyclic-pitch control. The validity of the control law determined in this way was demonstrated by a timewise calculation in which more sophisticated aerodynamics were considered. The selection of the feedback gain and the placement of the sensor were not based on any specific criterion. Establishing the optimal gain is left to future study.
APPENDIX A

Expressions for $K_{Wn_j}$, $K_{Bn_j}$, $K_{S}$, $K_{C}$, and $K_{0}$

Rotor-blade loading $p_{\theta n}$ and $p_{zn}$ are expressed in harmonic terms.

\[ p_{\theta n} = p_{\theta 0} + p_{\theta s} \sin \psi_n + p_{\theta c} \cos \psi_n \]
\[ p_{zn} = p_{zo} + p_{zs} \sin \psi_n + p_{zc} \cos \psi_n \]

Moreover the variables $p_{\theta 0}$, $p_{\theta s}$, $p_{\theta c}$, $p_{zo}$, $p_{zs}$, and $p_{zc}$ are subdivided into pitch-angle and generalized-coordinate terms:

\[
p_{\theta n} = p_{\theta 00} + \sum_{j=1}^{2} p_{\theta 0d_j} \dot{\theta}_j + \sum_{j=1}^{3} p_{\theta 0a_j} \dot{a}_j \\
+ (p_{\theta s}\dot{\theta}_s + \sum_{j=1}^{2} p_{\theta s\dot{q}_j} \dot{q}_j + \sum_{j=1}^{2} p_{\theta sq_j} q_{jc}) \\
+ \sum_{j=1}^{3} p_{\theta s\dot{a}_j} \dot{a}_j + \sum_{j=1}^{3} p_{\theta sa_j} a_j) \sin \psi_n \\
+ (p_{\theta c}\dot{\theta}_c + \sum_{j=1}^{2} p_{\theta c\dot{q}_j} \dot{q}_j + \sum_{j=1}^{2} p_{\theta cq_j} q_{jc}) \\
+ \sum_{j=1}^{3} p_{\theta ca_j} \dot{a}_j + \sum_{j=1}^{3} p_{\theta ca_j} a_j) \cos \psi_n
\]

\[
p_{zn} = p_{zo0} + \sum_{j=1}^{2} p_{zo\dot{q}_j} \dot{q}_j + \sum_{j=1}^{3} p_{zo\dot{a}_j} \dot{a}_j \\
+ (p_{zo}s\dot{\theta}_s + \sum_{j=1}^{2} p_{zs\dot{q}_j} \dot{q}_j + \sum_{j=1}^{2} p_{zsq_j} q_{jc}) \\
+ \sum_{j=1}^{3} p_{zs\dot{a}_j} \dot{a}_j + \sum_{j=1}^{3} p_{zs\dot{a}_j} a_j) \sin \psi_n \\
+ (p_{zo}c\dot{\theta}_c + \sum_{j=1}^{2} p_{zc\dot{q}_j} \dot{q}_j + \sum_{j=1}^{2} p_{zcq_j} q_{jc}) \\
+ \sum_{j=1}^{3} p_{zo\dot{a}_j} \dot{a}_j + \sum_{j=1}^{3} p_{zo\dot{a}_j} a_j) \cos \psi_n
\]
\[
\begin{align*}
p_{\theta 00} &= -\frac{1}{2\sqrt{1 - M^2}} \rho V_R^2 c C_{i\alpha} (\theta_0 - \phi) \sin \phi - \frac{1}{2} \rho V_R^2 c c_d \cos \phi \\
p_{\theta 0\dot{q}, 0} &= -\frac{1}{2\sqrt{1 - M^2}} \rho V_R^2 c C_{i\alpha} \frac{S_{B1\dot{q}}}{V_R} \sin \phi \\
p_{\theta 0\dot{a}, j} &= -\frac{1}{2\sqrt{1 - M^2}} \rho V_R^2 c C_{i\alpha} \frac{S_{W1\dot{a}}}{V_R} \sin \phi \\
p_{\theta \dot{s}, s} &= -\frac{1}{2\sqrt{1 - M^2}} \rho V_R^2 c C_{i\alpha} \sin \phi \\
p_{\theta \dot{s}q, j} &= -\frac{1}{2\sqrt{1 - M^2}} \rho V_R^2 c C_{i\alpha} \frac{S_{B2\dot{s}}}{V_R} \sin \phi \\
p_{\theta \dot{s}q, r} &= -\frac{1}{2\sqrt{1 - M^2}} \rho V_R^2 c C_{i\alpha} \frac{S_{B3\dot{s}}}{V_R} \sin \phi \\
p_{\theta \dot{s}a, j} &= -\frac{1}{2\sqrt{1 - M^2}} \rho V_R^2 c C_{i\alpha} \frac{S_{W2\dot{s}}}{V_R} \sin \phi \\
p_{\theta \dot{s}a, j} &= -\frac{1}{2\sqrt{1 - M^2}} \rho V_R^2 c C_{i\alpha} \frac{S_{W3\dot{s}}}{V_R} \sin \phi \\
p_{\theta \dot{e}, c} &= -\frac{1}{2\sqrt{1 - M^2}} \rho V_R^2 c C_{i\alpha} \sin \phi \\
p_{\theta \dot{e}q, j} &= -\frac{1}{2\sqrt{1 - M^2}} \rho V_R^2 c C_{i\alpha} \frac{S_{B4\dot{e}}}{V_R} \sin \phi \\
p_{\theta \dot{e}q, r} &= -\frac{1}{2\sqrt{1 - M^2}} \rho V_R^2 c C_{i\alpha} \frac{S_{B5\dot{e}}}{V_R} \sin \phi \\
p_{\theta \dot{e}a, j} &= -\frac{1}{2\sqrt{1 - M^2}} \rho V_R^2 c C_{i\alpha} \frac{S_{W4\dot{e}}}{V_R} \sin \phi \\
p_{\theta \dot{e}a, j} &= -\frac{1}{2\sqrt{1 - M^2}} \rho V_R^2 c C_{i\alpha} \frac{S_{W5\dot{e}}}{V_R} \sin \phi \\
p_{\phi 00} &= \frac{1}{2\sqrt{1 - M^2}} \rho V_R^2 c C_{i\alpha} (\theta_0 - \phi) \cos \phi - \frac{1}{2} \rho V_R^2 c c_d \sin \phi \\
p_{\phi 0\dot{q}, \mu} &= \frac{1}{2\sqrt{1 - M^2}} \rho V_R^2 c C_{i\alpha} \frac{S_{B1\dot{q}}}{V_R} \cos \phi \\
p_{\phi 0\dot{a}, j} &= \frac{1}{2\sqrt{1 - M^2}} \rho V_R^2 c C_{i\alpha} \frac{S_{W1\dot{a}}}{V_R} \cos \phi \\
p_{\phi \dot{s}, s} &= \frac{1}{2\sqrt{1 - M^2}} \rho V_R^2 c C_{i\alpha} \cos \phi 
\end{align*}
\]
Substituting these relations into equation (29) yields

\[ f_i = \sum_{j=1}^{3} (K_{W1j}^i \dot{a}_j + K_{W2j}^i a_j) \]

\[ + \sum_{j=1}^{2} \left( K_{B1j}^i \dot{q}_{j0} + K_{B2j}^i q_{j0} + K_{B3j}^i \dot{q}_{js} + K_{B4j}^i q_{js} + K_{B5j}^i \dot{q}_{jc} + K_{B6j}^i q_{jc} \right) + K_S^i \theta_s + K_C^i \theta_c + K_0^i \]

\[ K_{W1j}^i = \frac{N}{2} \int_0^R \left[ -X_1 p_{\theta s a_j} + 2Z_1 p_{z a_j} \right. \]

\[ + N_{y1}(-h p_{\theta c a_j} + r p_{z s a_j}) \]

\[ - N_{y1}(-h p_{\theta s a_j} + r p_{z c a_j}) \]

\[ + 2N_{r1} r p_{\theta 0 a_j} \] \, dr

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\[ K_{W2j}^i = \frac{N}{2} \int_0^R \left[ X_i(-p_{\theta sa_j} + 2N_{y_j}p_{z00}) ight. \\
+ N_{y_i}(-hp_{\theta ca_j} + rp_{zsa_j} + 2rN_{p_j}p_{\theta 00}) \\
+ N_{p_i}(-2N_{y_j}p_{\theta 00} - hp_{\theta sa_j} - hp_{zca_j}) \right] dr \]

\[ K_{B1j}^i = \frac{N}{2} \int_0^R \left( 2Z_i p_{z0q_{j,0}} + 2N_{r_i}r p_{\theta 0q_{j,0}} \right) dr \]

\[ K_{B2j}^i = \frac{N}{2} \int_0^R \left( -X_i p_{\theta sq_{j,s}} + N_{y_i} r p_{zsq_{j,s}} - N_{p_i} h p_{\theta sq_{j,s}} \right) dr \]

\[ K_{B3j}^i = \frac{N}{2} \int_0^R \left( -N_{y_i} h p_{\theta sq_{j,s}} - N_{p_i} r p_{zsq_{j,s}} \right) dr \]

\[ K_{B4j}^i = 0 \]

\[ K_{B5j}^i = \frac{N}{2} \int_0^R \left[ -X_i p_{\theta sq_{j,c}} + N_{y_i} (-hp_{\theta cq_{j,c}} + rp_{zsq_{j,c}}) \right. \\
+ N_{p_i} (V_j p_{z00} - W_j p_{\theta 00} - rp_{zsq_{j,c}} - hp_{\theta sq_{j,c}}) \left. \right] dr \]

\[ K_{B6j}^i = \frac{N}{2} \int_0^R \left[ -X_i p_{\theta sq_{j,c}} \right. \\
+ N_{y_i} (V_j p_{z00} - W_j p_{\theta 00} + rp_{zsq_{j,c}} - hp_{\theta cq_{j,c}}) \left. \right. \\
- N_{p_i} (hp_{\theta sq_{j,c}} + rp_{zsq_{j,c}}) \right] dr \]

\[ K_S^i = \frac{N}{2} \int_0^R \left( -X_i p_{\theta ss} + N_{y_i} r p_{zss} - N_{p_i} h p_{\theta ss} \right) dr \]

\[ K_C^i = \frac{N}{2} \int_0^R \left( -N_{y_i} h p_{\theta cc} - N_{p_i} r p_{zcc} \right) dr \]

\[ K_0^i = N \int_0^R (Z_i p_{z00} + N_{r_i} r p_{\theta 00}) dr \]
APPENDIX B

Matrix Elements of Final Equations

The following matrix elements for equations (112) through (114), with an exception for $E_{i,i}^2$, are given for: $i = 1, 2; j = 1, 2; k = 1, 3; \text{and } l = 1, 3$:

$$E_{31,3i-2}^2 = 1 \quad (i = 1 \sim 9)$$
$$E_{31-2,k+6}^2 = Q_{1k}^i$$
$$E_{31-1,k+6}^2 = Q_{2k}^i$$
$$E_{31,k+6}^2 = Q_{4k}^i$$
$$E_{k+6,3i-2}^2 = NA_{1k}^i$$
$$E_{k+6,3i-1}^2 = \frac{N}{2} A_{3i}^k$$
$$E_{k+6,3i}^2 = \frac{N}{2} A_{3i}^k$$

$$E_{31-2,3j-2}^1 = -I_{B1j}^i$$
$$E_{31-2,k+6}^1 = -I_{W1k}^i$$
$$E_{31-1,3j-1}^1 = -I_{B2j}^i$$
$$E_{31-1,3i}^1 = -2\Omega$$
$$E_{31-1,k+6}^1 = Q_{5k}^i - I_{W2k}^i$$
$$E_{31,3i-1}^1 = 2\Omega$$
$$E_{31,3i}^1 = -I_{B4j}^i$$
$$E_{31,k+6}^1 = Q_{5k}^i - I_{W4k}^i$$
$$E_{1+6,3i-2}^1 = -K_{B1i}^k$$
$$E_{k+6,3i-1}^1 = \frac{N}{2} A_{3i}^k + N\Omega A_{3i}^k - K_{B2i}^k$$
$$E_{k+6,3i}^1 = \frac{N}{2} A_{3i}^k - N\Omega A_{3i}^k - K_{B4i}^k$$
$$E_{k+6,3i+6} = I_B N\Omega(N_{yk}N_{pl} - N_{pl}N_{yk}) - G_{W3l}^i - K_{W1l}^k$$

$E_{i,j}^n$ (if not given above) = 0
$D_{i,j}$ (if not given above) = 0
APPENDIX C

Vortex Lattice Method for Wing Aerodynamics

Figure C1 shows the tilt-rotor-wing coordinate system for the bound-vortex calculation. Only the right half wing is shown because the whole wing is assumed to be symmetric. The semispan wing is equally divided into $k$ segments and the vortex $\delta \gamma_i$ is trailed from the outer edge of the $i$th segment. Downwash at the $i$th control point $v_i$ is given as

$$v_i = \sum_{j=1}^k B_{ij} \delta \gamma_j$$

where $B_{ij} \delta \gamma_j$ is the induced velocity on $i$th control point by the trailing vortex from the outer edge of $j$th segment. The wing’s mirror image and corresponding bound vortex connects the upper two trailing vortices. The coefficients $B_{ij}$ are given by

$$B_{ij} = \frac{1}{4\pi (y_{bj} - y_{ci})} \left[ 2 + \frac{x_c + y_{bj} - y_{ci}}{\sqrt{(y_{ci} - y_{bj})^2 + x_c^2}} + \frac{x_c + y_{bj} + y_{ci}}{\sqrt{(y_{ci} + y_{bj})^2 + x_c^2}} \right]$$

where $x_c$ is the distance between the bound vortex and the control point and is equal to the semi-chord, $y_{bj}$ is the distance of $j$th segment outer edge from the center of whole wing, and $y_{ci}$ is the distance of the $i$th control point from the center of the whole wing. Writing the boundary condition (i.e., downwash on the $i$th control point) as $b_i$, the equation which connects the intensity of each trailing vortex and the boundary conditions is given by

$$B \begin{pmatrix} \delta \gamma_1 \\ \delta \gamma_2 \\ \vdots \\ \delta \gamma_k \end{pmatrix} = \begin{pmatrix} b_1 \\ b_2 \\ \vdots \\ b_k \end{pmatrix}$$

$$B = \begin{pmatrix} B_{11} & B_{12} & \cdots & B_{1k} \\ B_{21} & B_{22} & \cdots & B_{2k} \\ \vdots & \vdots & \ddots & \vdots \\ B_{k1} & B_{k2} & \cdots & B_{kk} \end{pmatrix}$$

The intensity of the trailing vortex is given by

$$\begin{pmatrix} \delta \gamma_1 \\ \delta \gamma_2 \\ \vdots \\ \delta \gamma_k \end{pmatrix} = (B)^{-1} \begin{pmatrix} b_1 \\ b_2 \\ \vdots \\ b_k \end{pmatrix}$$

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The intensity of the bound vortex in the $j$th section $\gamma_{b_j}$ is

$$\gamma_{b_k} = \delta \gamma_k$$

$$\gamma_{b_{k-i}} = \gamma_k + \delta \gamma_{k-i} \quad i = 1, 2, \ldots k - 1$$

Finally, the lift distribution in $i$th section $l_i$ is

$$l_i = \rho V \gamma_{b_i}$$

To determine the direction of the lift, the induced velocity on the bound vortex $v_{bi}$ needs to be determined. The lift is caused by the trailing vortex only, since a straight and unswept wing is considered. The equation for the induced velocity is

$$v_{bi} = \frac{1}{4\pi} \sum_{j=1}^{k} \left( \frac{\delta \gamma_j}{y_{bj} - y_{ci}} + \frac{\delta \gamma_j}{y_{bj} + y_{ci}} \right)$$

The backward inclination of lift $\phi_w$ is given by

$$\phi_w = \frac{v_{bi}}{V} = \frac{1}{4\pi V} \sum_{j=1}^{k} \left( \frac{\delta \gamma_j}{y_{bj} - y_{ci}} + \frac{\delta \gamma_j}{y_{bj} + y_{ci}} \right)$$

---

Figure C1.- Vortex lattice geometry for wing aerodynamics.
REFERENCES

TABLE 1.- SPECIFICATION OF BASELINE TILT ROTOR MODEL.

<table>
<thead>
<tr>
<th>Rotor</th>
<th>Value</th>
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<tbody>
<tr>
<td>number of blades, $N$</td>
<td>3</td>
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<tr>
<td>radius, $R$</td>
<td>3.81 m</td>
</tr>
<tr>
<td>cutout, $R_c$</td>
<td>0.15R</td>
</tr>
<tr>
<td>chord, $c$</td>
<td>0.356 m</td>
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<tr>
<td>solidity, $\sigma$</td>
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<td>lift slope, $C_{l0}$</td>
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<tr>
<td>drag coefficient, $C_d$</td>
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<tr>
<td>rotational angular velocity, $\Omega$</td>
<td>40.4 rad/sed (386.0 rpm)</td>
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<tr>
<td>blade-flapping inertia, $I_B$</td>
<td>176.1 kg \cdot m$^2$</td>
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<td>blade mass, $M_B$</td>
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</table>

<table>
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<th>Wing</th>
<th>Value</th>
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</thead>
<tbody>
<tr>
<td>semispan, $L$</td>
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<td>chord, $c_w$</td>
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<tr>
<td>geometric angle of attack, $\alpha_W$</td>
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<tr>
<td>lift slope, $C_{l0}$</td>
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<tr>
<td>drag coefficient, $c_d$</td>
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<tr>
<td>pitching moment coefficient, $C_m$</td>
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<td>position of elastic axis from leading edge, $a_e$</td>
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<tr>
<td>position of center of gravity from leading edge, $c_g$</td>
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</tr>
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<tr>
<th>Pylon</th>
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<tr>
<td>mass, $M_P$</td>
<td>644.0 kg</td>
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<tr>
<td>mast height, $h$</td>
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<td>pitching mass moment of inertia, $I_{pp}$</td>
<td>257.6 kg \cdot m$^2$</td>
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<td>rolling mass moment of inertia, $I_{pr}$</td>
<td>57.5 kg \cdot m$^2$</td>
</tr>
<tr>
<td>yawing mass moment of inertia, $I_{py}$</td>
<td>223.4 kg \cdot m$^2$</td>
</tr>
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</table>
Figure 1.- Tilt-rotor model.

Figure 2.- Coordinate system and pylon rotations.
Figure 3.- Wing and blade deformation.

a.- Wing.

b.- Blade.
Figure 4. - Rotor azimuth angle and pylon rotation.
Figure 5.- Air-loading geometry on wing and blade.

Figure 5.- Air-loading geometry on wing and blade.
Figure 6.- Airflow and blade motion geometry.

Figure 7.- Blade structural properties.

a. - Mass and twist.

b. - Stiffness.
a.- Mass and moment of inertia.

b.- Stiffness.

Figure 8.- Baseline-wing structural properties.
Figure 9.- Pitch angle and thrust necessary for cruise flight.

Figure 10.- Real part of eigenvalues for baseline-wing configuration.
Figure 11.- First blade mode shape; $\omega_1 = 37.6$ rad/sec.

Figure 12.- Second blade mode shape; $\omega_2 = 60.6$ rad/sec.
Figure 13.- First wing mode shape; $\omega_{w1} = 6.19 \text{ rad/sec}$.

Figure 14.- Second wing mode shape; $\omega_{w2} = 11.2 \text{ rad/sec}$.
Figure 15.- Third wing mode shape; $\omega_{w3} = 18.1$ rad/sec.

Figure 16.- Real part of eigenvalues for one eighth stiffness wing configuration.
\[ \lambda = -0.00477 + 10.4i \]

\[ \lambda = -0.0892 + 17.7i \]

Figure 17.- Eigenvectors without feedback for lowest damped modes; \( V/\Omega R = 1.5 \).

Figure 18.- Real part of eigenvalues with wing feedback; \( V/\Omega R = 1.5 \).

a.- Cosine cyclic feedback; \( K_{wcd} \).
b.- Sine cyclic feedback; $K_{wd}$. 

Figure 18.- Concluded.
a.- No feedback control.

b.- Feedback control; $K_{wcd} = -0.03$.

Figure 19.- Wingtip upward deflection.
a.- No feedback control.

b.- Feedback control; $K_{wcd} = -0.03$.

Figure 20.- Wingtip forward deflection.
Figure 21.- Pylon-pitch angle.

a.- No feedback control.

b.- Feedback control; $K_{wcd} = -0.03$. 
a.- No feedback control.

b.- Feedback control; $K_{wcd} = -0.03$.

Figure 22.- First-blade inplane deflection.
a.- No feedback control.

b.- Feedback control; $K_{wcd} = -0.03$.

Figure 23.- First-blade out-of-plane deflection.
Figure 24.- Cosine cyclic-control pitch used for feedback; $K_{wcd} = -0.03$. 
TILT-ROTOR FLUTTER CONTROL IN CRUISE FLIGHT

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Tilt-rotor flutter control under cruising operation is analyzed. The rotor model consists of a straight fixed wing, a pylon attached to the wingtip, and a three-bladed rotor. The wing is cantilevered to the fuselage and is allowed to bend forward and upward. It also has a torsional degree of freedom about the elastic axis. Each rotor blade has two bending degrees of freedom. Feedback of wingtip velocity and acceleration to cyclic pitch is investigated for flutter control, using strip theory and linearized equations of motion. To determine the feedback gain, an eigenvalue analysis is performed. A second, independent, time-wise calculation is conducted to evaluate the control law while employing more sophisticated aerodynamics. The effectiveness of flutter control by cyclic pitch change was confirmed.

Key Words (Suggested by Author(s))
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Aeroelastic stability
Active control
Flutter

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