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HELICOPTER STALL ALLEVIATION USING INDIVIDUAL-BLADE CONTROL

AND

HELICOPTER ATTITUDE STABILIZATION USING INDIVIDUAL-BLADE-CONTROL

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# HELICOPTER STALL ALLEVIATION USING INDIVIDUAL-BLADE CONTROL

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## Abstract

A new, advanced type of active control for helicopters and its application to the solution of rotor aerodynamic and aeroelastic problems is described. Each blade is individually controlled in the rotating frame over a wide range of frequencies up to the sixth harmonic of rotor speed.

The paper describes the design of a system controlling retreating blade stall, and the testing of the system on a model rotor in the wind tunnel. The control inputs considered are higher harmonic blade pitch changes at 2P and 3P, of amplitude and phase such that rotor loading is increased in the fore and aft portions of the rotor disk while rotor loading is reduced on the lateral portions. In this manner retreating blade stall may be alleviated, with corresponding reduction in rotor power requirements and vibration.

## Introduction

A truly advanced helicopter rotor must operate in a severe aerodynamic environment with high reliability and low maintenance requirements. This environment includes:

(1) atmospheric turbulence (leading to impaired flying qualities, particularly in the case of hingeless rotor helicopters).

(2) retreating blade stall (leading to large torsional loads in blade structure and control system).

(3) blade-vortex interaction in transitional and nap-of-the-earth flight (leading to unacceptable higher harmonic blade bending stresses and helicopter vibration).

(4) blade-fuselage interference (leading to unacceptable higher harmonic blade bending stresses and helicopter vibration).

(5) blade instabilities due to flap-lag coupling and high advance ratio.

The application of feedback techniques make it possible to alleviate the effects described in items (1) to (5) above, while improving helicopter vibration and handling characteristics to meet desired standards. The concept of Individual-Blade-Control (IBC) embodies the control of broadband electrohydraulic actuators attached to each blade, using signals from sensors mounted on the blades to supply appropriate control commands to the actuators[1-7]. Note that IBC in involves not just control each blade independently, but also a feedback loop for each blade in the rotating frame. In this manner it becomes possible to reduce the severe effects of atmospheric turbulence, retreating blade stall, blade-vortex interaction, blade-fuselage interference, and blade instabilities, while providing improved flying qualities and automatic blade tracking.

It is evident that the IBC system will be most effective if it is comprised of several subsystems, each controlling a specific mode, e.g., the blade flapping mode, the first blade lag mode, the first blade flatwise bending mode, and the first blade torsion mode. Each sub-system operates in its appropriate frequency band.

The configuration used in this investigation employs an individual actuator to control each blade. These actuators rotate with the blades and, therefore, a conventional swash plate is not required. However, the same degree of individual blade control can be achieved by placing the actuators in the non-rotating system and controlling the blades through a conventional swash plate if the number of control degrees-of-freedom equals the number of blades. For more than three blades, the use of extensible blade pitch control rods in the form of hydraulic actuators is a possibility. Note that all IBC functions not involving differential collective pitch can be obtained on a four-bladed rotor using a conventional swash plate.

The present paper is primarily concerned with the application of the Individual-Blade-Control concept to rotor stall alleviation. Other applications are described in Refs. 2-5 and listed in Figure 1.

## Technical Discussion

A helicopter rotor blade operates in a three-dimensional, unsteady, rotating environment; numerous studies have shown that rotor airfoil lift and moment characteristics differ

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considerably from the results of conventional steady two-dimensional airfoil tests. Harris et al. empirically superimposed the airfoil characteristics of a yawed wing and those of an airfoil oscillating in pitch (Ref. 8): Figure 2 shows the improved rotor performance and test correlation obtained, in comparison with the prediction of 2D steady theory.

An extension of this line of thought suggests that if rotor loading is increased in the fore and aft portions of the rotor disk and reduced in the lateral portions, the loaded retreating blades will be operating at higher angles of yaw and higher pitch reduced-frequencies than before, with corresponding benefits in rotor stall alleviation and performance. Such a change in rotor loading can be obtained with the blade pitch time history shown in Figure 3. Though a completely arbitrary pitch schedule is possible with IBC, for ease of description a simple superposition of 1P, 2P, and 3P pitch is employed.

The present paper considers only open-loop implementation of this pitch time history; subsequent applications may involve closed-loop variation of pitch amplitude and phase in accordance with some measure of blade stall onset such as the RMS value of blade lag acceleration.

The pitch time history shown in Fig. 3 was tested on the model rotor of Fig. 4. Subsequent sections describe this model and the test procedure, results, and conclusions.

## Model Design and Construction

The rotor test facility is shown in Fig. 4. For simplicity and ease of modification it was decided to equip a single rotor blade with electromechanical pitch control, counter balanced by two "dummy" blades of 5/8 inch steel drill rod and adjustable counterweights. Geometric restrictions were imposed upon the hardware, however, to make it possible to add two more identical but separate pitch actuators without redesign.

The blade used in the test rotor was the same as that of Reference 3, having a NACA 0012 section with a 21.2-inch length and two-inch chord. It had an eight degree linearly decreasing twist from root to tip and was constructed of fiberglass with aluminum reinforcing. The blade was connected to the rotor hub by means of a ball-and-socket root fixture permitting flapping, lagging and feathering degrees of freedom about the same point. A complete set of rotor parameters can be found in Ref. 3.

The individual-blade-control assembly consisted of a shaft-mounted servo motor that, through a series of linkages, acted as a position controller of the rotor blade pitch angle. The motor/tachometer was mounted between two 1/4-inch-thick disks of aluminum, which also held two counterweights to offset the inertia contribution of the motor. These disks were fixed to the shaft by two aluminum blocks containing two setscrews and a keyway. Also, attached to the forward disk was an aluminum support for the transmission shaft of the control assembly.

This transmission shaft was mounted at a right angle to the motor shaft, and was give its rotation by a spiral-bevel gear that was driven by a pinion on the motor shaft, with a 2:1 gear reduction ratio. This same shaft was attached to a thin aluminum bar that had a threaded rod inserted through its other end, and parallel to the transmission shaft. Mounted on the threaded rod was yet another actuator link that consisted of two rod ends screwed together by a threaded metal coupling. The other end of the link was connected to a bolt that passes through the blade pitch axis.

The rotor blade was rigidly attached to a steel fork assembly that, in turn, bolted to the inner race of a spherical bearing. The spherical bearing was then contained within a steel support block that was clamped fast to the main rotor hub thus allowing fully articulated blade motion with concentric pitch, flap and lead-lag axes, offset from the hub by approximately two inches. The blade root fixture was instrumented with strain gauges mounted on a .005-inch-thick curved steel flexure that was free to turn about the lead-lag axis, but gave a torsional output corresponding to blade flapping, and a bending output corresponding to blade pitch angle. This particular flexure geometry was chosen as a solution to the problem of uncoupling the three rigid degrees of freedom of the blade for purposes of measurement. A thickness of .005 inches was selected for the flexure to produce a significant signal for small blade deflections, while at the same time providing negligible resistance to the blade flapping motion.

Since the servo motor was to function as a position control device, it was necessary to incorporate appropriately weighted feedback signals to the motor amplifier. These signals were the motor speed, taken from the tachometer and the angular position, measured from the torsional strain gage mounted on the steel fixture attached to the blade.

## Test Procedure and Results*

The test procedure was as follows. At a rotor rotational speed of 400 rpm and zero blade collective pitch (at three-quarter radius), tunnel speed was increased until a rotor advance ratio of 0.4 was achieved. Collective pitch was then incrementally increased; after each increment, 1P cyclic pitch was applied electronically to trim the rotor tip-path-plane to a position perpendicular to the rotor shaft. This procedure was necessary to ensure that a mechanical limitation on total blade pitch variation of twenty degrees peak-to-peak was not exceeded.

*The test results were obtained by R.M. McKillip and P.H. Bauer, using software developed by R.M. McKillip.

At the desired test condition, a data set was taken prior to the application of stall alleviation control to the blade. Then a computer-controlled 2P and 3P blade cyclic pitch time history similar to that of Fig. 3 was superimposed on the existing blade collective and 1P cyclic pitch. A further data set was then taken.

Typical test time history and spectral data are shown in Figs. 5 and 6, in terms of blade pitch and blade tip lag accelerometer signals for a case similar to that of Fig. 3; note that blade collective pitch was increased to ten degrees and the rotor shaft was tilted forward ten degrees. More extreme conditions could not be achieved due to mechanical interference between the blade pitch mechanism and the rotor hub fitting. It was expected that blade stall would be manifested by harmonics of lag acceleration higher than 3P in the vicinity of $\psi$=270 degrees; such a harmonic is seen to occur in Fig. 5(a). A significant 5P harmonic is also seen in Fig. 6(a). Application of 2P and 3P cyclic pitch eliminates the 5P peak at $\psi$=270 degrees in Fig. 5(b), and reduces the 5P peak in Fig. 6(b). Due to blade mechanical pitch limitations, substantial blade stall was not present, and therefore the stall alleviation indicated by the results of Figs. 5(b) and 6(b) could not be demonstrated for a more extreme stall condition. Further testing should be conducted with total model blade pitch capability increased fifty per cent, i.e., from twenty to thirty degrees.

## Concluding Remarks

Application of 2P and 3P cyclic pitch reduced 5P blade lag accelerations believed to be associated with rotor blade stall. However, due to blade mechanical pitch limitations, substantial blade stall was not encountered, and therefore conclusive demonstration of the success of 2P and 3P cyclic pitch in alleviating more extreme rotor blade stall must await testing with increased total model blade pitch capability.

The approach, however, is considered promising.

FIG. 1 HELICOPTER INDIVIDUAL-BLADE-CONTROL AND ITS APPLICATIONS

GUST ALLEVIATION
STALL FLUTTER SUPPRESSION
LAG DAMPING AUGMENTATION
VERTICAL VIBRATION ALLEVIATION
INPLANE VIBRATION ALLEVIATION
FLAPPING STABILIZATION AT HIGH ADVANCE RATIO
STALL ALLEVIATION
FLYING QUALITIES ENHANCEMENT
PERFORMANCE ENHANCEMENT
AUTOMATIC BLADE TRACKING

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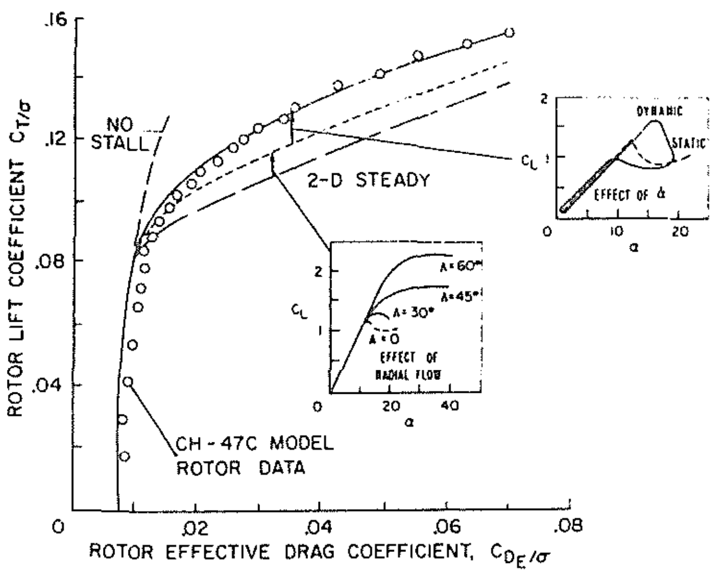


FIG. 2 Radial Flow and Dynamic Stall Effects on Rotor Performance at $\mu$ = 0.35 (Ref. 8 and W.J. McCroskey)

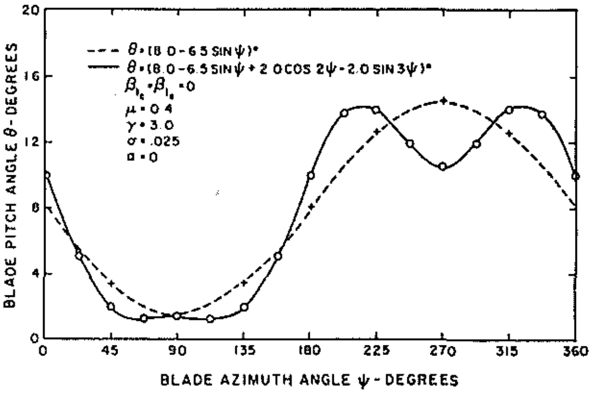


FIG. 3 Blade Pitch Angle Time History for Stall Alleviation

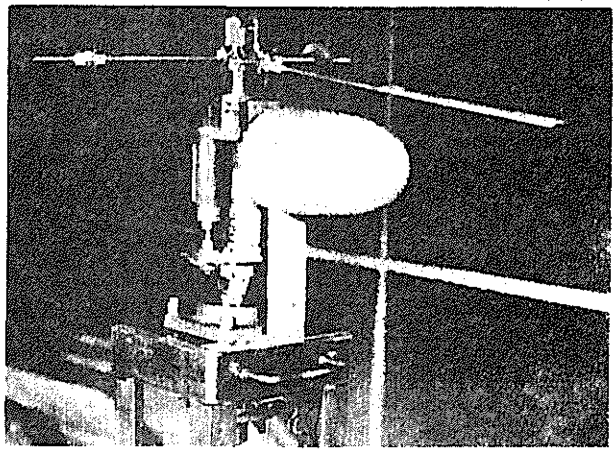

FIG. 4 Individual-Blade-Control Model Rotor Assembly

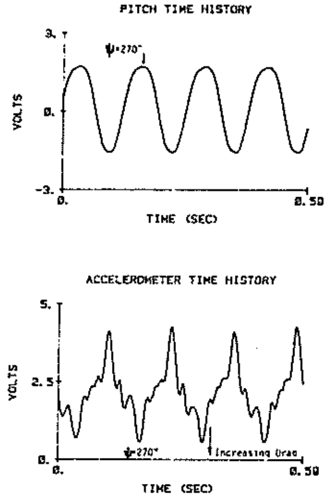


FIG. 5(a) Blade Pitch and Lag Accelerometer Time Histories: $\theta_0=10°$, $\alpha=-10°$, $\theta_{1_s}=-6.5°$, $\theta_{2_c}=\theta_{3_s}=0°$, $\mu=0.4$

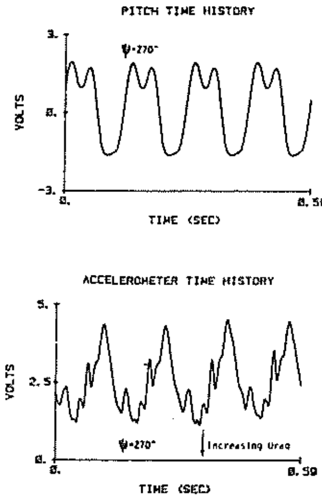


FIG. 5(b) Blade Pitch and Lag Accelerometer Time Histories: $\theta_0=10°$, $\alpha=-10°$, $\theta_{1_s}=-6.5°$, $\theta_{2_c}=-\theta_{3_s}=2.0°$, $\mu=0.4$

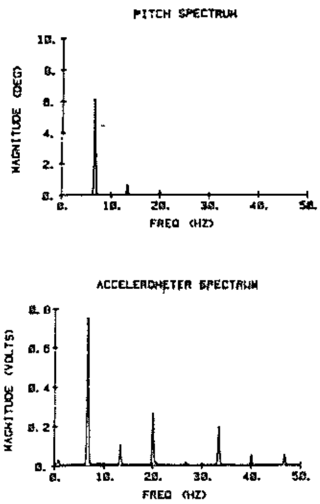


Fig. 6(a) Blade Pitch and Lag Accelerometer Spectra: $\theta_0=10°$, $\alpha=-10°$, $\theta_{1_s}=-6.5°$, $\theta_{2_c}=\theta_{3_s}=0°$, $\mu=0.4$

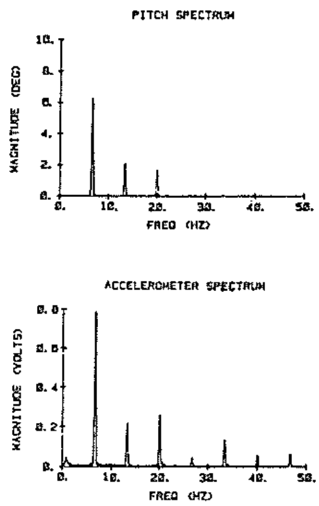


FIG. 6(b) Blade Pitch and Lag Accelerometer Spectra: $\theta_0=10°$, $\alpha=-10°$, $\theta_{1_s}=-6.5°$, $\theta_{2_c}=-\theta_{3_s}=2.0°$, $\mu=0.4$

*HELICOPTER ATTITUDE STABILIZATION USING INDIVIDUAL-BLADE-CONTROL*

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## Abstract

A new, advanced type of active control for helicopters and its application to the solution of rotor aerodynamic and aeroelastic problems is described. Each blade is individually controlled in the rotating frame over a wide range of frequencies up to the sixth harmonic of rotor speed.

The concept of *Individual-Blade-Control* (IBC) embodies the control of individual blade pitch by means of broad-band electrohydraulic actuators attached to the swash plate (in the case of three blades) or individually to each blade, using signals from accelerometers mounted on the blades to supply appropriate control commands to the actuators. Note that the IBC involves not only control of each blade independently, but also a feedback loop for each blade in the rotating frame. In this manner, it becomes possible to alleviate the severe effects of blade-vortex interaction, blade-fuselage interference, atmospheric turbulence, and adverse vehicle dynamics.

The present paper describes the design of a system controlling blade flapping dynamics, and related testing of the system on a model rotor in the wind tunnel. The control inputs considered are blade pitch changes proportional to blade flapping acceleration, velocity, and displacement. The effect of such a system on helicopter rotor damping-in-pitch, and angle-of-attack stability is then evaluated. It is shown that helicopter attitude stabilization is achieved, with a corresponding improvement in flying qualities.

## Introduction

A truly advanced helicopter rotor must operate in a *severe aerodynamic environment* with high reliability and low maintenance requirements. This environment includes:

(1) atmospheric turbulence (leading to impaired flying qualities, particularly in the case of hingeless rotor helicopters).

(2) retreating blade stall (leading to large torsional loads in blade structure and control system).

(3) blade vortex interaction in transitional and nap-of-the-earth flight (leading to unacceptable higher harmonic blade bending stresses and helicopter vibration).

(4) blade-fuselage interference (leading to unacceptable higher harmonic blade bending stresses and helicopter vibration).

(5) blade instabilities due to flap-lag coupling and high advance ratio.

The application of feedback techniques make it possible to alleviate the effects described in items (1) and (5) above, while improving helicopter vibration and handling characteristics to meet desired standards. The concept of Individual-Blade-Control (IBC) embodies the control of broadband electrohydraulic actuators attached to each blade or the swash plate, using signals from sensors mounted on the blades to supply appropriate control commands to the actuators.[1-7] Note that IBC involves not just control of each blade independently, but also a feedback loop for each blade in the rotating frame. In this manner it becomes possible to reduce the severe effects of atmospheric turbulence, retreating blade stall, blade-vortex interaction, blade-fuselage interference, and blade instabilities, while providing improved flying qualities.

It is evident that the IBC system will be most effective if it is comprised of several sub-systems, each controlling a specific mode, e.g., the blade flapping mode, the first blade lag mode, the first blade flatwise bending mode, and the first blade torsion mode. Each sub-system operates in its appropriate frequency band.

The configuration used in this investigation employs an individual actuator and multiple feedback loops to control each blade. These actuators and feedback loops rotate with the blades and, therefore, a conventional swash plate is not required. However, the same degree of individual-blade-control can be achieved by placing the actuators in the non-rotating system and controlling the blades through a conventional swash plate if the number of control degrees-of-freedom equals the number of blades. For more than three blades, the use of extensible blade pitch control rods in the form of

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hydraulic actuators is a possibility. Note that all IBC functions not involving differential collective pitch can be obtained on a four-bladed rotor using a conventional swash plate.

The present paper is primarily concerned with the application of the Individual-Blade-Control concept to helicopter attitude stabilization. Other applications are described in Refs. 1-6, and listed in Figure 1.

## Technical Discussion

Reference 3 describes the application of Individual-Blade-Control to helicopter gust alleviation. The feedback blade pitch control was proportional to blade flapping acceleration and displacement, i.e.,

$$\Delta\theta = -K\left(\frac{\ddot{\beta}}{\Omega^2} + \beta\right)$$

The Wright Brothers Wind Tunnel at M.I.T. was used for gust alleviation testing. The test section is a 7' x 10' oval, and for rotor testing the turntable is equipped with two trunnions for horizontal mounting of the rotor shaft. This particular orientation was chosen to permit use of the existing gust generator (Fig. 2).

Mounted outside of the test section was a hydraulic motor and slip-ring assembly, providing shaft rotation and data transmission from the rotating frame to the analog computer in the fixed frame. Clamped to the far trunnion was another slip-ring assembly that transmitted electrical current to the servo-motor and tachometer.

The rotor shaft was secured to the support bearings with the rotor plane in the center of the tunnel section. Instrumentation consisted of a difference amplifier, for the amplification of blade flapping and feathering strain gage signals; a portable analog computer and servo amplifier, for processing the feedback loop signals and supplying the motor driving signal; a dual-beam storage oscilloscope, for monitoring the flap and pitch signals; a spectrum analyzer, for on-line analysis of the blade flapping response; an X-Y plotter, for the production of a hard record of the analyzer output; another oscilloscope for quick visualization of the output of the spectrum analyzer; a difference amplifier for the amplification of the accelerometer signals; a hot-wire probe and amplifier, for measurement of the gust amplitude; and finally, a PDP-11 computer, for analog-to-digital data acquisition and real-time Fast Fourier Transform analysis.

In the wind tunnel test, the parameters varied were gust excitation frequency, tunnel speed, and feedback gain. A typical time history of the gust, flapping, pitch, and accelerometer signals for the $\mu = 0.4$ case can be seen in Figure 3, and the spectral decomposition of this run is shown in Figure 4.

Figure 5 shows the effect of increasing the open-loop gain K upon the IBC gust alleviation system performance. Note that the experimental reduction in gust-induced flapping response is in accordance with the theoretical closed-loop gain $1/(1+K)$.

The Lock number of the model blade was 3.0. For a full size rotor, the increase in damping due to the increase in Lock number results in the flapping at excitation frequency becoming the dominant response.[8] Also, with increased blade damping it becomes possible to use higher feedback gain for the same stability level, and as a consequence the IBC system performance improves with increasing Lock number.

Following the successful alleviation of gust disturbances using the IBC system, Ref. 3 showed the theoretical equivalence of blade flapping response to other low-frequency disturbances, e.g., helicopter pitch and roll attitude as follows:

The flapping equation of motion in rotating coordinates for a blade with zero hinge offset is

$$\frac{\ddot{\beta}}{\Omega^2} + \frac{\gamma}{8}\left[1 + \frac{4}{3}\mu\sin\psi\right]\frac{\dot{\beta}}{\Omega} + \left[1 + \frac{\gamma}{6}\mu\cos\psi + \frac{\gamma}{8}\mu^2\sin 2\psi\right]\beta$$

$$= \frac{\gamma}{8}\left[(1+\mu^2) + \frac{8}{3}\mu\sin\psi - \mu^2\cos 2\psi\right]\theta - \frac{\gamma}{6}\left[1 + \frac{3}{2}\mu\sin\psi\right]\Delta\lambda$$

where the incremental inflow $\Delta\lambda$ includes such effects as gusts, rotorcraft vertical disturbances, and blade-vortex/blade-fuselage interaction.

Blade pitch angle $\theta$ with respect to inertial space is

$$\theta = \theta_o + (\theta_{1_c} - \phi)\cos\psi + (\theta_{1_s} - \alpha)\sin\psi$$

where $\alpha$ and $\phi$ are rotorcraft pitch and roll angles.

For the special case of a gust or vertical disturbance, $\Delta\lambda = \lambda_G + \delta$ and the equation becomes

$$\begin{aligned}\frac{\ddot{\beta}}{\Omega^2} + \frac{\gamma}{8}[\ ]\frac{\dot{\beta}}{\Omega} + [\ ]\beta = {} & \frac{\gamma}{8}\left[\theta_o(1+\mu^2) + \frac{4}{3}\mu(\theta_{1_s} - \alpha)\right] \\ & + \frac{\gamma}{8}\left[(\theta_{1_c} - \phi)\left(1 + \frac{1}{2}\mu^2\right)\right]\cos\psi \\ & + \frac{\gamma}{8}\left[\frac{8}{3}\mu\theta_o + (\theta_{1_s} - \alpha)\left(1 + \frac{3}{2}\mu^2\right)\right]\sin\psi \\ & - \frac{\gamma}{6}(\lambda_G + \delta) \\ & - \frac{\gamma}{4}\mu(\lambda_G + \delta)\sin\psi\end{aligned}$$

neglecting harmonics above the first. It is seen that low frequency pitching ($\alpha$), rolling ($\phi$), horizontal ($\mu$), gust ($\lambda_G$), and vertical ($\delta$) disturbances can be alleviated by the same IBC system.

The following section considers the theoretical application of the IBC system to helicopter attitude stabilization.

## Theoretical Analysis

Reference 9 applies the theory of Ref. 10 and unpublished work by R.H. Miller to the successful prediction of helicopter longitudinal stability and control characteristics as measured in flight test. This theory will now be applied to an analysis of the effect of the IBC system on helicopter longitudinal attitude stability.

Taking moments about the flapping hinge leads to the flapping equation of motion

$$I_1\ddot{\beta} + I_1\Omega^2\beta = \int_o^R r dL \qquad (1)$$

where $I_1$ = blade flapping moment of inertia

$\Omega$ = rotor rotational speed

$dL$ = blade elemental lift at spanwise station $r$

$R$ = rotor radius

and zero hinge offset is assumed for simplicity.

In non-rotating inertial coordinates, blade flapping motion is given by

$$\beta = \beta_o(t) + \beta_{1_c}(t)\cos\psi + \beta_{1_s}(t)\sin\psi$$

neglecting harmonics higher than the first. Note that in disturbed flight $\beta_o$, $\beta_{1_c}$, and $\beta_{1_s}$ are functions of time.

The elemental lift is

$$dL = \frac{1}{2}\rho ac\, U_T^2 \left[\theta - \frac{U_P}{U_T}\right] dr$$

Where $\rho$ = air density

$a$ = blade section lift-curve slope

$c$ = blade chord

$U_T = \Omega r + \mu\Omega R \sin\psi$

$U_P = \lambda\Omega R + r\dot{\beta} + \mu\Omega R\beta \cos\psi$

$\lambda = \frac{v}{\Omega R} + \frac{V_{vertical}}{\Omega R}$

and $\theta = \theta_o + (\theta_{1_c} - \phi)\cos\psi + (\theta_{1_s} - \alpha)\sin\psi + \Delta\theta$

where $\lambda$, $\beta$, and $\theta$ are measured with respect to a horizontal inertial plane, and $\theta$ includes the effect of the IBC system:

$$\Delta\theta = -K_A \frac{\ddot{\beta}}{\Omega^2} - K_R \frac{\dot{\beta}}{\Omega} - K_P\beta$$

This expression assumes a high performance actuator, a reasonable assumption for the wide bandwidth characteristic of the IBC system.

Harmonic balance is applied to equation (1) to obtain the longitudinal flapping equation of motion:

$$-2\frac{\dot{\beta}_{1_c}}{\Omega}\left[1+\frac{\gamma}{8}K_A\left(1+\frac{3}{2}\mu^2\right)\right] \approx \frac{\gamma}{2}\left\{\frac{2}{3}\mu\,\theta_o + \frac{1}{4}\left[\beta_{1_c}(1+K_R) - \alpha + \theta_{1_s}\right]\left(1+\frac{3}{2}\mu^2\right) - \frac{1}{2}\mu\lambda - \frac{1}{2}\mu^2\beta_{1_c}\right\} \qquad (2)$$

not including coupling with lateral flapping $\beta_{1_s}$ *that is negligible at low frequency (see* Appendix).

For small displacements of the rotor tip-path-plane from equilibrium, equation (2) can be used to obtain the following perturbation relationship between shaft angle $\alpha$ and longitudinal flapping $\beta_{1_c}$:

$$\Delta\alpha - \Delta\beta_{1_c} = A\,\Delta\frac{\dot{\beta}_{1_c}}{\Omega} + B\Delta\beta_{1_c} \qquad (3)$$

where $A = 2\left[1+\frac{\gamma}{8}K_A\left(1+\frac{3}{2}\mu_o^2\right)\right] / \frac{\gamma}{8}\left(1+\frac{3}{2}\mu_o^2\right)$

$$B = K_R - (2\mu_o^2)/(1+\frac{3}{2}\mu_o^2)$$

and the subscript zero denotes trim conditions.

Figure 6 indicates that any effect tending to increase the quantity $\Delta\alpha - \Delta\beta_{1_c}$ produces a a stabilizing moment Th $(\Delta\alpha - \Delta\beta_{1_c})$ about the helicopter center-of-gravity. Therefore, positive terms on the RHS of equation (3) are stabilizing. It is seen that the IBC system *increases the rotor* damping-in-pitch parameter A and the rotor angle-of-attack stability parameter B.

## Discussion of Results

Equation 3 was used to investigate the effect of the IBC system on helicopter longitudinal attitude stability at various forward speeds. Stability parameters A and B are plotted in Figs. 7 and 8 as a function of advance ratio $\mu$ for a helicopter having a blade Lock number of 8 and IBC open loop gains $K_A = K_R = K_P = 0.5$ (see Appendix for effect of $K_P$). For these arbitrary values, it is seen that the rotor disk longitudinal damping-in-pitch is increased over fifty per cent (Fig. 7), while the rotor disk angle-of-attack dependence changes from unstable to stable (Fig. 8).

The physical origin of these effects is as follows. *To precess the rotor disk with a* longitudinal pitching velocity $\Delta\dot{\beta}_{1_c}$, the rotor disk must lag behind the shaft an amount

$(\Delta\alpha - \Delta\beta_{1_c})$ to generate the necessary rolling moment. Since the $K_A \ddot{\beta}/\Omega^2$ feedback represents an effective increase in blade flapping inertia, the required lag is increased, thus increasing the stabilizing moment proportional to $\dot{\beta}_{1_c}$, i.e., rotor damping-in-pitch. The rotor angle-of-attack instability is proportional to disk attitude perturbations $\Delta\beta_{1_c}$. The $K_R \dot{\beta}/\Omega$ feedback opposes increases in disk attitude $\beta_{1_c}$ (defined positive nose down) through the flapping velocity perturbation $\Delta\dot{\beta} = -\Delta\beta_{1_c}\,\Omega \sin\psi$ which produces an aerodynamic moment opposing $\Delta\beta_{1_c}$; the tendency of the disk to follow the shaft is reduced, producing a perturbation lag $(\Delta\alpha - \Delta\beta_{1_c})$ and a stabilizing moment proportional to $\Delta\beta_{1_c}$, i.e., rotor longitudinal angle-of-attack stability.

Roll attitude stabilization also results from the IBC system described above. If it were desired to reduce the roll stabilization due to the helicopter rolling inertia being less than its pitching inertia, gains could be varied in accordance with signals from a fuselage-mounted roll rate gyro.

Attenuation of the response to pilot's control can be prevented by biasing the feedback signals by a signal proportional to stick displacement.

## Conclusions

Following the successful theoretical and experimental demonstration of the IBC helicopter rotor gust alleviation system utilizing blade-mounted accelerometers, a theoretical study has shown that the same system can provide rotor attitude stabilization for disturbances in the non-rotating system (such as helicopter pitch and roll) similar to the gust disturbance previously investigated experimentally.

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FIG. 1 HELICOPTER INDIVIDUAL-BLADE-CONTROL AND ITS APPLICATIONS

- GUST ALLEVIATION
- STALL FLUTTER SUPPRESSION
- LAG DAMPING AUGMENTATION
- VERTICAL VIBRATION ALLEVIATION
- INPLANE VIBRATION ALLEVIATION
- FLAPPING STABILIZATION AT HIGH ADVANCE RATIO
- STALL ALLEVIATION
- FLYING QUALITIES ENHANCEMENT
- PERFORMANCE ENHANCEMENT
- AUTOMATIC BLADE TRACKING

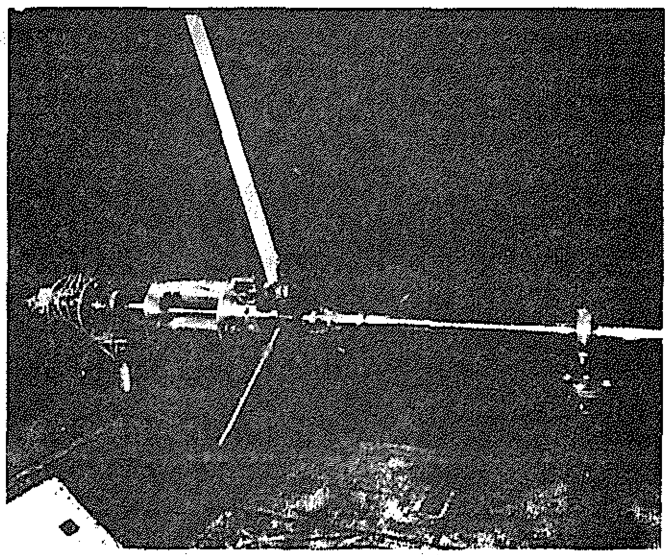

FIG. 2 Individual-Blade-Control Model Rotor Assembly, Upstream View

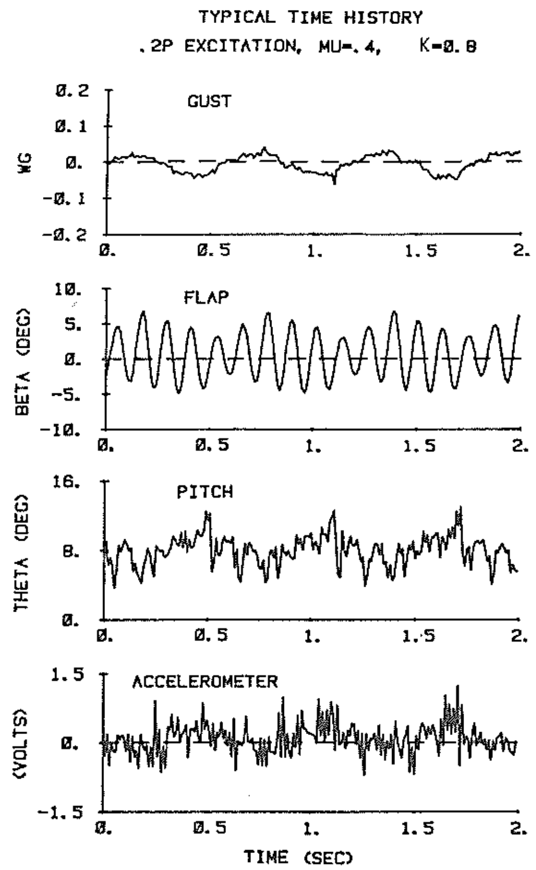


FIG. 3 Typical I.B.C. Gust System Experimental Time History ($\mu$ = 0.4)

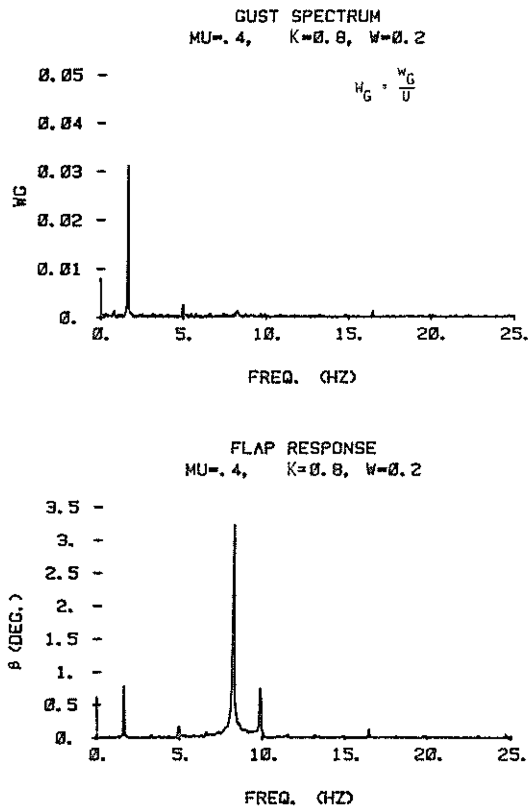


FIG. 4a Spectral Decomposition of Experimental Gust and Flap Angle Data

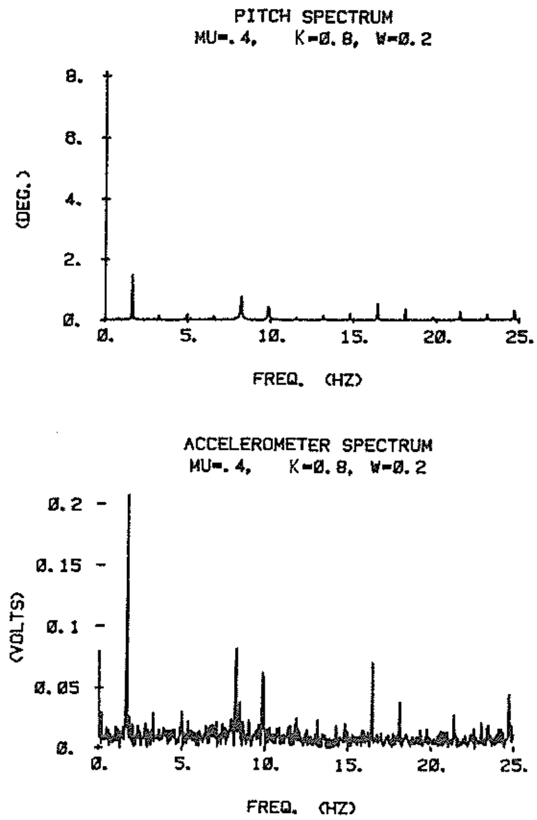


Fig. 4b Spectral Decomposition of Experimental Blade Pitch and Accelerometer Data

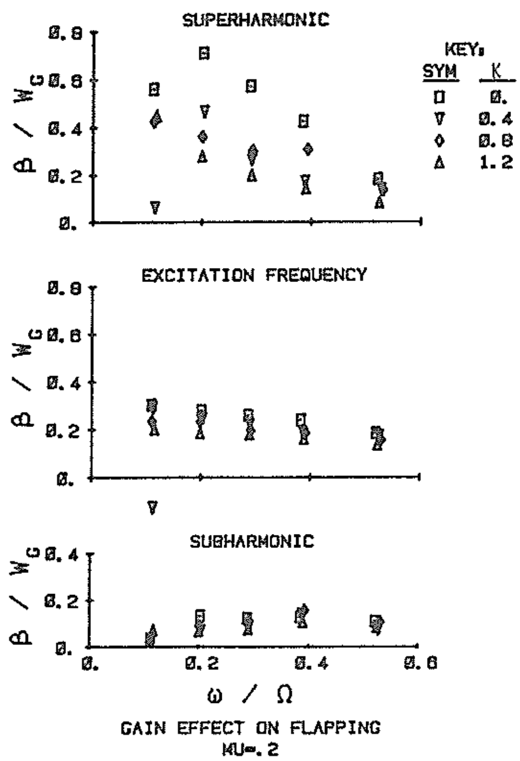


FIG. 5a Effect of Feedback Gain on Flap Response to Gust ($\mu$ = 0.2)

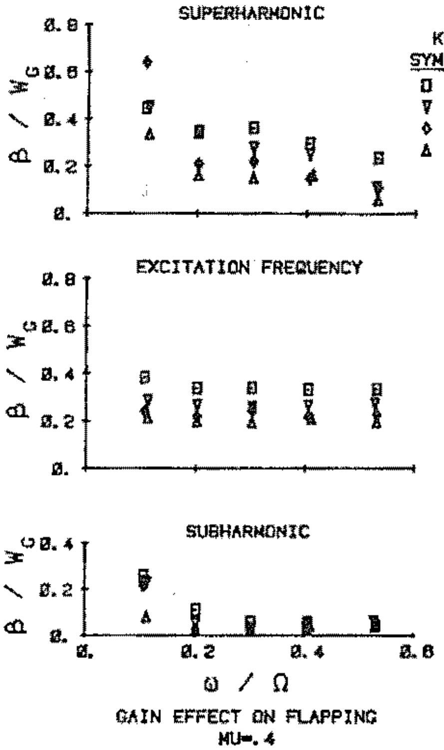


FIG. 5b Effect of Feedback Gain on Flap Angle Response to Gust ($\mu$ = 0.4)

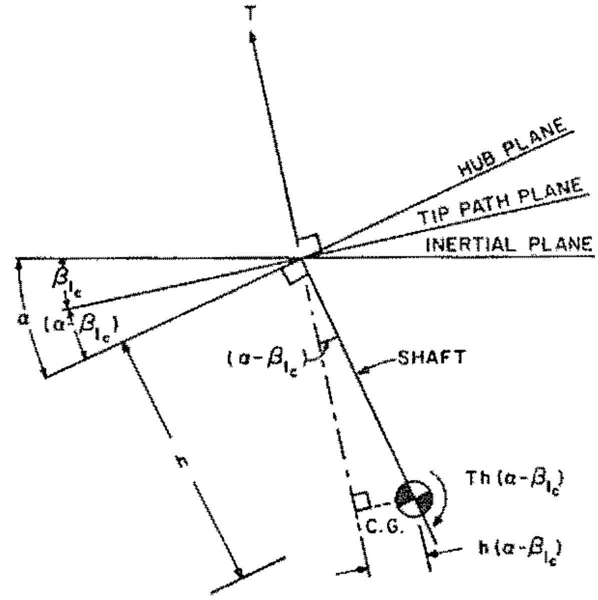


FIG. 6 Geometry for Longitudinal Stability Analysis

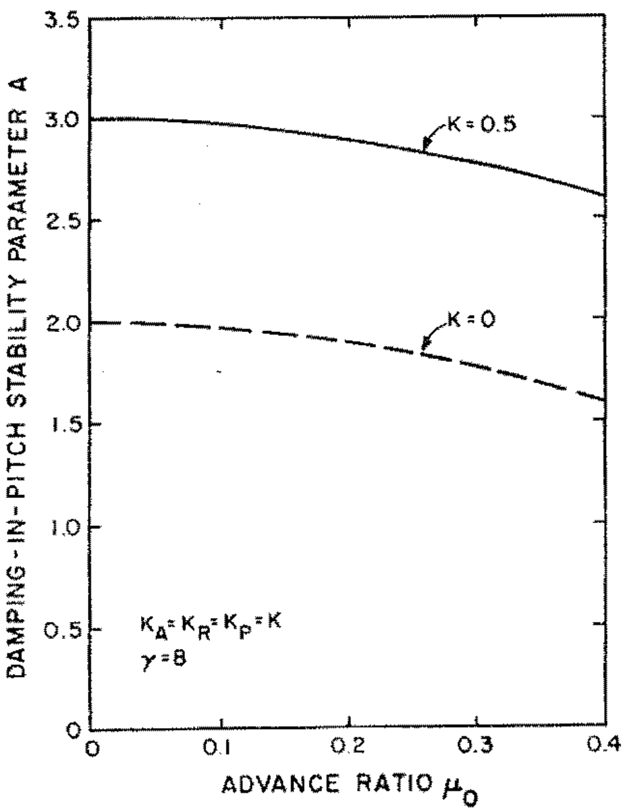


FIG. 7 Rotor Damping-in-Pitch Stability Parameter Versus Advance Ratio

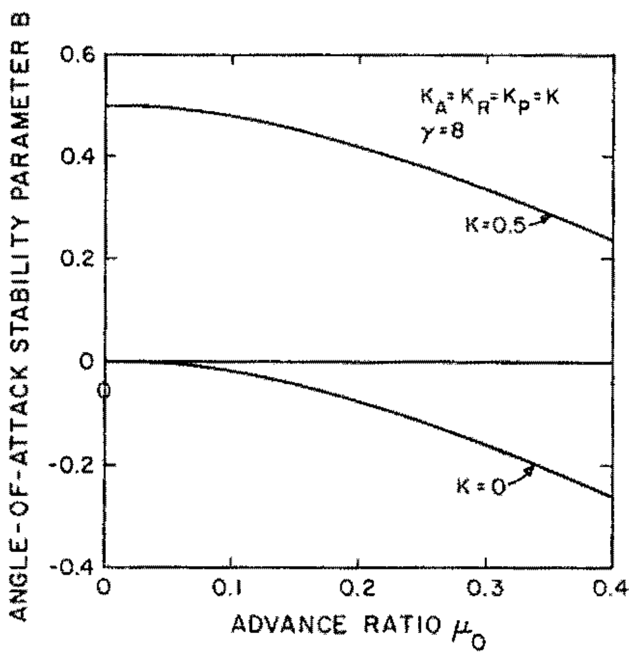


FIG. 8 Rotor Angle-of-Attack Stability Parameter Versus Advance Ratio

## Appendix

Following unpublished work by R.H. Miller, and including the effect of an IBC system having blade pitch commands

$$\Delta\theta = -K_A \frac{\ddot{\beta}}{\Omega^2} - K_R \frac{\dot{\beta}}{\Omega} - K_p\beta$$

the longitudinal and lateral flapping equations of motion in hover (for simplicity) are

$$(1 + \frac{\gamma}{8} K_A) \frac{\ddot{\beta}_{1_s}}{\Omega^2} + \frac{\gamma}{8} (1 + K_R) \frac{\dot{\beta}_{1_s}}{\Omega} + \frac{\gamma}{8} (K_P - K_A)\beta_{1_s} -$$

$$2(1 + \frac{\gamma}{8} K_A) \frac{\dot{\beta}_{1_c}}{\Omega} - \frac{\gamma}{8} (1 + K_R) \beta_{1_c} = \frac{\gamma}{8} (\theta_{1_s} - \alpha)$$

$$2(1 + \frac{\gamma}{8} K_A) \frac{\dot{\beta}_{1_s}}{\Omega} + \frac{\gamma}{8} (1 + K_R) \beta_{1_s} + (1 + \frac{\gamma}{8} K_A) \frac{\ddot{\beta}_{1_c}}{\Omega^2} +$$

$$\frac{\gamma}{8} (1 + K_R) \frac{\dot{\beta}_{1_c}}{\Omega} + \frac{\gamma}{8} (K_P - K_A) \beta_{1_c} = \frac{\gamma}{8} (\theta_{1_c} - \phi)$$

Taking $K_A = K_R = K_P = K$ and $\gamma = 8$, these equations become

$$\frac{\ddot{\beta}_{1_s}}{\Omega^2} + \frac{\dot{\beta}_{1_s}}{\Omega} - 2 \frac{\dot{\beta}_{1_c}}{\Omega} - \beta_{1_c} = \frac{1}{1+K} (\theta_{1_s} - \alpha) \tag{A1}$$

$$2 \frac{\dot{\beta}_{1_s}}{\Omega} + \beta_{1_s} + \frac{\ddot{\beta}_{1_c}}{\Omega^2} + \frac{\dot{\beta}_{1_c}}{\Omega} = \frac{1}{1+K} (\theta_{1_c} - \phi) \tag{A2}$$

Note that the choice $K_P = K_A$ eliminates the undesirable displacement coupling between $\beta_{1_c}$ and $\beta_{1_s}$.

Taking Laplace transforms of equations (A1) and (A2) term-by-term and solving the resulting algebraic equations for the flapping longitudinal and lateral response to a longitudinal shaft pitching disturbance $\alpha$, there results

$$(1+K) \frac{\bar{\beta}_{1_c}}{\bar{\alpha}} = \frac{2\nu + 1}{\nu^4 + 2\nu^3 + 5\nu^2 + 4\nu + 1} \tag{A3}$$

$$(1+K) \frac{\bar{\beta}_{1_s}}{\bar{\alpha}} = \frac{-\nu(\nu + 1)}{\nu^4 + 2\nu^3 + 5\nu^2 + 4\nu + 1} \tag{A4}$$

where the barred quantities represent Laplace transforms and $\nu = s/\Omega$.

The frequency response to shaft pitching excitation at $\omega$ is obtained by setting $s = i\omega$ in equations (A3) and (A4). It is seen that for $\omega/\Omega$ (vehicle frequency/rotor frequency) small, the longitudinal flapping response $\bar{\beta}_{1_c}/\bar{\alpha}$ is of order unity, while the lateral flapping response $\bar{\beta}_{1_s}/\bar{\alpha}$ is of order $\omega/\Omega$. Therefore the coupling between $\beta_{1_c}$ and $\beta_{1_s}$ is negligible for low frequency longitudinal excitation. Low-frequency decoupling for lateral excitation can be demonstrated in the same manner.