Testing of a Stall Flutter Suppression System for Helicopter Rotors Using Individual-Blade-Control

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The development and testing of a feedback system designed to alleviate rotor blade first torsion mode oscillations associated with stall flutter are described. The system employs blade-mounted sensors to detect torsional oscillations and provide feedback in order to increase the damping of the first torsional mode. A model of the blade and control system dynamics is developed and is used to give qualitative and quantitative guidance in the design process, as well as to aid in the analysis of experimental results. System performance in wind tunnel tests in forward flight is described, and experimental results show that the system can provide substantial additional damping to stall-induced oscillations.

Nomenclature

D	= motor viscous friction constant, N-m/s
D_A	= aerodynamic damping constant, N-m/s
G_{O_L}	= open loop transfer function
I	= motor current, A
J_B	= blade pitching inertia, kg/m^2
J_T	= inertia of motor, tachometer, and linkage, kg/m^2
K_E	= volts back EMF, V/s
K_F	= control system feedback gain
K_{NR}	= nonrotating torsional spring constant, N/m
K_p	= feedback potentiometer gain
K_T	= torque sensitivity of motor, N-m/A
K_θ	= pitch angle feedback gain
$K_{\dot\theta}$	= tachometer feedback gain
ℓ	= displacement of accelerometer from pitch axis, in.
L_a	= motor inductance, Ω-s
$p_{(\ )}$	= system poles
R	= motor resistance, Ω
S	= Laplace operator
V	= voltage input to motor, V
$\bar{(\ )}$	= complex conjugate
$\dot{(\ )}$	= differentiation *w/r* time
ζ	= damping ratio of blade pitch motion, % critical
θ	= blade pitch angle, rad
θ_c	= blade pitch angle command, rad
θ_0	= collective pitch, rad
θ_1	= cyclic pitch, rad
μ	= advance ratio
τ	= actuator time constant, sec
ω	= rotor rotation frequency, Hz
ω_θ	= blade first torsion frequency, Hz

Introduction

As increased demands on helicopter performance have pushed machines to higher values of blade loading and advance ratio, one persistent problem for the designer has been the aeroelastic instability known as stall flutter. This phenomenon has been studied extensively in a variety of other works (Refs. 1-4, for example) and a comprehensive discussion of its sources and effects is not necessary for present purposes. However, a brief summary of the salient points is helpful for posing the design problem dealt with herein.

It has been well documented that an airfoil oscillating rapidly in pitch is able to operate transiently at angles of attack considerably in excess of its static stall angle without flow separation taking place. However, at sufficiently high angles of attack, the airfoil stalls, though this so-called dynamic stall differs considerably from conventional static airfoil stall. As shown by Ham,[4] dynamic stall is characterized by the loss of leading-edge suction and the subsequent movement of a strong shed vortex aft from the leading edge, a movement which generates strong nose-down pitching moments on the airfoil. With proper combinations of airfoil mean angle of attack, amplitude of motion, and reduced frequency, this stalling phenomenon can cause hysteresis in the aerodynamic pitching moment which can then lead to a net influx of energy to the airfoil's pitching motion.

This hysteresis is the source of the well-known torsional instability referred to as stall flutter. The motion associated with stall flutter is generally unstable over only part of the azimuth and damps out rapidly as the blade swings around toward the advancing side. However, even the one or two cycles of large amplitude torsional motion that do occur are sufficient to put extreme loads on the rotor control system (see Fig. 1).

The adverse effects of stall flutter could, of course, be alleviated by such expedients as increasing blade solidity. However, such a change would increase the drag of the rotor

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and would penalize the performance of the helicopter at all speeds. Also, the development of airfoils with more benign dynamic stall characteristics may be possible, but attempts to identify such airfoils have so far been inconclusive (e.g., Ref. 5).

Applying Individual Blade Control (IBC) techniques to this problem offers one possible solution. Reference 6 showed that appropriate feedbacks to a position control servo governing blade pitch motion could help reduce undesirable blade motions due to low-frequency gust inputs (see also Ref. 8 for a general discussion of IBC concepts). It was felt that similar methods could be applied to alleviate the torsional motions associated with stall flutter. To understand the overall concept employed, consider again for a moment the mechanism which drives the stall flutter oscillations. As noted previously, at high blade angles of attack and certain reduced frequencies, aerodynamic moment hysteresis causes a net input of energy to blade torsional motion, so that any small blade ocscillation grows with time. Such a situation is equivalent to a one-degree-of-freedom oscillating system with negative damping. Indeed, stall flutter can be conceived of as a phenomenon caused by a once-per-revolution variation in the effective damping of the blade in pitch. On the advancing side the blade experiences positive damping at low angles of attack, but on the retreating side the effective damping can temporarily become negative, leading to the self-limiting oscillations described above.

An effective stall flutter suppression system, then, would be one which could eliminate this one-per-rev excursion into negative damping. One way to achieve this end, which is suggested by classical control theory, is to provide a pitch rate feedback from the blade to the pitch control servo. The details of the rationale for this concept, its implementation, and the results of experiments based on it are given in the following sections.

An Idealized Stall Flutter Suppression System

The IBC concept postulates that the pitch of each rotor blade will be controlled individually by the action of electromechanical or electrohydraulic actuators. Thus it provides a natural framework for feedback control of the rotor blades. To see how this general design concept was applied to stall flutter suppression, consider the diagram of an idealized IBC system, given in Fig. 2a. A pitch rate feedback to the actuator control was an appropriate choice for bringing about additional damping of the blade's stall-induced torsional oscillations. Such a pitch rate signal was extracted by mounting appropriately oriented accelerometers on the blade in order to sense the onset of stall flutter oscillations. The acceleration signal was then integrated and an appropriate gain applied to provide the desired weighted feedback.

The plausibility of the rate feedback appproach is indicated by the simple root locus diagram in Fig. 2b, the blade dynamics are represented there by poles residing near the imaginary axis. The effect of the stall-induced negative aerodynamic damping is, essentially, to push these poles into the right half-plane for some fraction of the rotor rotation period. Using rate feedback to place a zero at the origin and applying a nonzero feedback gain, the damping of the blade oscillation will increase. For this simple model, any desired level of positive damping can be applied.

Clearly, the inclusion of actuator dynamics alters this simplified presentation. In such a case the pole trajectories given in Fig. 2b are substantially changed, even if aerodynamic effects are neglected. Also, root locus analysis is not, in a rigorous sense, applicable to this problem because of the highly nonlinear aerodynamic effects associated with the stall flutter. Despite these caveats, however, the following discussion will show that simple models and analysis methods such as root locus techniques serve as very useful points of departure for the design of an effective flutter suppression system.

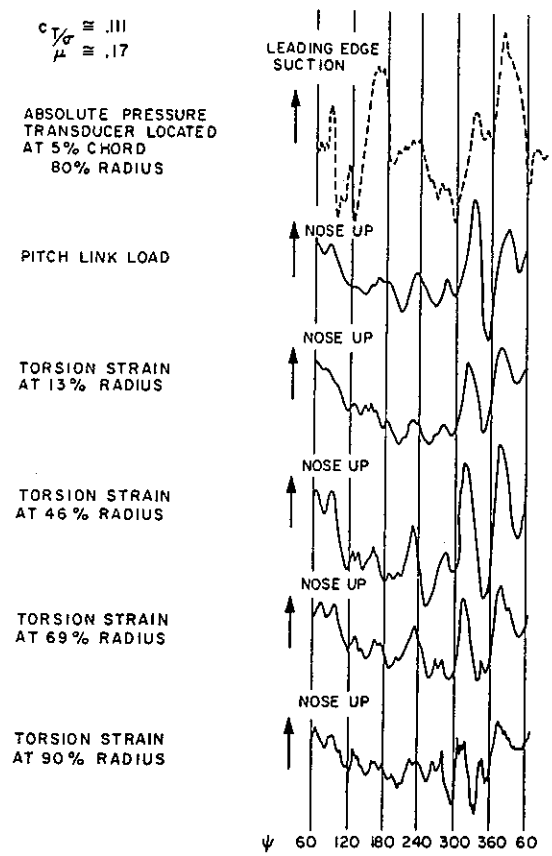
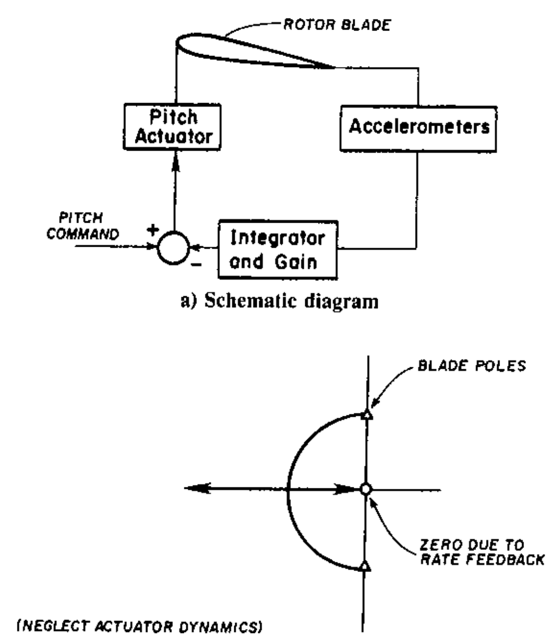


Fig. 1 Typical torsion loads on a rotor blade undergoing stall flutter.



a) Schematic diagram

b) Root locus

Fig. 2 Idealized Flutter Suppression System.

Model Design and Description

The model used here to test the proposed stall flutter suppression system is shown in Fig. 3 and was identical in most particulars to that used in Ref. 6. The test rotor used only a single blade, with a NACA 0012 section, a 28.7-in. radius, a 26% root cutout, and a 2.0 in. chord; further details are given in Table 1. The blade was attached to the hub by means of a steel fork which was in turn connected to a