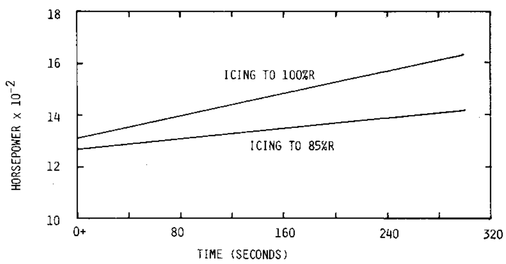

Fig. 3 Horsepower required to maintain a C_T of 0.004 as a function of icing time, CH47D helicopter rotor blade, hover condition.

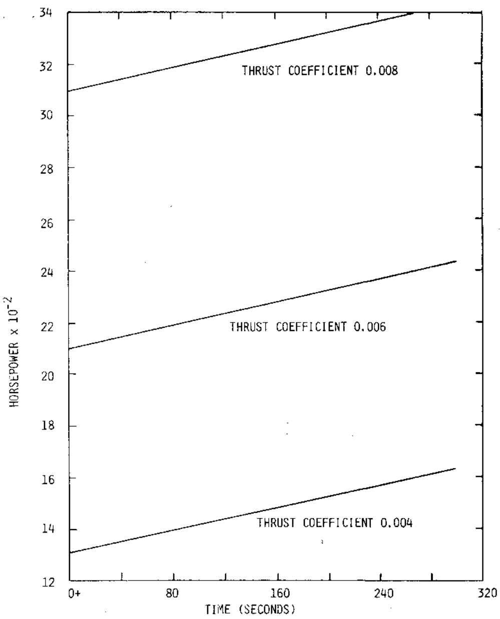

Fig. 4 Horsepower required to maintain a C_T of 0.004, 0.006, and 0.008 as a function of icing time, CH47D helicopter rotor blade, hover condition.

is dependent on the airfoil type.[3] The step drag increase at the inception of icing is based on surface roughness data.[3] The drag of the airfoil during icing is then found from the expression

$$C_{D_{\text{ice}}} = (1 + \Delta C_D / C_D) C_D \tag{2}$$

where C_D is the two-dimensional profile drag prior to icing as described in the airfoil data bank or found by existing airfoil analyses and/or experiment.

The factor $1 + \Delta C_D / C_D$, plotted as a function of radial location for selected times of 60, 180, and 300s in Fig. 2, indicates the strong dependence of the performance degradation on actual icing time. Also, these data can be displayed as a function of time for selected radial locations, where the growth in drag is linear with time starting from the frost point which is noted as a step in the $(1 + \Delta C_D / C_D)$ variation at time 0 + s.

When these drag increments are included in the airfoil data used for the hover analysis of the CH47D rotor, the resulting increase in required horsepower necessary to maintain a thrust level C_T of 0.004 as a function of icing time is shown in Fig. 3. The approximate 24% increase in required horsepower for a five minute natural icing encounter for this C_T is evident. It should also be noted that if ice accretion is allowed to take place only up to the 85% radial location, the horsepower increment needed to maintain a C_T of 0.004 is reduced to approximately 12%. This illustrates the sensitivity of the rotor tip region in the degradation of the helicopter performance in an icing encounter, and stresses the need to quantify the mechanism of both ice accretion and shedding.

Other thrust coefficient levels, such as C_T of 0.006 and 0.008, for the assumed natural icing encounter are shown in Fig. 4. As can be seen, the trend is the same for all values of C_T for required horsepower. However, it may be noted that the slope of the required horsepower is linear with icing time and is approximately the same for all three thrust coefficients examined.

Summary

The analytical model which provides theoretical values of performance degradation due to rime ice accretion on helicopter rotor blades in hover yields values that are representative of those experienced in actual flight. However, further test/theory correlation is needed to determine the validity of the present approach. Emphasis is also being placed on selected forward flight conditions through performance data available at the Boeing Vertol Company. In this case, the rotor disk is divided into 24 15-deg sectors and the cyclic variation in local Mach number and angle of attack is assessed for each specified flight condition, i.e., forward flight, hover, etc. The results of analytical predictions, such as torque rise as a function of time, are currently being examined in detail to determine the validity of such an approach in modifying only the steady component of the forces experienced by the rotor blade in a natural icing condition.

Acknowledgment

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A Method for Predicting Wing Response to Buffet Loads

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Introduction

BUFFETING is the aeroelastic response of aircraft structures to aerodynamic excitation arising from random loading due to flow separations on the wing. It is almost always encountered when the aircraft approaches the limiting usable lift at high speeds. The maneuvering capability of aircrafts is thus usually limited by buffet or buffet-related unsteady phenomena which induce the pilot to restrict the maneuver. Methods for predicting the buffet intensity as the aircraft penetrates into the buffet regime are extremely useful and much needed in aircraft design.

The random nature of the loading on the wing due to flow separations requires statistical theory in predicting the

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dynamic response of the wing during buffeting. All previous studies consider only the structural response, without including the effects of the unsteady aerodynamics around the vibrating wing. In this Note, the method of Ref. 1 is extended to include the coupling between the structural vibration and aerodynamics of the oscillating wing under random loading. The panel method described in Ref. 1, which is a modification of a method proposed by Schweiker and Davis[2] for the study of the response of shells to aerodynamic noise, is most suitable for adapting the doublet-lattice technique[3,4] in computing the unsteady aerodynamics. As an example, the response of a F-4E wing is studied using structural and experimental data given by Mullans and Lemley.[5] From rigid model wind tunnel pressure fluctuation measurements, the acceleration of the aircraft wing at flight conditions is computed and comparison is made with flight test data. The results should be viewed with caution, since at buffeting conditions the use of inviscid aerodynamics to predict the unsteady air loads due to wing vibration is only a rough approximation. However, an indication of the importance of the interaction between aerodynamics and structural dynamics can be obtained.

Analysis

The wing surface is divided into elements or panels arranged in strips parallel to the freestream, and all surface edges, fold lines, and hinge lines lie on the panel boundaries (Fig. 1). In each panel the steady flow is represented by a horseshoe vortex having the bound vortex of the horseshoe system lying along the quarter-chord line. Superimposed on the bound vortex are acceleration potential doublets of uniform strength. The control point is located at the three-quarter chord point.

The external loading on the wing due to buffeting causes vibration of the wing, which, in turn, generates unsteady aerodynamic forces on the wing surface. Reference 6 gives a detailed description of the method used to compute the response spectrum, with only a brief outline of the theory given herein. The displacement of the wing can be expressed in terms of a set of normal coordinates $q_\alpha(t)$ as

$$z(x,y,t) = \sum_\alpha \phi_\alpha(x,y) q_\alpha(t) \tag{1}$$

where $\phi_\alpha(x,y)$ is the mode shape function of the αth mode, and $z(x,y,t)$ and $q_\alpha(t)$ are nondimensionalized with respect to the average chord, $\bar{c}$. From the dynamic equation governing the response of $q_\alpha(t)$ to a load $\ell_\alpha(t)$, the spectral response $Q_\alpha(\omega)$ can be written as

$$Q_\alpha(\omega) = K_\alpha H_\alpha(\omega)[L^A_\alpha(\omega) + L^D_\alpha(\omega)] \tag{2}$$

where

$$K_\alpha = \rho U^2 \bar{c}/2M_\alpha \omega^2_{n_\alpha} \tag{3}$$

and

$$H_\alpha(\omega) = [1-(\omega/\omega_{n_\alpha})^2 + i2\zeta_\alpha \omega/\omega_{n_\alpha}]^{-1} \tag{4}$$

In addition, U is the freestream velocity; M_α, ζ_α, and ω_{n_α} are the generalized mass, structural damping ratio, and natural frequency of the αth mode, respectively, and L^D_α and L^A_α are the Fourier spectrum of the generalized input load and aerodynamic forces, respectively. The power spectral density of the random load can be written as

$$S_{L^D_\alpha}(\omega) = \sum_{k=1}^{K} (\bar{\phi}^k_\alpha)^2 \iint_{A_k} S_{\Delta C^D_p}(x_1,y_1,x_2,y_2,\omega)\, dA_k\, dA_k \tag{5}$$

where A_k is the nondimensional area of the kth panel, $\bar{\phi}^k_\alpha$ an average mode shape for the kth panel, and $S_{\Delta C^D_p}(x_1,y_1,x_2,y_2,\omega)$ the pressure coefficient cross-power spectral density. For the aerodynamic load, the power spectral density is

$$S_{L^A_\alpha}(\omega) = \sum_k \sum_\beta (\bar{\phi}^k_\alpha)^2 \Delta Cp^k_\beta(\omega) \Delta Cp^{*k}_\beta(\omega) S_{Q_\beta}(\omega) A^2_k) \tag{6}$$

where $S_{Q_\beta}(\omega)$ is the power spectrum of $q_\beta(t)$ and the asterisk denotes the complex conjugate. $\Delta Cp^k_\beta(\omega)$ is the pressure

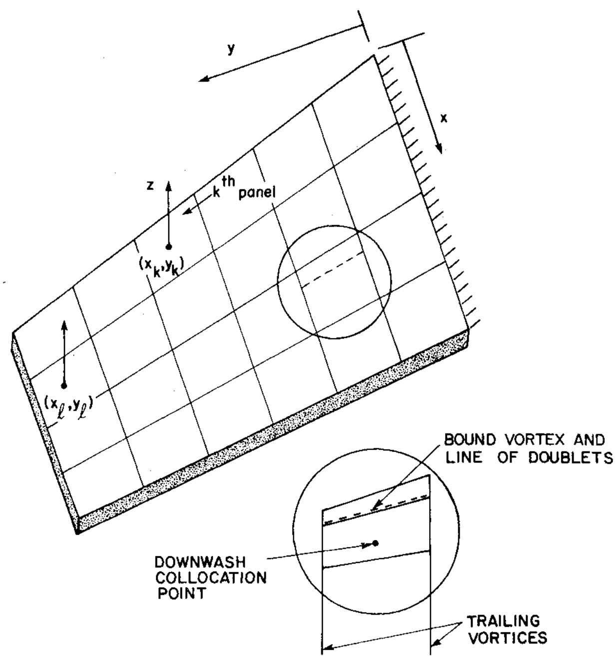

Fig. 1 Representation of wing surface by panels.

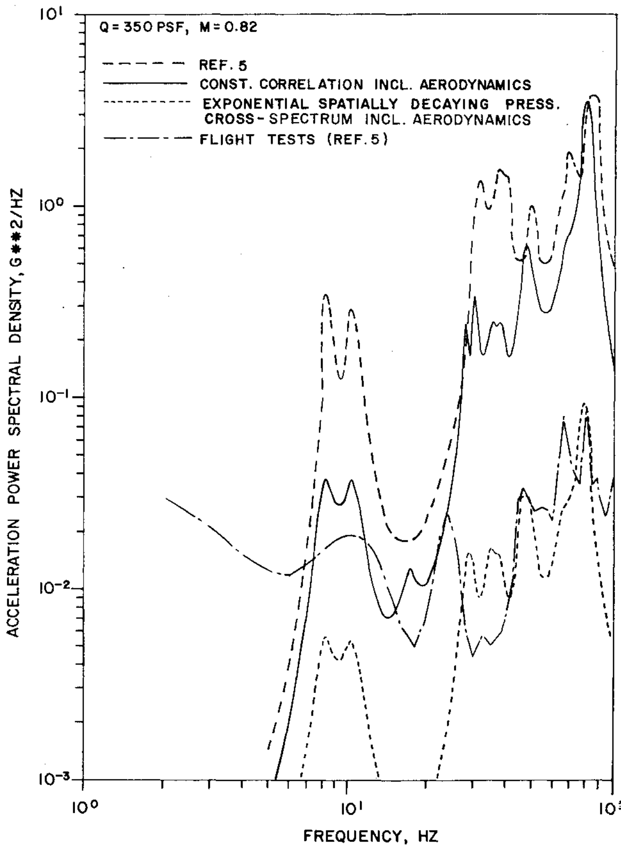

Fig. 2 Comparison of response acceleration power spectral density between theoretical predictions and flight tests.

coefficient at the kth panel for the β mode, which is calculated from the doublet-lattice method using the following equation

$$\frac{\partial \bar{\phi}_\beta^k}{\partial x} + ik\bar{\phi}_\beta^k = \sum_s D_{ks}\Delta Cp_\beta^s(\omega) \tag{7}$$

where D_{ks} is the normal wash factor and k the reduced frequency. From Eqs. (2), (5), and (6), the power spectrum $S_{Q_\alpha}(\omega)$ is given as

$$\begin{aligned} S_{Q_\alpha}(\omega)\{1 - H_\alpha(\omega)E_{\alpha\alpha}(\omega) - H_\alpha^*(\omega)E_{\alpha\alpha}^*(\omega) \\ + |H_\alpha(\omega)|^2 |E_{\alpha\alpha}(\omega)|^2\} \\ + |H_\alpha(\omega)|^2 \sum_{\substack{\beta \\ \beta\neq\alpha}} |E_{\beta\alpha}(\omega)|^2 S_{Q_\beta}(\omega) \\ = |H_\alpha(\omega)|^2 K_\alpha^2 S_{L_\alpha^D}(\omega) \end{aligned} \tag{8}$$

where

$$E_{\alpha\beta}(\omega) = K_\alpha \sum_k \bar{\phi}_\alpha^k \Delta Cp_\beta^k(\omega) A_k \tag{9}$$

Knowing $S_{Q_\alpha}(\omega)$, the acceleration spectra can be determined using Eq. (1) by differentiating with respect to time twice.

Example

As an example, the response of a F-4E wing is studied using structural and experimental wind tunnel rigid wing pressure fluctuation measurements given by Mullans and Lemley.[5] A total of 18 panels are used in the computations and the first 10 symmetric and antisymmetric bending modes are considered. Figure 2 shows a comparison between predictions and flight test data for the acceleration power spectral density measured at 84% semispan and 26% chord at $M = 0.82$ and a dynamic pressure $Q = 350$ psf. Two representations of the input random load power spectrum are used. The first assumes a constant correlation model where the pressure spectrum at any point in a panel and the pressure cross spectrum between two points in the same panel are both taken to be equal to the pressure spectrum at the panel center.[2] The second assumes an exponential spatially decaying form for the pressure cross spectrum using spearated flow decay coefficient data from Coe et al.[7] In all of the computations a value of 0.05 is used for the structural damping ratio. Using the constant correlation assumption, the comparisons with results from Ref. 5, which does not include the unsteady aerodynamic forces due to the wing vibration, show that unsteady air loads induced by the vibrating wing are quite important in the low-frequency range of the spectrum. However, at higher frequencies, they become less significant. Comparisons with flight test results show that the constant correlation assumption overestimates the response at the higher frequencies, but a great improvement in the predicitions is obtained by using the exponential spatially decaying form for the pressure cross-power spectrum. At low frequencies, due to inaccuracies in evaluating the decay coefficients obtained by extrapolating data[7] measured at much higher frequencies, it is found that predictions using the constant correlation assumption are in better agreement with flight test data than those obtained from the spatially decaying pressure cross-power spectrum representation.

Conclusions

A method for predicting the response of a wing to buffet loads has been presented. The unsteady aerodynamic forces generated by the vibrating wing are included in the analysis. It is shown that unsteady aerodynamic forces have a significant effect on the response predictions at low frequencies, while at higher frequencies they become less important.

In modeling the input load, the constant correlation assumption gives reasonable results at low frequencies, but at higher frequencies, the exponential spatially decaying form of the cross-power spectrum for the fluctuating pressures gives results which are in much better agreement with flight test data.

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Airfoil Probe for Angle-of-Attack Measurement

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Introduction

CURRENT methods of sensing angle of attack[1] include use of vanes that line up with local flow, tapped hemispherical probes from which pressure differential measurements are obtained and translated into angle-of-attack and sideslip-angle information, and servo-driven null-

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