Engineering Notes

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A Correlation of Vortex Noise Data from Helicopter Main Rotors

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THE present and future use of helicopters as flexible air transportation in and out of highly restricted heavily populated areas is threatened by the community noise problems associated with these vehicles. In the next generation of vehicles, noise requirements may play a major role in the design process.[1] It is important to have in the early stages of the design process an estimate for the effect of a given design change upon both the noise and DOC of the vehicle. This Note presents a correlation of existing data on main rotor vortex noise which is, at the present state-of-the-art, an irreducible effect of operating the main rotor. Although basic research into the effect of airfoil cross-section and tip planform shape on the noise from main rotors may eventually lead to noise reductions below this curve, an examination of the present data can also serve as a base to compare later improvements.

For the typical helicopter, aerodynamic noise is produced by the main rotor, the tail rotor, and the engine. Current efforts in design should succeed in reducing the noise from the tail rotor and engine to a point where the dominant source of noise is the main rotor.

Aerodynamic noise from main rotors is usually grouped into three classifications: rotational noise, vortex noise, and blade slap (see, for example, Refs. 2 and 3). Rotational noise can be defined as the noise a main rotor would produce in an invisid fluid, including all harmonic orders of unsteady potential flow airloads. Vortex noise is considered to be the additional noise radiated by operation in a fluid of slight viscosity, air, due to the turbulent flow on the blade sections and in the rotor plane (boundary-layer separation, vortex shedding, and the operation of airfoils in a turbulent wake). Blade slap is a characteristic impulsive sound which occurs when strong interaction occurs between a blade and a trailing vortex or when a blade tip experiences strong compressibility effects. Obviously, there is a smooth transition from rotational noise into the blade slap condition, but the distinction is usually made. When blade slap occurs it dominates all other noise sources.

Assuming that the helicopter operates in a flight condition which avoids blade slap, an important source of noise in the frequency range of interest for community noise is vortex noise. Most of the data on noise from main rotors presented in this Note were obtained from whirl tower or hover tests. Under these conditions, vortex noise would dominate the spectrum at high frequencies even though, in high-speed forward flight, rotational noise might become more important.

We will discuss a model for vortex noise which indicates how and why the various physical parameters appear in the data correlation. We will then present the resulting collapse of over-all vortex sound power data in terms of the gross properties of the vehicle and rotor rpm.

Model for Vortex Noise

Vortex noise is caused by the random fluctuations in lift on the helicopter blades due to turbulent flow separation and vortex shedding. The phenomenon was so named because of its similarity to the vortex sound from rotating cylindrical rods studied by Yudin.[4] Considering the sections of a long circular rod to be incoherent radiators, Yudin derived a scaling law to predict the sound power;

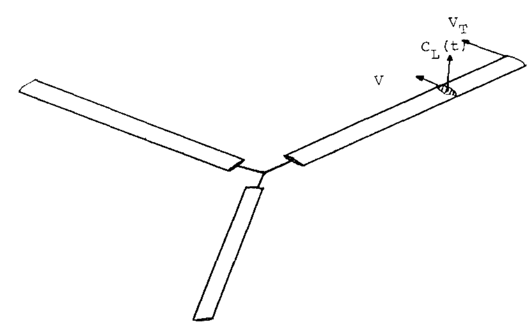
$$\Pi = \text{const}(\rho/a^3)C_D{}^2V_T{}^6Dl \tag{1}$$

where C_D is the steady-state drag coefficient, V_T the tip speed, a the speed of sound, and D and l the diameter and length of the cylinder. (This formula is specialized for the case where the frequency of vortex shedding is not a function of Reynolds number.) This formula was derived from the following considerations basic to the theory of aerodynamic noise: 1) The unsteady random lift fluctuations are coherent only over a short section of the span. 2) The acoustic power radiated by a fluctuating force of frequency $f \sim Hz$ is[5]

$$\Pi = (f^2F^2/\rho a^3)(\pi/3) \tag{2}$$

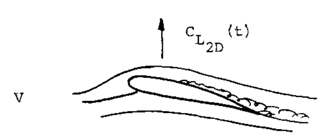
3) The average fluctuating lift per unit length due to vortex shedding is proportional to the steady drag force

$$L_{\text{rms}} = \tfrac{1}{2}\rho V^2 C_{L\text{rms}} D \sim \tfrac{1}{2}\rho V^2 \bar{C}_D \cdot D \tag{3}$$



a) Correlation area for random unsteady lift fluctuations

b) Locally two-dimensional unsteady flow



c) Spectrum of unsteady lift fluctuations

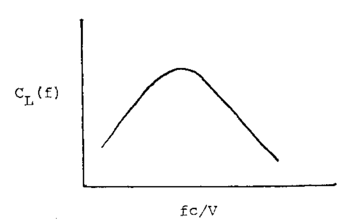


Fig. 1 Model for vortex noise from main rotors.

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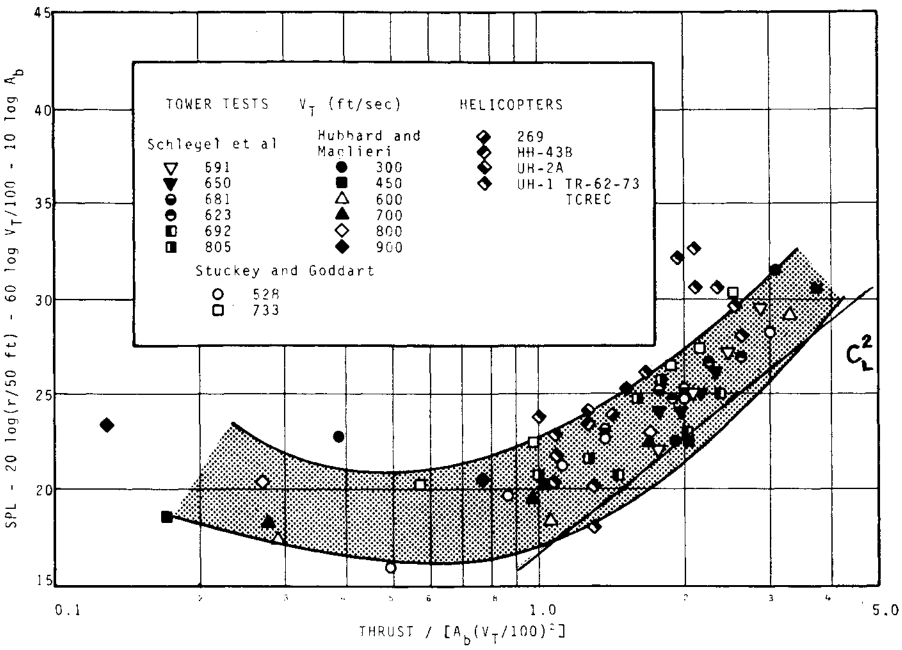

Fig. 2 Whirl tower and helicopter vortex noise data.

4) The frequencies of the fluctuating forces of aerodynamic origin are inversely proportional to the diameter D and directly proportional to the velocity V

$$f \sim V/D \tag{4}$$

For a rotating rod or a helicopter blade, the frequencies and velocities vary linearly with distance from the hub. The effect of this on the magnitude of the sound power is included in the constant of proportionality; the effect on the frequencies is to broaden the spectrum as compared with a truly two-dimensional flow.

To apply this formula to the helicopter rotor as sketched in Fig. 1, we must consider airfoil sections rather than circular cylinders. This adds two additional parameters, angle of attack and airfoil shape (although most rotor blades currently use the NACA 0012 airfoil section).

To predict rotor vortex noise, we would need the level and spectrum of the unsteady lift coefficient on the blades as well as the spanwise correlation length. It seems reasonable to assume that the details of the unsteady flow on the blades are a function of the local steady-state lift coefficient and that the frequency content is a function of the nondimensional parameter fc/V where c is blade chord, taken as constant.

For these assumptions, Eq. (1) for a helicopter rotor becomes

$$\Pi = \text{const } (\rho/a^3)V_T{}^6\, cl[G(C_L)]^2 n \tag{5}$$

where $G(C_L)$ is some function of the steady-state lift coefficient on the rotor blades, and n is the number of blades. We can define a mean blade lift coefficient

$$C_L = \text{thrust}/\tfrac{1}{2}\rho V_T{}^2 A_b \tag{6}$$

where $A_b = lcn$ the area of the blades. Equations (5) and (6) combine to give an equation for the power from a rotor

$$\Pi = V_T{}^6 A_b[\bar{G}(\text{thrust}/V_T{}^2 A_b)]^2 \tag{7}$$

assuming constant values for ρ and a. Rather than assume a form for $\bar{G}(C_L)$ as was done by Schlegel et al.,[3] $\bar{G}(C_L) \sim C_L$, or attempt to fit it with a power law $\bar{G}(C_L) \sim C_L{}^m$ as was done by Stuckey and Goddard,[6] we will determine $\bar{G}(C_L)$ from the experimental data by plotting

$$(\Pi/V_T{}^6)A_b \text{ vs thrust}/V_T{}^2 A_b$$

This allows a collapse of the data over a wider range of lift coefficients from low values, where the blades are operating in their own wake, to high values, where stall begins.

The units used in the data correlation are SPL, db re 0.0002 μb, thrust, lbs, ft^2, and V_T, ft/sec.

Correlation of Main Rotor Vortex Noise

A correlation of existing data on vortex noise from helicopter main rotors in hover was made. The data were taken from whirl tower tests and actual helicopters in hover. For the various flight tests, data from the 0° direction were taken to minimize the effect of the tail rotor. The data have been corrected to a distance of 50 ft. Directivity effects have been ignored.

The sound pressure level, corrected to 50 ft, was normalized by the sixth power of the tip velocity V_T and the blade area A_b. The results have been plotted vs a quantity which is proportional to the blade lift coefficient, thrust/$A_b(V_T/100)^2$. The results are shown in Fig. 2. When scaled in this manner, the experimental data collapse to a universal curve within about 5 db. This amount of scatter was present in the extensive whirl tower tests of Hubbard and Maglieri[7] for a single rotor, so the collapse for many different rotors and vehicles is felt to be quite acceptable. Also shown is the $C_L{}^2$ curve of Schlegel et al.,[3] which for typical operating conditions is a reasonable fit to the data.

The data show the effect of operating the blades in a highly loaded condition; the noise increases faster than $C_L{}^2$. They also show the effect of operating at very low lift coefficients where the blades are operating in their own wake.

The results of the data correlation, which is based on a quasi-two-dimensional model of vortex noise, should not be interpreted to mean that details of the unsteady viscous flow at the tips are not important. All of these rotors had square blade tips. Other tip plan form shapes are currently being investigated for the reduction of vortex noise.

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A Comparison between Transonic Wind-Tunnel and Full-Scale Store Separation Characteristics

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Nomenclature

a = speed of sound
C_F = static aerodynamic force coefficient
g = acceleration of gravity
l = length
m = mass
M = Mach number
p = static pressure
V = velocity
V_e = ejection velocity
W = weight
γ = ratio of specific heats of air
λ = scale factor, l_f/l_m
ρ = air density

Subscripts

f = full scale
m = model scale

Introduction

STORE separation characteristics from the parent aircraft can best be determined in a wind tunnel either by utilizing a trajectory-following mechanism which successively iterates the forces and moments on the store in the aircraft flow field and computes new locations for the store,[1] or by actual release of scaled stores from the aircraft model.[2] Black[3] discusses some of the advantages and disadvantages of each method. For example, it would be extremely difficult to simulate a rocket launch by the actual release method, whereas a release condition that created very large angular excursions would be nearly impossible for a trajectory follower to duplicate. The intent of this Note is to outline the two scaling methods usually employed in wind-tunnel free-flight store separation testing, to describe the anomalies and limitations of each, and to show the resulting good correlation between wind-tunnel and full-scale test results.

Scaling Laws[4–9]

In any reduced-scale wind-tunnel test involving store separation, it must be assumed that the flowfield around the aircraft is scaled and that the force field acting on the model is directly scalable to full scale without corrections for Reynolds number or Mach number. In addition, the weights and moments of inertia of the store models must be properly scaled. According to incompressible theory, the two parameters to be considered are Froude number V^2/lg and the ratio of store density to air density, $m/\rho l^3$. This implies that full-scale performance may be simulated in the wind tunnel by setting

$$V_m = \frac{1}{(\lambda)^{1/2}} V_f; \quad \left(\frac{W}{\rho}\right)_m = \frac{1}{\lambda^3}\left(\frac{W}{\rho}\right)_f$$

and

$$\left(\frac{I}{\rho}\right)_m = \frac{1}{\lambda^5}\left(\frac{I}{\rho}\right)_f$$

Considering the significant scaling effect on velocity, a direct application of this method to the compressible case does not appear reasonable. Simulation of Mach number in addition to Froude number and density ratio imposes a completely impractical restriction upon the testing procedure. Theoretically, retention of the three parameters can be accomplished by at least two methods. One method involves scaling the velocity of sound in the test medium as follows: $a_m = 1/(\lambda)^{1/2}a_f$. This cannot be achieved in a conventional wind tunnel because of the resulting low stagnation temperature required. The second method is to impose upon the model an artificial gravitational field[10] equal to the scale factor times the normal gravitational field, that is, $g' = \lambda g$, or to accelerate the airplane model away from the store at the same rate.[11] For typical scale factors of 10 to 20, this is usually impractical.

It is apparent then that some theoretical anomalies must exist in the scaling aspect of practical experimentation. One approach, which has shown promise in the case of ejected releases, is to retain density ratio and duplicate the freestream Mach number, that is,

$$(W/\rho)_m = (1/\lambda^3)(W/\rho)_f; \quad (I/\rho)_m = (1/\lambda^5)(I/\rho)_f$$

and

$$M_m = M_f$$

This has been arbitrarily called "light" scaling. Indications are that, with sufficient ejection, the correct pitch oscillation and an approximate trajectory will be obtained in the immediate neighborhood of the aircraft. Ejections causing a store vertical velocity at release of about 30 fps are considered sufficient.

Another method, used for example by Rainey,[2] places most emphasis on the trajectory of the center of gravity. Assuming a simulated flowfield in the vicinity of the scaled carrier, it is theorized that a scaled trajectory will be obtained, in the normal gravitational field, if the pitch oscillation at release is small. Scaling factors considered are freestream Mach number and the ratio of static aerodynamic force to gravity force, that is,

$$M_m = M_f \text{ and } \left(\frac{C_F(\gamma/2)pM^2l^2}{mg}\right)_m = \left(\frac{C_F(\gamma/2)pM^2l^2}{mg}\right)_f$$

It can then be shown that model weight and inertia are established by the relationships

$$(W/p)_m = (1/\lambda^2)(W/p)_f \text{ and } (I/p)_m = (1/\lambda^4)(I/p)_f$$

This has been arbitrarily called "heavy" scaling.‡ Analysis

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‡ Note that in "heavy" scaling, static pressure p is the appropriate parameter rather than density ρ.