AIAA Journal
Vol. 42, No. 8, August 2004

Assessment of the Contribution of Panel Vibration to Airframe Noise

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Aerodynamic noise sources not associated with propulsion are termed "airframe noise." As well-known sources such as flow separation, landing gear vortex shedding, and flap-edge noise are further reduced, the noise sources from the clean airframe will need to be better understood. Although direct radiation from boundary layers is generally agreed to be an insignificant source, the boundary-layer surface pressures can drive the panels to excitation levels that may produce significant airframe noise. An estimate of airframe noise due to structural vibration is presented. This analysis shows that panel vibration driven by turbulent boundary-layer fluctuations can produce airframe noise levels comparable to those measured on aerodynamically clean airframe configurations. The analysis contains a number of engineering approximations, which are discussed along with their implications for either over- or underestimating the acoustic radiation results. Because the analysis shows levels comparable to measured data rather than dramatically lower levels, it is concluded that structural vibration may be a potentially important source of airframe noise and deserves further study.

Nomenclature

a, b	=	panel dimensions
c_0	=	speed of sound in air
k	=	wave number
$\bar{k}$	=	constant
M	=	generalized mass
q	=	generalized coordinate
q_∞	=	dynamic pressure
R_1	=	resistive impedance
r, θ, ϕ	=	spherical coordinates
r_1	=	radial distance
S_p	=	panel area
t	=	time
t_p	=	panel thickness
U_∞	=	flow speed
W_p	=	power associated with a panel
w	=	panel displacement
x, y	=	spatial coordinates
δ	=	boundary-layer thickness
δ^*	=	boundary-layer displacement thickness
ζ_n	=	modal damping
ξ	=	correlation length
ρ_p	=	panel density
ρ_0	=	density of air
Φ_p	=	power spectral density
Ψ	=	mode shape
ω	=	angular frequency
ω_n	=	modal natural frequency

Superscripts

-	=	root mean square
$\sim$	=	dimensionless quantity

Introduction

THE contribution to far-field aircraft noise from nonpropulsive sources is termed airframe noise. Unsteady flow phenomena, especially separated flows on flaps, slats, and landing gear, are considered primary sources of airframe noise. It has been recognized that structural vibration excited by these flow phenomena, as well as by engine vibration and turbulent boundary-layer flow, may also be a significant source of airframe noise. However, very little research has actually been conducted in this area.

The airframe noise contribution of the vibrating structure will become even more important as airplanes are designed to be more fuel efficient and, thus, lighter weight and more flexible. Furthermore, it can be expected that the sources of airframe noise that are currently dominant will be dramatically reduced in the future. If overall airframe noise is to be reduced by 12–16 dB over the next 20 years, while simultaneously making the aircraft lighter and more fuel efficient, the role of structural vibration must be clearly understood.

This paper describes a new estimate of the level of airframe noise due to structural vibration. The unsteady aerodynamic pressures cause structural vibration, potentially leading to significant amplification. Effectively, the elastic airframe structure acts as a "sounding board" to enhance the aerodynamic sources. This mechanism, which involves the flexibility and resonant behavior of the structure itself, has received much less attention and is much less well understood than the purely aerodynamic noise sources.

Historical Background

From the past to the present much work has been done on airframe noise. In 1991, Crighton[1] reviewed the current understanding of airframe noise as represented by experiment (at model and at full scale), by theoretical modeling and by empirical correlation models. Aside from aerodynamic noise sources associated with deflected trailing-edge flaps, undercarriage gear elements, gear wheel wells, and so forth, he concluded that "it seems worthwhile to stress vibration as a probably significant source of airframe noise, particularly with large flap deflections and with gear down . . . and to note the *broadband* increase that can apparently occur in surface pressure spectra when a panel vibrates predominantly in one or two discrete modes" (Ref. 1, p. 427). In Ref. 2, Hardin suggested that panel vibration may be the source of a high-frequency peak observed in the noise spectrum of the 747, CV-990, and Jetstar aircraft because it was considerably higher than was expected due to other noise mechanisms and was rather insensitive to flight speed. A similar unexplained peak is evident in the experiments of Ref. 3 for several aircraft. These peaks are not associated with flap deflection and gear deployment, which tend to produce increases at lower frequencies. The peak is not narrowband. In addition, Ref. 4 shows an increase

Received 28 July 2003; revision received 22 February 2004; accepted for publication 26 February 2004.

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in surface pressure that is believed to be caused by panel vibration. The data are presented for dimensionless groupings and show that the elevated level is substantial and broadband, albeit centered on a particular frequency.

In 1988, Howe[5] published an analysis of mechanisms by which sound is produced during low-Mach-number turbulent boundary-layer excitation of a finite panel in a thin plate. He concluded that the principal source of radiated noise was diffraction at the joints, where the panel was attached to the supporting side plates. This analysis was done for fluid-loaded plates and was applied to steel plates in water. It was also assumed that the acoustic wavelength was smaller than the panel dimension.

In more recent work, Bayliss et al.[6] and McGreevy et al.[7] have considered multiple panels, rather than single clamped or simply supported plates. In Ref. 7, the authors consider a two-panel system interacting with jet noise at a Mach number of 0.9, whereas in Ref. 6 the authors consider a four-panel system at a Mach number of 0.65. Results of a numerical study are presented in both cases. In Ref. 7, the authors conclude that the effect of the panels is to act as a narrow filter, converting the broadband forcing (due to jet instabilities) into radiation concentrated in narrow spectral bands. Bayliss et al.[6] studied the effects of forward motion on jet noise, panel vibration, and radiation and concluded that there was a reduction in sound in downstream directions and an increase in sound in upstream directions.

Graham[8] has developed a fairly simple model for predicting noise levels transmitted into the interior of an aircraft cabin due to boundary-layer excitation based on the radiation from a single flat elastic plate. In Ref. 9, this model is further refined to include the effects of cabin trim panel dissipative layers. Maury et al.[10] have used a similar model of sound transmitted through an elastic panel for the application of active noise control of interior aircraft cabin noise.

In a recent paper, Lilley[11] discusses the major components of airframe noise, including some predictions and comparison with experiment. A current review of airframe noise prediction methods is given in this paper. An earlier study of acoustically important airframe noise components from traditional sources, including prediction procedures for each component, can be found in Ref. 12.

Technical Discussion

The results of the present paper suggest that panel vibration driven by turbulent boundary-layer fluctuations, as shown in Fig. 1, can produce airframe noise levels comparable to those measured on aerodynamically “clean” airframe configurations (flaps, slats, and retracted landing gear). Although direct radiation from boundary layers is generally agreed to be insignificant, the boundary-layer surface pressures can drive the panels to excitation levels that produce significant airframe noise.

The analysis considers the dynamic response of panels given the level and spatial correlation of the aerodynamic excitation and models the radiation impedance and directivity of panels on the fuselage and wing. A sample calculation of radiated noise is compared to published airframe noise experimental data. This sample calculation demonstrates the credibility of the hypothesis that structural vibration is a significant source of airframe noise, because the estimated levels are in the vicinity of sound pressure levels for a “clean” airplane. Therefore, reduction in radiated noise from the panels will result in overall reductions in airframe noise.

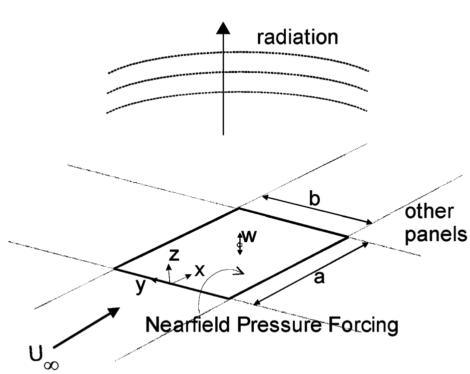

Fig. 1 Panel excited by a turbulent boundary layer.

Panel Response to Uniform Forcing

Modeling a single panel as a plate (Fig. 1) whose structural response is dominated by a single mode, as in Ref. 13, the panel displacement w is given by

$$w = q(t)\psi(x, y) \tag{1}$$

where $q(t)$ is a generalized coordinate. The mode shape $\psi(x, y)$ will be approximated for a clamped rectangular or shallow curved plate of length a and width b as

$$\psi(x, y) = [1 - \cos(2\pi x/a)][1 - \cos(2\pi y/b)] \tag{2a}$$

$$\psi(x, y) = \sin\pi x[1 - \cos(2\pi x/a)]\sin\pi y[1 - \cos(2\pi y/b)] \tag{2b}$$

or

$$\psi(x, y) = \sin\pi x[1 - \cos(2\pi x/a)][1 - \cos(2\pi y/b)] \tag{2c}$$

The first two mode shapes are sketched along one spatial coordinate in Fig. 2, and the third mode shape is just a combination of the other two.

Assuming the plate is excited by a uniform spatial pressure distribution, as in Ref. 14, it is determined that

$$\bar{q}^2 = \left(\frac{\pi}{4}\right)\left[\frac{\Phi_p(\omega_n)}{M^2\omega_n^3\zeta_n}\right]\left[\iint \psi\,\mathrm{d}x\,\mathrm{d}y\right]^2 \tag{3}$$

where $M \equiv \rho_p t_p \iint \psi^2\,\mathrm{d}x\,\mathrm{d}y$ is the generalized mass; $\Phi_p(\omega)$ is the power spectral density of the near-field pressure at frequency $\omega = \omega_n$; ω_n is the modal natural frequency; ζ_n equals modal damping; and ρ_p and t_p are the density and thickness of the panel, respectively. In this model, it is assumed that the panel is lightly damped and the response is dominated by the panel resonant response near $\omega = \omega_n$. Effects of spatial correlation over the plate area will be accounted for as a correction to this model.

Correction for Correlation Effects

According to Dowell,[13] this simple model based on a spatially uniform pressure distribution can be corrected for spatial correlation by the factor

$$\frac{4\xi_1\xi_2 \iint \psi^2\,\mathrm{d}x\,\mathrm{d}y}{\left(\iint \psi\,\mathrm{d}x\,\mathrm{d}y\right)^2} \tag{4}$$

where ξ_1 and ξ_2 are correlation lengths along the plate directions, as shown in Fig. 3. From Willmarth,[15] these correlation lengths scale with the boundary-layer displacement thickness δ^* and for a constant instantaneous spatial correlation coefficient of 0.1, $\xi_1 \approx 3\delta^*$ and $\xi_2 \approx 8\delta^*$ (see Fig. 8 in Ref. 15). Equation (3) is then modified to correct for the spatial correlation, which was not assumed in the simple model of Miles.[14] [For the parameters chosen in this paper, representative of two realistic airplanes, this correction factor resulted in a 10-dB reduction in sound pressure level (SPL), which is included in the estimation presented.]

Using Eqs. (3) and (4), and the definition of generalized mass, the average mean-square velocity over the panel is

$$\bar{\dot{q}}^2 = -\frac{\pi\Phi_p(\omega_n)\xi_1\xi_2}{\omega_n\zeta_n(\rho_p t_p)^2 \iint \psi(x, y)^2\,\mathrm{d}x\,\mathrm{d}y} \tag{5}$$

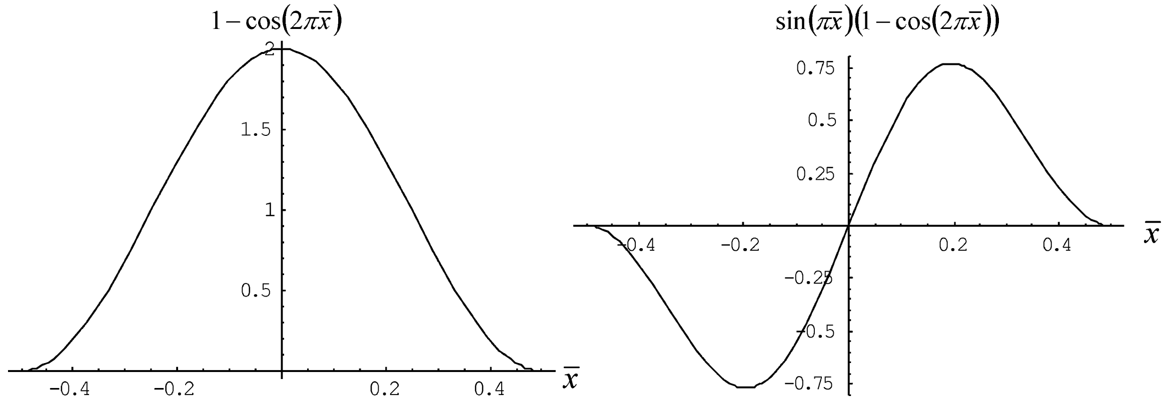

Fig. 2 Assumed mode shapes.

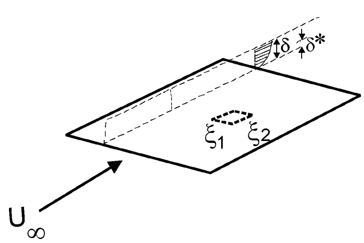

Fig. 3 Correlation correction factor parameters.

Acoustic Radiation from a Panel

By analogy to the radiation from a planar piston, as described by Morse and Ingard,[16] the acoustic power radiated from a single vibrating panel is described by

$$W_p = [R_1(\omega_n)S_p]\langle\bar{\dot{w}}^2\rangle = [R_1(\omega_n)S_p]\frac{\pi\,\Phi_p(\omega_n)\xi_1\xi_2}{\omega_n\zeta_n(\rho_p t_p)^2} \tag{6}$$

where R_1 is the real part of the impedance function associated with the vibrating panel and S_p is the panel area. Unlike the planar piston, the velocity over the surface of the panel is not uniform; therefore, the spatial average of mean-square velocity $\langle\bar{\dot{w}}^2\rangle$ is used along with the impedance to compute the power. To obtain R_1, the total far-field power radiated from the vibrating panel is determined by assuming the panel consists of a distribution of baffled acoustic monopoles, whose strengths vary with the mode shape. The contribution of each infinitesimal radiator is integrated over the surface area of the panel and the far-field pressure and intensity are obtained. The total power is then determined by an integration of the far-field intensity over a hemisphere in spherical coordinates. This analysis is outlined as follows.

The pressure from a monopole, whose volumetric flow is Q, is given by

$$p = \frac{i\rho_0 c_0 k Q e^{i(\omega t - kr)}}{2\pi r} \tag{7}$$

The differential pressure from an infinitesimal area on the panel surface to a point out in the field is

$$dp = \frac{i\rho_0 c_0 k \dot{w}\, dS\, e^{i(\omega t - kr_1)}}{2\pi r_1} \tag{8}$$

where r_1 is the distance from the incremental area to the point of evaluation. Expressing r_1 in terms of a spherical coordinate system centered on the panel and making the approximation that the point of evaluation is far away from the panel, the integral over the surface area becomes

$$\begin{aligned} p &= \int_{-a/2}^{a/2}\int_{-b/2}^{b/2}\frac{-\rho_0 c_0^2 k^2}{2\pi r} q_0\psi(x_0, y_0) \\ &\quad\times e^{i\omega t}e^{-ikr}e^{ikx_0\sin\theta\cos\phi}e^{iky_0\sin\theta\sin\phi}\,\mathrm{d}x_0\,\mathrm{d}y_0 \\ &= \frac{-\rho_0 c_0^2 k^2 abq_0}{2\pi r}D(ka, kb, \theta\phi) \end{aligned} \tag{9}$$

where r, θ, and ϕ locate the point of evaluation, and x_0, y_0 are points on the surface of the panel with displacement equal to $q_0\Psi(x_0, y_0)e^{i\omega t}$. The variable q_0 is the generalized coordinate associated with modal amplitude. The directivity function D, written in terms of dimensionless variables ($\tilde{x} = x/a$, $\tilde{y} = y/b$) is defined as

$$D(ka, kb, \theta, \phi) \equiv \int_{-\frac{1}{2}}^{\frac{1}{2}}\int_{-\frac{1}{2}}^{\frac{1}{2}}\psi(\tilde{x}_0, \tilde{y}_0)e^{ika\tilde{x}_0\sin\theta\cos\phi}e^{ikb\tilde{y}_0\sin\theta\sin\phi}\,\mathrm{d}\tilde{x}_0\,\mathrm{d}\tilde{y}_0 \tag{10}$$

The mean-square pressure is then given by

$$\bar{p}^2 = \frac{|p|^2}{2} = \frac{\left(\rho_0 c_0^2 k^2 abq_0\right)^2}{2(2\pi r)^2}|D(ka, kb, \theta, \phi)|^2 \tag{11}$$

and the far-field intensity is $I = \bar{p}^2/\rho_0 c_0$, where $\bar{p}$ is the notation for root-mean-square pressure. The total power radiated from the vibrating panel is found by integrating the acoustic intensity over a hemisphere of radius r, namely,

$$\begin{aligned} W &= \int_0^{\pi/2}\int_0^{2\pi}\frac{\bar{p}^2}{\rho_0 c_0}(r\sin\theta\,\mathrm{d}\phi)r\,\mathrm{d}\theta \\ &= \int_0^{\pi/2}\int_0^{2\pi}\frac{\rho_0 c_0}{2}\frac{(c_0k^2q_0)^2}{(2\pi)^2}(ab)^2|D(ka, kb, \theta, \phi)|^2\sin\theta\,\mathrm{d}\phi\,\mathrm{d}\theta \end{aligned} \tag{12}$$

Equating this total power with the power given by Eq. (5), and solving for the normalized impedance function $\tilde{R}_1$, yields

$$\tilde{R}_1 = \frac{R_1}{\rho_0 c_0} = \frac{1}{(2\pi)^2}(ka)(kb)\int_0^{\pi/2}\int_0^{2\pi} \times |D|^2\sin\theta\,\mathrm{d}\phi\,\mathrm{d}\theta \Bigg/ \int_{-\frac{1}{2}}^{\frac{1}{2}}\int_{-\frac{1}{2}}^{\frac{1}{2}}\psi(\tilde{x}_0, \tilde{y}_0)^2\,\mathrm{d}\tilde{x}_0\,\mathrm{d}\tilde{y}_0 \tag{13}$$

In Fig. 4, this function is plotted vs kd [where $d = \sqrt{(ab)}$] for different aspect ratios (a/b) and for the different mode shapes given in Eq. (2). Also shown in Fig. 4 is a dashed line which is an approximate curve, where

$$\tilde{R}_1 = \frac{(kd)^2}{2\pi + (kd)^2} \tag{14}$$

For the purposes of calculating the radiated power from Eq. (6) the approximate formula preceding for $\tilde{R}_1$ will be used. Except at very low frequencies, this approximation will result in an underestimate of the power radiated for the first mode shape given by Eq. (2a). For the other modes, it overestimates at low frequency and

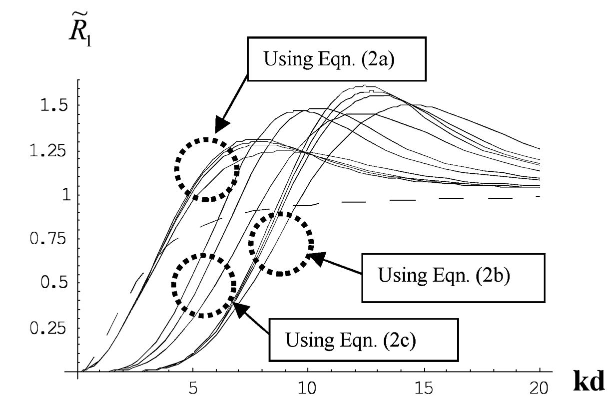

Fig. 4 Real part of panel impedance function for different aspect ratio plates and different mode shapes (solid lines) and approximate curve (dashed line). Aspect ratios represented are 1.0, 1.5, 0.5, and 0.75 for each of the three mode types.

underestimates at higher frequencies. In reality, the modes given by Eq. (2a) will be predominate at low-to-mid frequencies and the other modes will be excited from mid-to-high frequencies. Thus, the simple approximation of Eq. (14) seems overall to be a reasonable approximation of the aggregate effects of various panel modes. In fact, it likely underestimates the radiated power when considered across the entire frequency range.

A point of clarification regarding Fig. 4 may be in order. At intermediate frequencies the radiation impedance overshoots unity by a value that is higher than might be expected, especially for the higher modes. This behavior is a consequence of the way that the impedance is normalized in terms of the spatial average temporal mean-square velocity.

Power from the Airframe Structure

Assuming that a certain fraction γ of all panels have resonance frequencies that are near one another (within the same bandwidth) but not identical, the power of the total structure equals the sum of the individual powers. Therefore,

$$W_{\mathrm{TOTAL}} = \gamma W_p(Area_{\mathrm{TOTAL}}/S_p) \tag{15}$$

where W_p is calculated using Eq. (6) and the radiation impedance given by Eq. (14).

To determine γ, the first question that needs to be addressed is the number of resonances that a given panel has in a frequency band. Because of the dispersive nature of waves on a plate, the number of modes below a given frequency is roughly linearly proportional to that frequency; thus, the frequency density of modes is approximately a constant. This means that the number of participating modes in a proportional band will double with each octave increase in frequency.

Within each octave band it is assumed that each aircraft panel has about the same number of resonances, but because of physical variations among the many types of panels on an aircraft, it is further assumed that the resonances of all the panels are equally distributed over each octave band. Thus one-third of these resonances will occur in each of the constituent one-third-octave bands.

Although increasing frequency increases the number of resonances, the higher panel modes are less well coupled to the acoustic field. Figure 4 shows the radiation efficiency of mode shapes that, at least at low frequency, are monopole, dipole, and quadrupole radiators. It is seen that, at low frequencies, the higher modes are less efficient radiators; however, at higher frequencies when the panels are no longer acoustically compact, all three mode types are efficient radiators. The approximate function for radiation loading, Eq. (14), roughly matches the monopole radiation at low frequencies, passes through the middle of higher mode radiators at midfrequencies, and falls somewhat below the high-frequency results. For instance, at $kd = 2.5$, it agrees well with the monopole mode radiation; at $kd = 5.0$ (an octave higher), it gives higher levels than the dipole modes, but the number of participating modes has doubled, and so it seems reasonable to assume that the poor acoustic coupling of higher modes maybe be offset by the number of participating modes.

To be conservative, it has been assumed that the cumulative effects of any one panel in an octave band can be represented by a single resonance with the radiation loading of Eq. (14). Thus, for the whole airplane in each one-third-octave band, one-third of the panels contribute at this level. In fact, the panel resonance curves shown by Graham[8] show an average of roughly six resonance peaks for a single panel over almost four contiguous octave bands. Therefore, even if the average coupling is worse than Eq. (14), this estimate seems very conservative; that is, it will underestimate the radiated noise.

The SPL will be related to the sound power level (PWL) by the relationship[17]

$$\mathrm{SPL} = \mathrm{PWL} - 10\log\left(S_{\mathrm{radiated}}\big/r^2\right) \tag{16}$$

where r is the distance in meters. At low frequencies, the panel sources radiate omnidirectionally, so the mean square pressure at the ground is approximately

$$\bar{p}^2_{\mathrm{ground}} = \frac{\rho_0 c_0 W_{\mathrm{TOTAL}}}{4\pi H^2} \tag{17}$$

where H is the altitude and $S_{\mathrm{radiated}} = 4\pi H^2 = S_{\mathrm{radiated\ sphere}}$. At high frequencies, the radiation is directional and depends on the overall shape of the radiator, and the radiation from the fuselage and wing is approximated as cylindrical and $S_{\mathrm{radiated}} = S_{\mathrm{radiated\ cylindrical}} = 2\pi HL$, where L is the fuselage length or wing span. The behavior of the panels as radiators is sketched in Fig. 5 and extended to an aircraft in Fig. 6.

For the panel radiation model, it is therefore assumed that at low frequencies the sound is radiated spherically, and at high frequencies the sound is radiated cylindrically from the aircraft. A weighting function is used to transition from one regime to the other, based on the behavior of the panel impedance function normalized by $\rho_0 c_0$ [Eqn. (14)]. This function approaches zero at low frequency and unity at high frequency. Therefore, the radiated area in Eq. (16) becomes

$$S_{\mathrm{radiated}} = S_{\substack{\mathrm{radiated}\\ \mathrm{sphere}}}(1 - \tilde{R}_1) + S_{\substack{\mathrm{radiated}\\ \mathrm{cylinder}}}\tilde{R}_1 \tag{18}$$

This radiation behavior is consistent with observations of experimental data where the SPLs have not fallen off on the sidelines as would be expected from a dipole (characteristic of edge noise). The measured directivity is always more omnidirectional as noted by

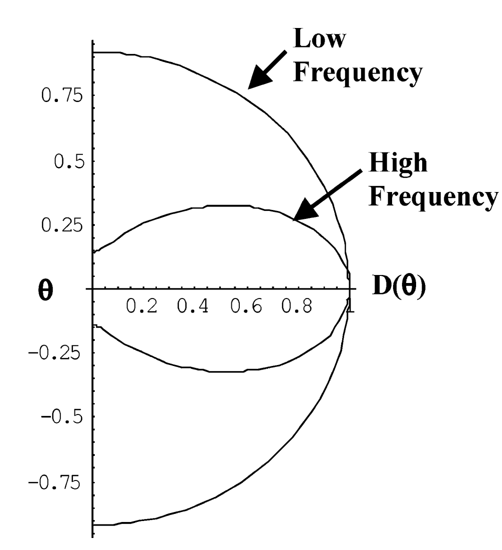

Fig. 5 Directivity pattern of radiating panel for low and high frequencies.

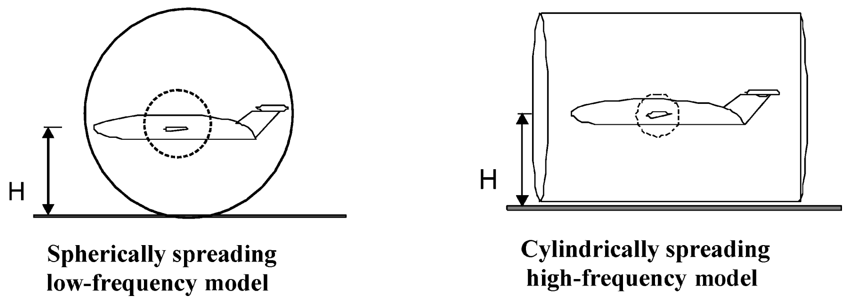

Fig. 6 Two types of radiation from airplane fuselage.

Fethney.[3] This more omnidirectional behavior is consistent with the panel directivity argument that appears in this paper and supports the idea that panel vibration may be an important source of airframe noise in the clean configuration.

Realistic Parameters

Physical dimensions and material properties were chosen for the plate model that correspond to a panel on a typical airplane. The plate density is $\rho_p = 2700$ kg/m^3 (aluminum), the thickness of the plate is $t_p = 0.00127$ m (about 0.05 in.), the structural damping is $\zeta_n = 0.02$, and the plate area is $S_p = 0.2787$ m^2 (3 ft^2, corresponding to a 24×18 in. plate).

The density and sound speed in air were $\rho_0 = 1.21$ kg/m^3 and $c_0 = 343$ m/s, respectively. The flow speed U_∞ was taken as 160 kn (82.311 m/s = 270 ft/s), which is the flow speed of the experimental airframe noise data with corresponding dynamic pressure $q_\infty = 4098.9$ Pa (about 0.04 atm). For this flow, the Reynolds number was 4.48×10^7. An average boundary-layer thickness was estimated to be $\delta = 0.086$ m (approximately 3 in.), with the corresponding boundary-layer displacement thickness $\delta^* = 0.0107$ m (about 0.4 in.).

Airframe noise data were found for two midsize airplanes, a VC-10 and a CV-990. These two airplanes are similar in dimension. The CV-990 has a wing area of 2433 ft^2, a mean chord of 19 ft, and a wing length of 1570 in. The VC-10 has a wing area of 2851 ft^2, a mean chord of 19 ft, and a wing length of 1752 in. In estimating the radiated noise from these airplanes, average quantities were used for wing area and so forth. The fuselage was estimated to be approximately 130 ft long and 10 ft in diameter. These dimensions were used to calculate the total surface area of the airplane, from which the ratio A_{total}/S_p (an estimate of the number of panels) was deduced.

A turbulent pressure spectrum $\Phi_p(\omega)$ was calculated based on the Houbolt approximate formula, given in Ref. 18 [as Eq. (16)]. From Ref. 18,

$$\Phi_p(\omega) = \frac{q_\infty^2 \delta^*}{U_\infty} \frac{(\sigma/q_\infty)^2 (2/\pi)\bar{k}}{1 + \bar{k}^2 \left(\omega\delta^* / U_\infty\right)^2} \tag{19}$$

According to Ref. 18, the most appropriate algorithm for predicting incompressible power spectral density consists of choosing $\bar{k}$ equal to unity and $\sigma/q_\infty = 0.006$. This predicted power spectral density agrees well with the experimental data in Fig. 5 of Ref. 18. However, it should be noted that in Ref. 19, in which a turbulent boundary layer excited a vehicle side window, the turbulent boundary-layer data (experimental) were approximately 10 dB higher than results predicted by the Houbolt formula [Eq. (19)], for the parameters used in the study. Because Eq. (19) was used without any surface roughness and irregularity, the panel forcing is probably underestimated.

Using the Houbolt formula for the power spectral density (PSD) and Eqs. (6) and (24), the power radiated from a single panel is

$$W_p = \frac{(kd)^2}{2\pi + (kd)^2} \frac{\pi \xi_1 \xi_2}{\omega_n \zeta_n (\rho_p t_p)^2} \frac{q_\infty^2 \delta^*}{U_\infty} \frac{(\sigma/q_\infty)^2 (2/\pi)\bar{k}}{1 + \bar{k}^2 \left(\omega\delta^* / U_\infty\right)^2} \tag{20}$$

Results

Two sources of experimental data were used to compare to the approximate model. The data were compiled from separate sources. Fethney[3] presents data for four aircraft, ranging from a slender delta to a long-range transport. From this data set, the experimental results for the VC-10 were extracted. The VC-10 is described as a long-range passenger transport with 110 seats powered by four rear-mounted engines. Experimental data were presented for both the clean and dirty configurations. "Clean" is defined as having the undercarriage and the high-lift devices retracted, whereas "dirty" is defined as having the undercarriage lowered and high-lift devices deployed. The second source of experimental data came from the study by Putnam et al.[20] In this study flight tests were conducted on four aircraft, an AeroCommander, JetStar, CV-990, and B-747. The aircraft were also tested in both clean and dirty (landing) configurations. The CV-990 aircraft is most similar to the VC-10, and therefore these data were extracted from the study of Ref. 3 for purposes herein, namely, comparison of airframe noise levels with estimated noise emissions from panel vibration.

As an estimate of the amount of noise radiated from the structural vibration of the airplane, sample calculations were made based on a simple isolated vibrating panel, as described earlier. The power from the individual panel was calculated, assuming spatially uniform pressure across the panel, which was then corrected to account for spatial correlation of the pressure field. Using realistic physical parameters for the test aircraft, and the assumption that one-third of the individual panels will be resonant within a one-third-octave bandwidth of each other (but are uncorrelated), the power of the individual panels was calculated from Eq. (15) with $\gamma = \frac{1}{3}$. The approximate number of panels covering the aircraft (wing and fuselage surface areas), which is used in Eq. (15), is simply the ratio of total surface area of the aircraft to the area of a single panel. This power was then assumed to spread spherically at low frequencies and cylindrically at high frequencies. The SPLs at the ground, 500 ft below the modeled aircraft, were computed from the power and its assumed directivity pattern. No account was taken for level changes due to ground reflection. Also, note that other clean configuration sources of airframe noise,[20] such as trailing-edge noise, have not been included.

In Fig. 7, the SPLs predicted from this model, which included only panel radiation from a turbulent boundary-layer excitation, are plotted as a function of frequency. Also on the graph are the corresponding data from Refs. 3 and 20 for the clean and dirty configurations of the two test airplanes. From this figure, it is apparent that panel vibration noise is at a level that could be considered significant for airframe noise. There appears to be a midfrequency peak in the estimated noise level. This is particularly important because noise reduction may be targeted in frequency ranges around 1000 Hz.

Additionally, future aircraft such as the Airbus A380 have significantly more radiating surface area, and therefore may result in substantially more airframe noise due to panel radiation. In fact, using the current specifications for the Airbus A380 in the current estimation method yields an increase of approximately 5 dB over the theoretical curve shown.

Gibbs et al.[21] performed active noise control experiments on a set of panels excited by turbulent boundary-layer excitation in a

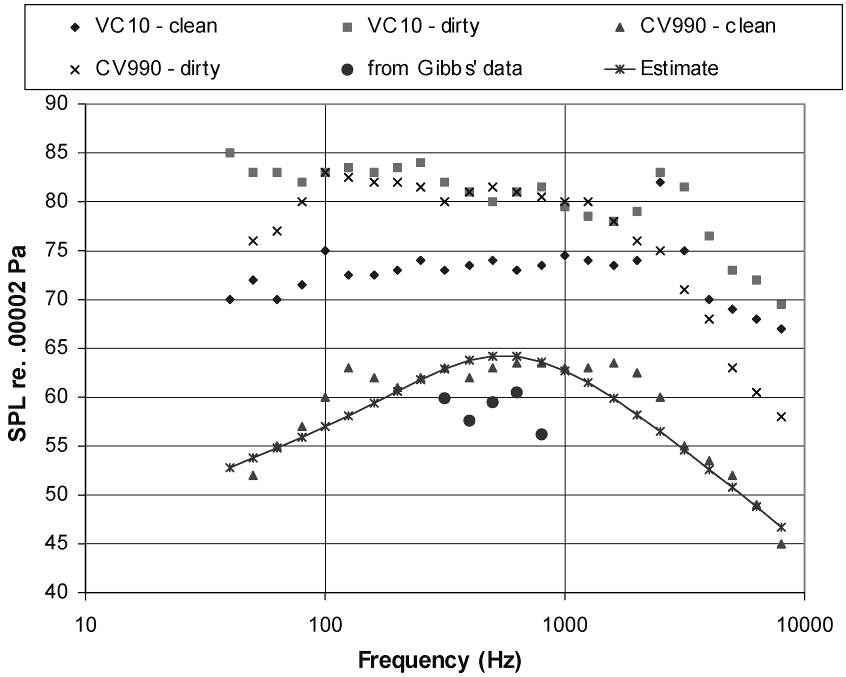

Fig. 7 Comparison of the airframe noise due to panel radiation based on an approximate model and experimental airframe noise data for clean and dirty configurations of two airplanes, and an estimate based on Gibbs data.

wind tunnel at NASA Langley Research Center. The panels were tensioned in the crossflow direction to simulate the hoop stress of actual aircraft panels on a fuselage. In Ref. 21, the power radiated from a single panel is predicted from the acceleration data of the excited panel and is presented with and without control for 1.25-Hz bands. At the author's request, these narrowband data were transmitted for further postprocessing to compare with the radiation model presented in this paper. The narrowband data were converted to one-third-octave band levels and a speed correction was applied because the Gibbs data were at Mach 0.2 and the experimental data were at Mach 0.24. The estimate of PWL on the ground follows as presented earlier, with the exception that the dimensions of the Gibbs panel were 20×10 in., changing the resulting S_p and, therefore, the number of panels radiating. Note that the levels predicted in this manner are close to those of the estimate derived in this paper.

Discussion of Results

This sample calculation demonstrates the credibility of the hypothesis that structural vibration is a significant source of airframe noise. The estimated levels are in the vicinity of SPLs for a clean airplane. The outlined prediction scheme embodies many approximations, each of which can be questioned and refined. An attempt has been made to be conservative throughout, that is, not to overestimate effects. If such an analysis had shown the effect of structural vibration to be dramatically lower (say, 10–20 dB) than clean airframe measurements, then it could be said with confidence that this effect is not an important contribution to airframe noise. However, the effect appears comparable to measured data and, because the estimate is unlikely to be high by an order of magnitude (10 dB), it appears that this effect deserves further research.

Factors Affecting Estimated Levels

Surface irregularities and roughness on a real fuselage/wing configuration would lead to higher PSD levels that would increase the noise estimate. Furthermore, when the aircraft is in landing configuration rather than in clean configuration, surface flow turbulence levels may be considerably higher due to flow separation and vortex shedding, causing a further increase in noise due to structural vibration.

It was noted earlier that the PSD for the turbulent boundary layer, that is, the forcing on the panel used in the estimate, was low when compared to a wind-tunnel experiment of flow over a vehicle side window.[19] This discrepancy between the calculated PSD based on Houbolt's formula, Eq. (19), and that shown graphically in Ref. 19 was as much as 10 dB. It is speculated that this is due to the type of turbulent boundary layer and that dirty flow may provide PSD levels closer to that of Ref. 19, whereas clean flow may be closely approximated by Eq. (19). Therefore, it is expected that the estimated curve would rise by at least a few decibels at certain frequencies due to dirty flow excitation.

On the other hand, there are other effects that would lower the estimated levels. The preceding calculation assumes that all panels have resonances that contribute in an octave frequency band. Although the panels will each have many modal resonances, perhaps not all will contribute in a given frequency band. Furthermore, as the assumed mode shapes indicate, many panel modes may not be acoustically well coupled in certain frequency ranges (e.g., low-frequency odd modes). However, as the following discussion indicates, there may be a large number of long-wavelength multipanel or overall structural modes that are very well coupled to the fluid. These modes, which would raise the estimated structural vibration noise, have also been ignored.

Other Considerations

The calculation was intended to give a "rough estimate" of the magnitude of noise attributable to structural vibration. However, the estimate does come close to experimental data for clean aircraft. In this model only the vibration of individual clamped panels has been considered rather than treating the airframe as a single, integrated, complex structure in which stringers, frames, and panels are all coupled as a single structural system. Structures of this type exhibit many forms of complex behavior, including wavelengths that extend over many subsystems (even at high frequency), stop and pass bands, and localization behavior.

It has been shown in naval applications, a field where large complicated structures are considered a single system, that there are long-wavelength flexural waves on shell structures with ribs even at high frequencies.[22] Similar phenomena will occur in aerospace structures such as fuselages, although this fact appears to have gone largely unrecognized. These long waves, called Bloch waves, are very efficient radiators of sound because their phase speed tends to be supersonic relative to the speed of sound in the fluid. Typically, they couple well to the fluid and are capable of propagating over long distances along the structure. When panels are considered in isolation, these long-wavelength vibrations are ignored. For far-field acoustic radiation from airframe noise, the contribution of the long-wavelength flexural (Bloch) waves could be extremely important, especially at higher frequencies. This phenomenon is an example of the type of effects that can occur over large scales on a complex, periodic structure.

Conclusions

A simple model has been used to estimate the contribution of panel vibration to airframe noise. The model suggests that structural vibration of flexible panels may be a significant source of airframe noise. Although many approximations are made throughout the analysis, most are thought to be on the conservative side and would result in an underestimate of the radiated power. Future research in the area of acoustic radiation from the structural vibration of fuselage panels ought to be undertaken, particularly as noise from other sources is reduced and as larger airplanes become commercially available.

Acknowledgments

The author thanks Gary Gibbs at NASA Langley Research Center for providing data files that were used here for the calculation of comparative estimates of panel radiation. The author also thanks Don Bliss at Duke University for intellectual discussions and for making suggestions that were used to improve the paper.

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