Journal of Aircraft

VOLUME 5 JULY-AUGUST 1968 NUMBER 4

Design Criterion of Panel Structure Excited by Turbulent Boundary Layer

LUCIO MAESTRELLO*
The Boeing Company, Seattle, Wash.

Using a relatively simple functional representation of the space-time correlation of the wall-pressure fluctuation, the motion of a simply supported panel and the resulting acoustic radiation can be predicted within 1 order of magnitude from the experimental results. Criterion for designing a panel for given flow conditions by this method is considered. The governing parameter is the so-called coherence distance, the distance over which a given turbulent pattern remains distinguishable. Calculations indicate that, when coherence distance is much smaller than the panel length, the response is mostly due to coincidence. From knowledge of the panel motion, the radiated sound intensity is obtained. For a panel much longer than the coherence distance, the acoustic power radiated is considerably reduced. Significant results were obtained from suggested practical methods of lessening the panel response and vibration noise level. Structural excitation by separated flow is localized on an airplane, but since it is severe it is given careful consideration.

Nomenclature

$A, k, K(M), N, \sigma, n$	=	constants
C_l	=	longitudinal wave velocity
F	=	δ^*/U
K_1, K_2	=	wave numbers
K_a	=	bending wave velocity
L_x	=	panel length
M	=	Mach number
$P(\ )$	=	turbulent power spectrum
PWL	=	total sound power level
$R(\ )$	=	correlation coefficient
U, U_s	=	freestream velocity
U_c	=	broadband convection velocity
Y	=	panel displacement
$Y(\ )$	=	panel power spectrum
$a_{m,n}$	=	plate modal damping
a,b	=	lengths of panel sides
c	=	speed of sound
f or ω	=	frequency ($\omega = 2\pi f$)
h	=	panel thickness
m,n	=	mode number
$(p^2)^{1/2}$	=	rms wall-pressure fluctuations
u	=	local velocity
u_τ	=	frictional velocity
x,y	=	coordinates
W	=	plate mass
$\Pi(\omega)$	=	normalized power-spectral density
δ^*	=	boundary-layer displacement thickness
η	=	lateral partial separation
ρ	=	density
θ	=	eddy lifetime
ξ	=	longitudinal spatial separation
σ	=	standard deviation
τ	=	time delay
τ_w	=	wall-shear stress
ϕ	=	wave form
ν	=	kinematic viscosity

Presented as Paper 67-12 at the AIAA 5th Aerospace Sciences Meeting, New York, January 23–26, 1967; submitted January 9, 1967; revision received March 4, 1968. The author wishes to express his indebtedness to M. R. Gedge, T. Reddaway, and R. P. Skoog for their assistance and criticism.

* Research Specialist, Boeing Scientific Research Laboratory.

Introduction

THE skin structure of an airplane is curved, continuous, and divided into bays of panels that have frame stiffeners around the circumferential edges and stringers along the longitudinal edges. The skin is excited to vibrate by turbulence, sound, or both. When the structural excitation is due solely to a turbulent boundary layer, high internal noise levels result; however, in general there will be no structural damage unless a severe loading within the structural resonant frequency persists.[1] When the excitation is due to engine noise,[2–5] separated flow,[6] and oscillating shock, large amplitude stress reversals result which can lead to fatigue cracks and failure. More specifically, in the case of passenger-carrying airplanes, the acoustic near field from the propulsion unit and from pseudosound[7,8] induces skin vibrations that in turn act like a transducer and radiate sound in the interior of the aircraft. In recent years, owing to the increased speed of aircraft, it has become necessary for the manufacturer to devote a great amount of effort to reducing interior noise to a comfortable level.

The interior noise level is mainly set by the skin panel vibration,[9–26] caused by the unsteady boundary layer, jet and propeller noise, compressor whine, and thrust reversal noise. This paper is concerned with the establishment of a

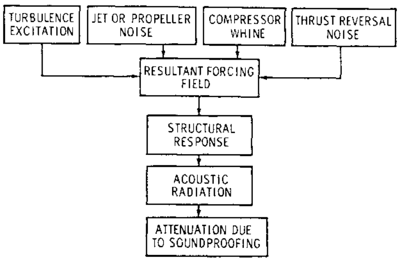


Fig. 1 Approach to establishing structural response and interior noise level of aircraft.

design criterion for a panel excited by turbulent boundary layer. The method of approach, for both aerodynamic and acoustic sources, is represented in Fig. 1.

Boundary-Layer Forcing Field

The boundary layer develops uniformly along the fuselage of a large aircraft except at the cabin region, where large disturbances are initiated as a result of an abrupt change in profile. The skin frictional drag over the aircraft fuselage is higher than that of a flat plate because of leakage of air into the boundary layer from the pressurized fuselage and because of the waviness of the skin due to pressurization. The velocity distribution in the turbulent boundary layer, on the aircraft,[27] can be represented by

$$u/u_\tau = 5.6 \log(yu_\tau/\nu) + 3.5 \tag{1}$$

The statistical properties of the turbulent boundary-layer pressure field show that the component of the pressure fluctuation in each panel mode generates small force oscillations, but the panel does not respond similarly for all frequencies and wave numbers of the turbulence: certain combinations of wave number and frequency are excited more strongly than others. When the wall pressure matches both in the wave number and in the frequency of a particular panel mode, a condition called coincidence occurs. This results in a very strong excitation of particular modes, which causes a peak response.

Model of the Wall-Pressure Covariance

The author[9–14] proposes three models of the wall-pressure field suitable to study the response between the wall pressure and the panel motion. One of the models is of nondimensional form for ranges of subsonic Mach number and boundary-layer thickness at nearly zero pressure gradient. The flow is considered semifrozen (the convection velocity is constant) and the correlation function decays for large separation and time delay. The choice is partly satisfactory since the panel spectrum responds as a narrow filter to the turbulent boundary-layer spectrum. The model is constructed from the power-spectrum density from three components of the wall pressure. The power-spectral density $P(\omega)$ and the normalized $\pi(\omega)$ power-spectral density satisfy the following relationships:

$$\int_0^\infty \Pi(\omega)d\omega = 1 \qquad \int_0^\infty P(\omega)d\omega = \overline{P^2} = [K(M)]^2\tau_w^2 \tag{2}$$

The power spectrum is plotted in Fig. 2 in a nondimensional form, and it may be described by

$$\frac{\Pi(\omega)U[K(M)]^2}{\delta^*} = \sum_{n=1}^{3} A_n e^{-k_n(\omega\delta^*/U)} \tag{3}$$

where

$$k_1 = 0.47 \qquad k_2 = 3.0 \qquad k_3 = 14.0$$
$$A_1 = 1.6 \qquad A_2 = 7.2 \qquad A_3 = 12.0$$

The function $K(M) = (P^2/\tau_w)^{1/2}$ is plotted in Fig. 3.

Experimentally, the cross-correlation of the wall-pressure fluctuations can be obtained at a constant spatial separation and a variable time delay or vice versa. The normalized model of the cross-correlation consistent with the measurements made at a constant spatial separation, as well as with the preceding power-spectrum model, is

$$R(\xi, \eta; \tau) = e^{-|\xi|/U_c\theta} \times \left\{\sum_{n=1}^{3} \frac{A_n k_n}{k_n^2 + (1/FU_c)^2[(\xi - U_c\tau)^2 + \eta^2]}\right\} \Big/ \sum_{n=1}^{3} \frac{A_n}{k_n} \tag{4}$$

where the convection velocity is defined by

$$\partial R(\xi; \tau)/\partial\tau = 0 \qquad \text{when} \qquad U_c = \xi/\tau_{\max}$$

As reported by Ffowcs-Williams[33] and Lighthill,[34] the optimum-time scale is more appropriate than the optimum-space scale for aerodynamic noise problems. This indicates an alternate convection velocity to Eq. (4), which has been adopted by most workers (see Refs. 28–32). The model of the normalized cross-correlation complying with correlation made at constant time delay and variable spatial separation is given by

$$R(\xi, \eta; \tau) = e^{-|\tau|/\theta} \times \left\{\sum_{n=1}^{3} \frac{A_n k_n}{k_n^2 + (1/FU_c)^2[(\xi - U_c\tau)^2 + \eta^2]}\right\} \Big/ \sum_{n=1}^{3} \frac{A_n}{k_n} \tag{5}$$

where the convection velocity is defined by

$$\partial R(\xi; \tau)/\partial\xi = 0 \qquad \text{when} \qquad U_c = \xi_{\max}/\tau$$

The cross-correlations Eqs. (5) and (4) can be related through Taylor's hypothesis where the space and time variations are interrelated by assuming that the time decay can be described by the spatial decay, $|\tau|/\theta \simeq |\xi|/U_c\theta$. Difficulty arises in obtaining the time Fourier transform of Eq. (5); however, comparison was made by numerical integration and the results showed close correspondence between spectra. In light of this, Eq. (5) was chosen to determine the response of the panel structure to turbulence. Further discussion on

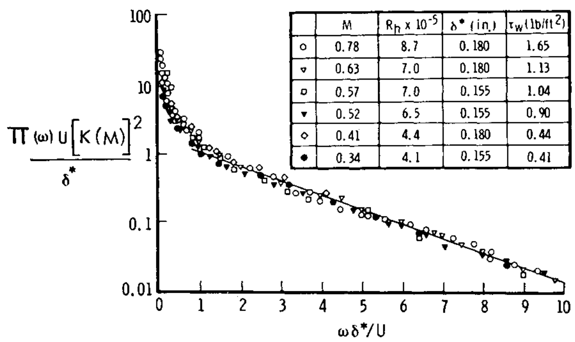


Fig. 2 Dimensionless power spectrum of the wall pressure.

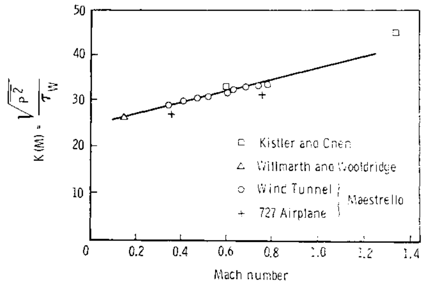


Fig. 3 Wall-pressure fluctuation.

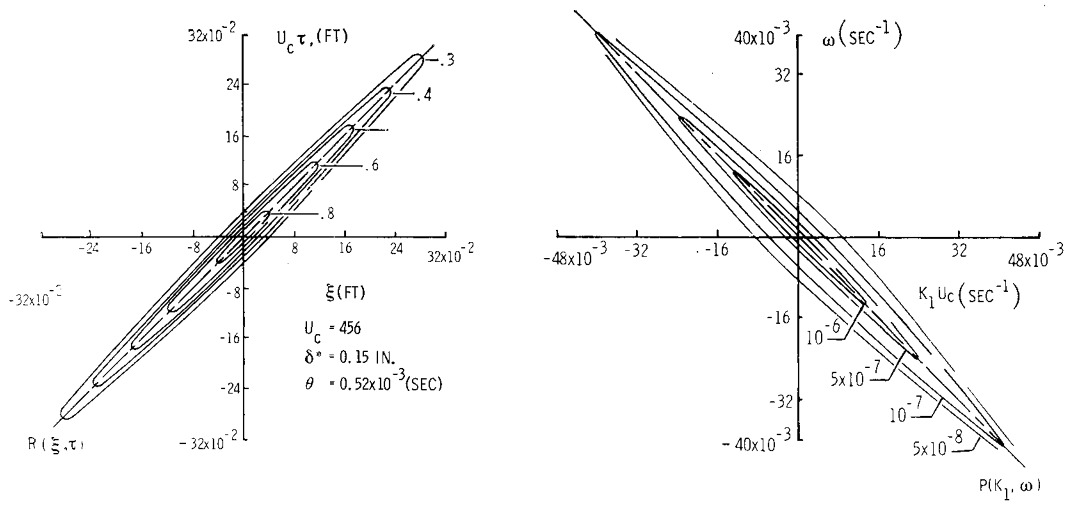


Fig. 4 Contours of constant correlation and spectral level for convected semifrozen pattern.

the cross-correlation function to be used to evaluate the panel response has been reported.[9]

An equivalent representation of Eq. (5) in terms of wave number and frequency spectrum is possible when the Fourier transform is taken first with respect to space and then to time. Using the definition

$$P(K_1,K_2;\,\omega) = \frac{1}{(2\pi)^3}\times \int_{-\infty}^{\infty}\int_{-\infty}^{\infty}\int_{-\infty}^{\infty} R(\xi;\,\eta;\,\tau)e^{-i[(K_1\xi+K_2\eta+\omega\tau)]}\,d\xi d\eta d\tau \quad (6)$$

the longitudinal normalized power-spectra density is

$$P(K_1;\,\omega) = \frac{\theta}{1+\theta^2(\omega+K_1U_c)^2}\left(\frac{FU_c}{2\pi}\right)\times \sum_{n=1}^{3} A_n e^{-|K_1|k_nFU_c}\Big/\sum_{n=1}^{3}\frac{A_n}{k_n} = [P(K_1)/\pi][\theta/1+\theta^2(\omega+K_1U_c)^2] \quad (7)$$

Equation (7) indicates that the maximum occurs when

$$U_c = -\omega/K_{1_{\max}} \quad (8)$$

corresponding to

$$\partial P(K_1;\,\omega)/\partial K_1 = 0$$

Figure 4 shows contours of constant correlation $R(\xi;\,\tau)$ and contours of constant power-spectral density $P(K_1;\,\omega)$. The correlation contour is symmetrical about the convection velocity line with 45° angle in the ξ, $U_c\tau$ plane. This indicates that $R(\xi;\,\tau) = R(-\xi;\,-\tau)$. A similar display is shown for the power-spectral density, except that $P(-\omega;\,K_1) = P(\omega;\,-K_1)$.

Structural Response to a Turbulent Boundary Layer

Since the pressure field over the panel is responsible for generating the motion of the panel, which consists of a multimode vibration, a Fourier analysis in two space dimensions ξ and η and one time dimension τ would determine the extent of the influence of the pressure field in controlling the response of the panel. This response would not be uniform, since certain combinations of wave numbers and frequencies in the component of the surface displacement are excited more than others.[9] The panel tends to respond with running wave ripples having trace speeds equal to the eddy convection speeds of the turbulent boundary layer as predicted by Corcos-Liepmann,[16] and Ribner,[17-19] and extended by Powell,[18] and Ffowcs-Williams and Lyon.[26]

Response of Simple Panels

The assessment of the vibration response of simple panels to turbulence in terms of model shape, cross-spectral density, and mean-square displacement is considered. Numerical computation of the response for simply supported panels has been previously made by el Baroudi,[15] using an idealized model from Dyer[20] with pressure covariances having a delta function frequency spectrum with exponential decay. Maestrello,[9] Jacobs and Lagerquist,[13] and Wilby[21] extend this analysis using a turbulent model for pressure covariance obtained from experimental data.

For a practical airplane structure, a panel is divided into bays of panels with a frame stiffener around the edges and stringers along opposite edges. Lin[35] described the combination of stringer-bending and stringer-twisting response modes for a large continuous section of the fuselage.

Measurements show that the response of such structures behaves as a running wave ripple, as predicted by Ribner, which favors the type of deformation corresponding to higher-order modes. Each adjacent panel becomes independent. Previous work on multiple panels[22] shows that two adjacent panels separated by stringers are uncorrelated by running waves, but are correlated for low-frequency modes corresponding mainly to those modes near the fundamental frequency of the entire bay of panels. Work on more complex structures has been carried out by Rattayya[23] and Lyamshev.[36]

Theoretical Approach

In previous work,[9,22] the author has shown that by using a relatively simple functional representation of the space-time correlation of the wall-pressure fluctuations, the motion of a simply supported panel and the resulting acoustic radiation can be predicted.

The present paper is concerned with obtaining design information for a panel excited by turbulent boundary layer. The governing parameter is the so-called coherence distance L_ξ, the length over which a given turbulent pattern remains distinguishable. It can be obtained experimentally by measuring the spatial correlation of the wall-pressure fluctuations for various delay times and integrating under the envelope of the maxima maximorum:

$$L_\xi = \int_0^\infty e^{-|\xi|/U_c\theta}\,d_\xi = U_c\theta \quad (9)$$

Using the pressure covariance given by Eq. (5), the response of the panel is

$$\overline{Y(x,y,t)Y(x',y',t')} = \frac{ab\overline{P^2}}{2\pi^2 \sum\limits_{n=1}^{3} \frac{A_n}{k_n} W^2} \sum_{m,n} \frac{\phi_{m,n}(x,y)\phi_{m,n}(x',y')}{(mn)\omega_{m,n}(a_{m,n}^2 + \omega_{m,n}^2)} \times \int_{-m\pi}^{m\pi} f(\bar{z}) \left(\int_0^{n\pi} f(\bar{y}) \times \left[\int_0^{\infty} g(\bar{x}) \sum_{n=1}^{3} \left\{ \frac{A_n k_n e^{-|(\tau-\bar{x})|/\theta}}{k_n^2 + \left(\frac{1}{FU_c}\right)^2 \left\{ \left(\left[\frac{a\bar{z}}{m\pi} - U_c\tau \right] + U_c\bar{x} \right)^2 + \left(\frac{b\bar{y}}{n\pi} \right)^2 \right\}} + \frac{A_n k_n e^{-|(\tau+\bar{x})|/\theta}}{k_n^2 + \left(\frac{1}{FU_c}\right)^2 \left\{ \left(\left[\frac{a\bar{z}}{m\pi} - U_c\tau \right] - U\bar{x} \right)^2 + \left(\frac{b\bar{y}}{n\pi} \right)^2 \right\}} \right\} d\bar{x} \right] d\bar{y} \right) d\bar{z} \quad (10)$$

where

$$f(\bar{z}) = \cos\bar{z} + (1/m\pi)(-|\bar{z}|\cos\bar{z} + \sin|\bar{z}|)$$

$$f(\bar{y}) = \cos\bar{y} + (1/n\pi)(\sin\bar{y} - \bar{y}\cos\bar{y})$$

$$g(\bar{x}) = e^{-\bar{a}_{m,n}\bar{x}}[\sin\omega_{m,n}\bar{x} + (\omega_{m,n}/a_{m,n})\cos\omega_{m,n}\bar{x}]$$

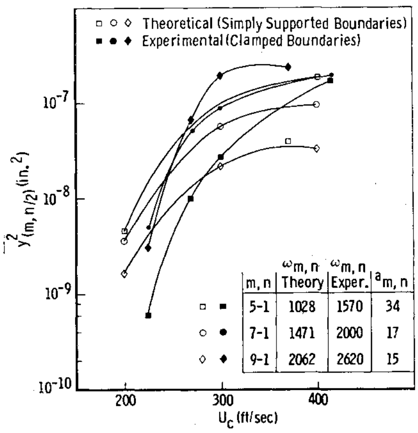
The derivation of Eq. (10) is reported in Ref. 9.

Comparison is made in Fig. 5a of the theoretical modal mean-square displacement evaluated at the center of the modes $\overline{Y^2}_{(m,n/2)}$ and that obtained experimentally for a 36- × 6.5- × 0.04-in. aluminum panel measured at the same locations at various convection velocities. The experimental panel had clamped boundaries, whereas the theory assumed simply supported boundaries. The objective of the comparison between theory and experiment is to evaluate the characteristics of the model with experiment rather than an absolute comparison. The largest error is attributed to the differences between the theoretical and experimental modal shapes, which occur because the theory gives the maximum displacement at the center of the mode, whereas experimentally the modes are not always symmetrical. This error is accentuated by the differences in boundary conditions, implying not only differences in modal shape but also in frequency. A more realistic comparison can be made using the mean-square modal displacement as an integrated space and time average rather than only a time average at the center of the mode. An additional error is due to damping measurement which has been estimated to be approximately 30%.

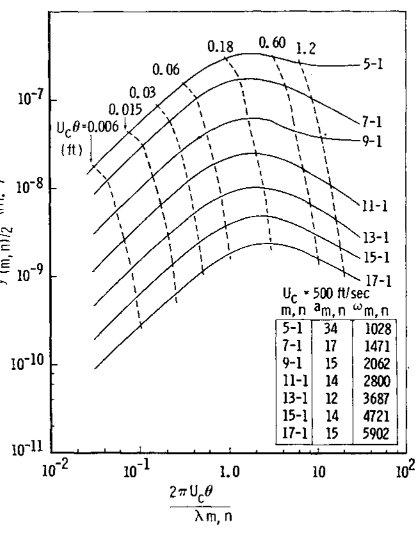
The panel response described in Eq. (10) does not include the cross-modal effect; however, presently under consideration is an evaluation to incorporate the cross-modal effect and clamped-boundary conditions. Correlation reported in a previous paper[9] indicates that the panel motion has a convective feature with the flexural wave velocity of the order of the convection velocity of the pressure field; this is coincident with the prediction of Ribner[19] and others and the measurements of el Baroudi.[15] Computations of modal mean-square displacement are made (Fig. 5b) within the range $U_c\theta \ll L_x$, and $U_c\theta > L_x$, at constant U_c and variable θ to illustrate the behavior of modal mean-square displacement in this range.

When $2\pi U_c\theta/\lambda_{m,n} \ll 1$ the modal mean-square displacement is inversely proportional to the total damping, whereas the peak of the frequency spectrum is inversely proportional to the square of the damping.[13,17,26] This is the region where the damping has the greatest effect. The modal mean-square displacement increases with $U_c\theta$, until $U_c\theta$ is equal to the inverse of the modal wave number at which point coincidence occurrs. This phenomenon would be observed at the forward portion of the aircraft where the boundary layer is thin and the coherent length small compared with the panel length.

In the region where $2\pi U_c\theta/\lambda_{m,n} > 1$ (Fig. 5b), the coherent length of the turbulence is longer than the modal wavelengths. In this case, coincidence is not possible, adjacent panels respond in phase. In this region, damping has the least influence and the modal mean-square displacement decreases with increasing coherent lengths. An example of this excitation is typical in the aft section of a large aircraft.



a)



b)

Fig. 5 Modal mean-square displacement.

Radiation of Sound by Flexible Panels

To estimate the sound power level of the radiation resulting from the vibration of panels, the modal volume displacement of an equivalent radiation efficiency must be ascertained. Theoretical predictions of the radiation efficiency and measurement of the acoustic damping have been reported by Maidanik,[37] Lyon,[38] Smith,[39] Kraichnan,[24] Nikiforov,[40] and Maestrello.[14] For a finite panel, the major source of radiation below the critical frequency $f_c = \frac{1}{2}\pi c^2/K_a C_l$ arises from the interaction of the bending wave with the discontinuity of the boundary. The present panel sizes have modes that radiate mainly below the critical frequency, where the radiation is somewhat less efficient than it is at the critical frequency or above. Below the critical frequency, the radiation from a quarter wavelength of the mode segment is cancelled by the radiation from the adjacent quarter wavelength that is out of phase. This process of cancellation is extended across the four boundaries of the panel, and the effective radiation lies between the panel edges and the nearest quarter wavelength. However, those modes in which the separation is larger than acoustic wavelengths will radiate sound like independent sources.

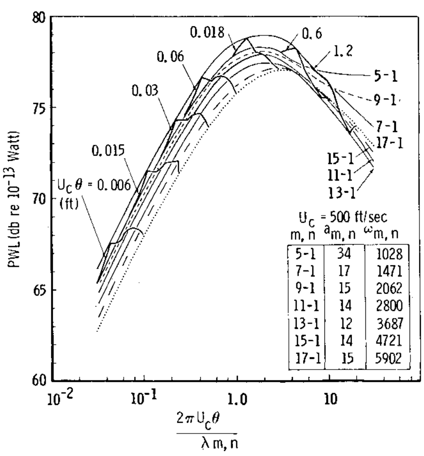
The modal acoustic power radiated in a reverberant field in terms of modal velocity is given by

$$PWL_{m,n} = \frac{N\omega^2 \rho c K_a{}^2}{4\pi}\,\overline{Y^2}\,\frac{2P_r + P_p}{P_p} \times \left[\int_0^a \int_0^b |\phi_m(x)||\phi_n(y)|dxdy\right]^2 \quad (11)$$

The value of N depends on the radiation mode, which can be classified as an edge radiation or piston radiation. In this instance piston radiation, for which $N = 4$, was assumed for the purpose of calculation. The value of N also depends on the distance between sources in relation to the acoustic wavelengths, as well as on the panel boundary. The ratio $(2P_r + P_p)/P_p$ is the contribution from stringers mounted on the panel, following Maidanik.[37]

The radiated power obtained by direct measurement and that obtained by Eq. (11), using a measured value of the mean-square displacement, are compared in Ref. 9. There is satisfactory agreement for simple panels of 0.08- and 0.04-in. thickness, as well as for panels with stringer combinations using the idealized ratio suggested by Maidanik. It is believed that, with the proper choice of damping, the total acoustic power is obtainable directly from the structural response of the panel. Then, from the knowledge of panel response computed in the previous section, the radiated sound power was obtained from Eq. (11). Assuming that the convection velocity U_c is constant, the only variable is the eddy lifetime for a given mode. The computation is shown in Fig. 6.

In the region $U_c\theta \ll \lambda_{m,n}$ the modal acoustic power is proportional to $U_c\theta$ for all odd modes, since the even modes have zero resultant volume displacement. This is valid for those modes in which the separation is less than an acoustic wavelength. The power spectra is proportional to the volume velocity. Figure 6 shows the variation of the power spectra with $U_c\theta$. The behavior is well known from the experiment[9]; when $U_c\theta \ll \lambda_{m,n}$; $\omega_{m,n}\theta \ll 1$, the spectra tend to



m, n	$a_{m,n}$	$\omega_{m,n}$
5-1	34	1028
7-1	17	1471
9-1	15	2062
11-1	14	2800
13-1	12	3687
15-1	14	4721
17-1	15	5902

Fig. 6 Sound power level.

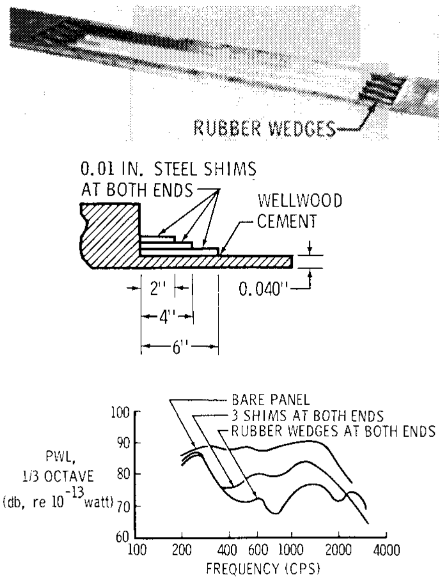


Fig. 7 Two methods of reducing the sound power level of a 36- × 6.5- × 0.04-in. panel.

peak in the higher modes, whereas for $U_c\theta > \lambda_{m,n}$; $\omega_{m,n}\theta > 1$ the peak will occur in the lowest modal vibration. In the case of airplane structure, the panel dimensions are usually the same throughout the aircraft fuselage; variations occur in the boundary-layer thickness along the fuselage, that is, $U_c\theta$ increases from $U_c\theta \ll \lambda_{m,n}$ to $U_c\theta > \lambda_{m,n}$.

From an acoustic point of view the ideal panel would be one in which $U_c\theta \ll \lambda_{m,n}$. Ribner[19] proposed that the aircraft panel be supported solely by a longitudinal stringer in order to eliminate boundary discontinuities in the direction of the running waves.

Since the modern aircraft structure is constructed to prevent the propagation of cracks, the skin thickness is not continuous. The skin gage is increased every so often by a tear stopper. This new configuration is disadvantageous from the acoustic point of view because the tear stopper induces multiple reflections.

A method has been devised to modify the boundary condition sufficient to alleviate the reflection since the skin discontinuities are not conducive to minimum radiation level. Measurements of space-time correlation indicate that the panel response is predominantly excited by running flexural waves, with reflection at every discontinuity of the skin. The transmitted waves are predominantly low frequency with wavelengths of the order of the panel size or larger, consistent with the former criterion.

Altering the panel boundary conditions is one means of reducing the radiated noise level. The best results have been obtained with rubber wedges (Fig. 7) at both ends of the panel in the direction of the running wave. Another method is gluing three layers of different lengths of 0.01-in.-thick steel membranes to the upstream and downstream edges of the panel. In effect, the membranes increase the panel stiffness gradually toward the boundaries, minimizing the reverberation buildup caused by sharp discontinuities. These methods have been considered of practical importance in reducing noise inside an aircraft fuselage.

The noise levels obtained using these devices are compared with those of the bare panel in Fig. 7. Significant noise reduction is obtained using these devices, which add approximately 15% to the weight of the bare panel. Therefore, in the case of an aircraft, the alternative is to maintain the same weight by reducing the weight of the inner blanket. This results in lower sound power level with no change in weight of the aircraft. Smith[39] and Nikiforov[40] have indicated the sensitivity in the radiation level and efficiency due to the change in boundary conditions.

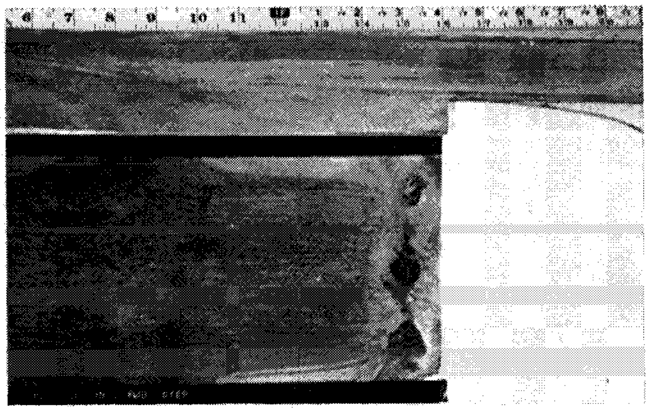


Fig. 8 Flow visualization by means of lamp black—side and bottom views.

Measurements of the Response of a Panel Excited by Separated Flow and the Resulting Radiation Field

It is also necessary to discuss the separated flowfield because of the severity of the panel response and the noise radiated from it. Kistler[41] first reported measurements of the wall-pressure fluctuation in a separated flow. Separated flow arises locally on a flight vehicle whenever there is a change in shape; however, flight vehicles are usually well streamlined, and the regions of separated flow are localized.

A simple experiment was devised using a 1-in.-high backward-facing step in a closed channel. Measurements of the velocity profile and static-pressure distribution downstream of the step substantiated the well-known results that the pressure distribution is independent of the step height, as well as of the condition of the incoming flow, and that the reattachment point occurs approximately at 8-step height. Figure 8 shows the flow visualized by means of lamp black. The vortex generated by the step leaves unperturbed regions at the base of the step, which indicates that the trap vortex in the corner is not of uniform strength.

The distribution of the wall-pressure fluctuation reaches a maximum at the reattachment point (Fig. 9) consistant with the results reported by Mohsen.[42] Published data by Tani[43] and others on the measurement of Reynolds stresses in similar flow indicate an increase in Reynolds stresses up to reattachment. It is hypothesized that at this point the velocity gradient at the wall becomes zero; that is, the wall-shear stresses are zero at reattachment point. Downstream of the reattachment point the wall-shear stresses will be positive, corresponding to the region where the wall-pressure fluctuation decreases to the value of boundary-layer pressure. That the reattachment point in the boundary layer oscillates back and forth may be caused by the ratio of the rate at which the energy from the mixing is dissipated to that supplied by the mean flow.

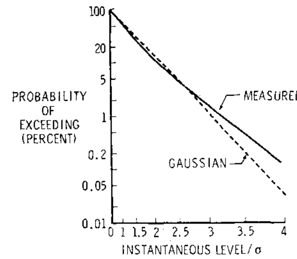


Fig. 10 Probability measured at the center of a 12- × 6.5- × 0.04-in. panel (U_s = 370 fps).

A 12- × 6.5- × 0.04-in. panel was clamped to one wall of the duct in line with the separated flow induced by the step. The spread of the separated flow region covered two thirds of the panel area. Both the response and radiation field were measured. The response of the panel exhibited non-Gaussian-type distribution similar to the one described by Lambert and Smits.[44] The resultant acoustic radiation in the panel excitation was considerably higher than that in the same panel excited by boundary-layer flow. The distribution is shown in Fig. 10.

A listener in the reverberation room hears a series of sound pulses coming from the panel, which has an amplitude in excess of the usual boundary-layer noise excitation. The maximum amplitude occurs at lower frequency, and for the present panel, occurs near the fundamental frequency. The displacement spectrum is shown in Fig. 11. It is evident that the spectrum obtained from the panel response excited by separated flow is the highest. The comparison of the mean-square modal displacement shows that

$$\overline{Y^2}_{m,n\,\mathrm{separated}} \equiv 10\overline{Y^2}_{m,n\,b.\ \mathrm{layer}} \qquad M < 0.4$$

$$\overline{Y^2}_{m,n\,\mathrm{separated}} \equiv 100\overline{Y^2}_{m,n\,b.\ \mathrm{layer}} \qquad M > 0.4 \qquad M < 0.7$$

The total acoustic power measured in a reverberation room for the two cases is shown in Fig. 12. The power radiated by the panel excited by separated flow is also higher than the boundary-layer flow at the same speed (proportional to an equal value of displacement). For turbulent boundary layer at higher subsonic Mach number, an effective mismatch will occur and the total acoustic power PWL will be proportional to a lower power of the Mach number, whereas for separated flow the PWL will be proportional to a higher power of the Mach number.

Boundary-layer excitation:

$$PWL \propto M^5 h^{-1.6} \qquad M < 0.4$$

$$PWL \propto M^{2.3} h^{-1.0} \qquad M > 0.4$$

$$\text{to } M^{1.5} h^{-1.0} \qquad M > 0.4$$

Separated flow excitation:

$$PWL \propto M^5 \qquad M < 0.4$$

$$PWL \propto M^8 \qquad 0.7 > M > 0.4$$

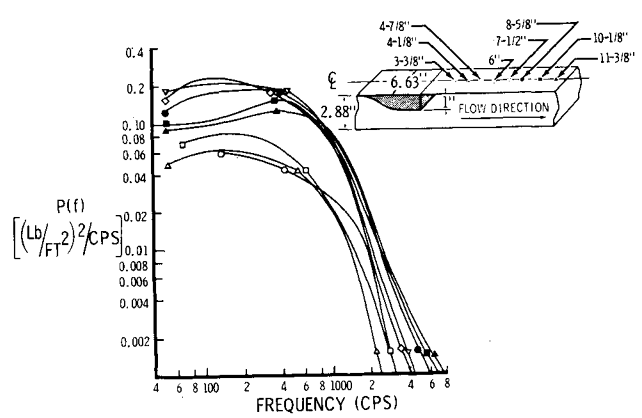


Fig. 9 Power spectra of the wall-pressure fluctuation downstream of the step.

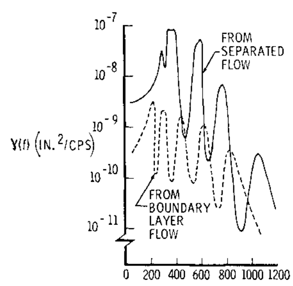


Fig. 11 Displacement spectra for a 12- × 6.5- × 0.04-in. panel ($U = U_s$ = 370 fps).

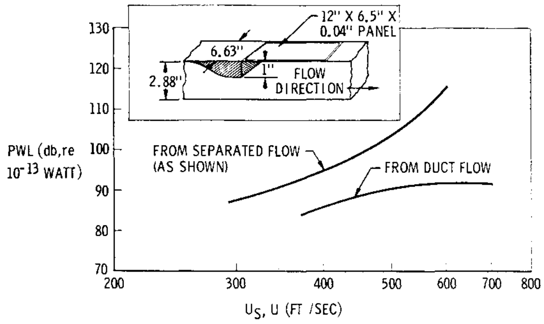


Fig. 12 Acoustic power radicated by separated and non-separated boundary-layer flow.

The present experiment suggests using a panel shorter than the mixing region of the separated zone, since running waves are not present. This modification may not reduce the sound level since a large panel is replaced by bays of small panels, all acoustically coupled, but the alternating stresses will be reduced. Since the separated region is generally localized, it may not be detrimental to increase the mass of the panel in that region. The noise attenuation for all generality will then be proportional to the added mass of the panel. A more practical method for minimizing both structural response and radiation for this type of forcing field is to utilize honeycomb structure, which has the advantage of high fundamental modal frequency. Measurements made by the author[9] are very encouraging.

There are other kinds of flow perturbation that may have considerable effect on the response of structure and acoustic radiation. For example, in an airplane, there are rolling vortices generated at the cockpit that significantly contribute to additional noise in the interior. Observation indicates that an increase in yaw angle results in an increase in sound level in the cockpit.

Conclusion

A criterion useful for the design of an aircraft panel excited by turbulent boundary layer is considered. The most favorable panel is one that is very long compared with coherence length of the turbulence. This is because the panel excitation is solely due to running waves. Since it is not generally practical to have extremely long panels on aircraft, an alternate practical method of minimizing the reflection from the boundary was found that results in a lower structural response and radiation level.

The response of the panel structure to separated flow is more severe than to the turbulent boundary layer and indicates a non-Gaussian response. A criterion is to have a panel shorter than the separation distance of the flow. In conclusion, whereas the turbulent boundary layer excites the running wave on the panel, the separated flowfield excites the panel with a series of impulses.

References

[1] Fuller, J. R., "A Theoretical Approach for the Determination of Acoustic Transmission Losses through Loaded Rectangular Skin Panels," D6-7606 TN, Feb. 1964, The Boeing Co.

[2] Powell, A., "On the Fatigue Failure of Structures Due to Vibrations Excited by Random Pressure Field," *Journal of The Acoustical Society of America*, Vol. 30, 1958.

[3] Chu, W. T., "Turbulence Measurements Relevant to Jet Noise," Rept. 119, 1966, Univ. of Toronto Institute for Aerospace Studies.

[4] Jones, I. S. F., "Model for Prediction of Jet Noise from Measurements of Fluctuating Turbulent Stresses," Rept. 5, 1967, University of Waterloo.

[5] Ribner, H. S., "Noise of Aircraft," Rev. 24, 1964, Univ. of Toronto Institute for Aerospace Studies.

[6] Speaker, W. V. and Ailman, C. M., "Spectra and Space Time Correlations of the Fluctuating Pressures at a Wall beneath a Supersonic Turbulent Boundary Layer Perturbed by Steps and Shock Waves," Rept. CR-486, 1966, NASA.

[7] Maestrello, L., Gedge, M. R., and Reddaway, A. R. F., "Pseudo Sound Field of a Jet and Its Effects on a Simple Panel (Using a Continuum and a Finite Element Method)," Doc. D1-82-0652, Sept. 1967, The Boeing Co.

[8] Trubert, M. R. P., "Response of Elastic Structure to Statistically Correlated Multiple Random Excitation," *SASA*, Vol. 35, No. 7, 1963.

[9] Maestrello, L., "Use of Turbulent Model to Calculate the Vibration and Radiation Response of a Panel with Practical Suggestions for Reducing Sound Level," *Journal of Sound and Vibration*, Vol. 5, 1967.

[10] Ludwig, G. R., "An Experimental Investigation of the Sound Generated by Thin Steel Panels Excited by Turbulent Flow (Boundary Layer Noise)," Rept. 87, 1962, Univ. of Toronto.

[11] Tack, D. H. and Lambert, R. F., "Response of Bars and Plates to Boundary-Layer Turbulence," *Journal of the Aerospace Sciences*, Vol. 29, No. 3, March 1962, pp. 311–322.

[12] el Baroudi, M. Y., "Turbulence Induced Panel Vibrations," Rept. 98, 1964, Univ. of Toronto.

[13] Jacobs, L. D. and Lagerquist, D. R., "A Finite Element Analysis of Simple Panel Response to Turbulent Boundary Layers," Doc. D6-19392, 1967, The Boeing Co.

[14] Maestrello, L., "Measurements and Analysis of the Response Field of Turbulent Boundary Layer Excited Panels," *Journal of Sound and Vibration*, Vol. 2, 1965.

[15] el Baroudi, M. Y., Ludwig, G. R., and Ribner, H. S., "An Experimental Investigation of Turbulence-Excited Panel Vibration and Noise (Boundary Layer)," Rept. 465, 1963, AGARD.

[16] Corcos, G. M. and Liepmann, H. W., "On the Transmission Through a Fuselage Wall of Boundary Layer Noise," Rept. SM 19570, 1955, Douglas Aircraft Co.

[17] Ribner, H. S., "Response of a Flexible Panel to Turbulent Flow; Running-Wave Versus Modal-Density Analysis," *Journal of the Acoustical Society of America*, Vol. 40, 1966.

[18] Powell, A., "On the Approximation to the Infinite Solution by the Method of Normal Modes for Random Vibration," *Journal of the Acoustical Society of America*, Vol. 30, Dec. 1958, pp. 1136–1150.

[19] Ribner, H. S., "Boundary Layer Induced Noise in the Interior of Aircraft," Rept. 37, 1956, Univ. of Toronto.

[20] Dyer, I., "Sound Radiated into a Closed Space from Boundary Layer Turbulence," Rept. 602, 1958, Bolt, Beranek and Newman, Inc.

[21] Wilby, J. F., "The Response of Panels to Turbulent Boundary Layer Excitation," Ph.D. thesis, 1966, Univ. of Southampton.

[22] Maestrello, L., "The Effect of Length and Thickness on the Panel Response Due to Turbulent Boundary Layer Excitation," presented at the 5th International Congress of Acoustics, Liege, Belgium, Sept. 1965.

[23] Rattayya, J. V. and Junger, M. C., "Flow Excitation of Cylindrical Shell and Associated Coincidence Effects," *Journal of the Acoustical Society of America*, Vol. 36, 1964.

[24] Kraichnan, R. H., "Noise Transmission from Boundary Layer Pressure Fluctuations," *Journal of the Acoustical Society of America*, Vol. 29, 1957.

[25] White, P. H., "Transduction of Boundary Layer Noise by Rectangular Panel," *Journal of the Acoustical Society of America*, Vol. 40, No. 6, 1966.

[26] Ffowcs-Williams, J. E. and Lyon, R. H., "The Sound Radiated from Turbulent Flows near Flexible Boundaries," Rept. 1054, Aug. 1963, Bolt, Beranek and Newman Inc.

[27] Gyorgyfalvy, D., "Effects of Pressurization on Airplane Fuselage Drag," *Journal of Aircraft*, Vol. 2, No. 6, Nov.-Dec. 1965, pp. 531–537.

[28] Willmarth, W. W. and Wooldridge, C. E., "Measurements of the Correlation between the Fluctuating Velocities and the Fluctuating Wall Pressure in a Thick Turbulent Boundary Layer," Rept. 456, April 1963, AGARD.

[29] Wills, J. A. B., "On Convection Velocities in Turbulent Shear Flow," *Journal of Fluid Mechanics*, Vol. 20, 1964.

[30] Kistler, A. L. and Chen, W. S., "The Fluctuating Pressure Field in a Supersonic Turbulent Boundary Layer," TR 32-277, 1962, Jet Propulsion Lab.

[31] Bull, M. K., "Properties of the Fluctuating Wall-Pressure Field of a Turbulent Boundary Layer," AASU Rept. 234, March 1963, Univ. of Southampton.

[32] Maestrello, L., "Measurement of Noise Radiated by Boundary Layer Excited Panels," *Journal of Sound and Vibration*, Vol. 2, 1965.

[33] Ffowcs-Williams, J. E., "On the Convected Turbulence and its Relationship to Near-Field Pressure," AASU Rept. 234, June 1960, Univ. of Southampton.

[34] Lighthill, M. J., "Sound Generated Aerodynamically," The Bakerian Lecture, *Proceedings of the Royal Society (London): Series A*, Vol. 267, 1962.

[35] Lin, Y. K., "Non-Stationary Response of Continuous Structure to Random Loading," *Journal of the Acoustical Society of America*, Vol. 35, 1963.

[36] Lyamshev, L. M., "The Theory of Vibration of Non-Homogeneous Elastic Plates," *Soviet Physics—Acoustics*, Vol. 10, 1964.

[37] Maidanik, G., "Response of Ribbed Panels to Reverberant Acoustic Fields," *Journal of the Acoustical Society of America*, Vol. 34, 1962.

[38] Lyon, R. H. and Maidanik, G., "Statistical Methods in Vibration Analysis," *AIAA Journal*, Vol. 2, No. 6, June 1964, pp. 1015–1024.

[39] Smith, P. W., Jr., "Response and Radiation of Structural Modes Excited by Sound," *Journal of the Acoustical Society of America*, Vol. 24, 1962.

[40] Nikiforov, A. S., "Radiation from a Plate of Finite Dimensions with Arbitrary Boundary Conditions," *Soviet Physics—Acoustics*, Oct.-Dec. 1964.

[41] Kistler, A. L., "The Fluctuating Wall Pressures under a Separated Supersonic Flow," Rept. NAS 7-100, 1963, NASA.

[42] Mohsen, A. M., "Wall Pressure Fluctuation under Subsonic Separated Flow," D6-17094, 1967, The Boeing Co.

[43] Tani, I., "Experimental Investigation of the Flow Separation over a Step," Symposium Freiburg BR 26, BIS 29, Aug. 1957.

[44] Lambert, R. F., Smits, T. I., and Langseth, R. E., "Some Non-Gaussian Noise Processes Relevant to Fatigue and Reliability Prediction," TR 64, 412, June 1965, Air Force Materials Lab.