NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

A STUDY OF THE RESPONSE OF PANELS TO RANDOM ACOUSTIC EXCITATION

By Robert W. Hess, Leslie W. Lassiter,
and Harvey H. Hubbard

SUMMARY

An application is made of the method of generalized harmonic analysis to the problem of prediction of stresses in airplane-skin panels due to excitation by jet noise. The concepts of the theory are reviewed briefly and some of the significant parameters are evaluated in the tests. Measurements of stresses in some panels due to random acoustic excitation are presented and are found to be in general agreement with calculated results.

INTRODUCTION

It is well known that jet noise has in many instances caused fatigue failures of airplane-skin panels in proximity to the jet-engine exhaust stream. These failures have been mainly on the fuselage or the wings, depending on the type of engine installation. Some configurations having the engine exits relatively far aft have also experienced damage to panels in the tail assembly.

Very few data are available which would permit a designer to estimate dynamic stresses in panels exposed to a given random excitation. As a result, "ad hoc" modifications, such as an increase of skin gage or the addition of stiffeners, or both, have frequently been necessary after construction of the airplane in order to alleviate fatigue problems. Thus, there is serious need for a means of predicting in the design stage the dynamic stresses of a panel subjected to a given random excitation.

In reference 1, the techniques of generalized harmonic analysis have been applied to a theoretical treatment of this problem, and it appears that these techniques may afford a relatively simple approach to the problem. Therefore, the main purpose of the present paper is to determine how well this type of analysis applies in predicting panel stresses.

Although the discussion is mainly concerned with the problem of jet-noise excitation, certain of the results will be shown to apply to the case of propeller-noise excitation as well.

METHOD OF ANALYSIS

The concepts involved in analysis of the panel-response problem are illustrated in figure 1, which is a schematic illustration of the input-output relationships involved. The top sketch in figure 1 represents the spectrum of jet noise which is causing the vibration of the panel. The second sketch is the panel transfer function, or the square of the frequency-response quantity of interest – whether it be displacement or stress. This transfer function is necessary in relating the output to the input. The curve may have several peaks corresponding to the various modes of vibration but, for simplicity, only the first-mode response is shown. The bottom sketch is the output-response spectrum and is the product of the input function and the panel transfer function. If the panel transfer function is expressed in terms of stress per unit input, as in this case, then this output curve is the stress-response spectrum of the panel.

A useful index of the overall response of the panel is the mean-square stress, and this is the area under the output curve in figure 1. For the case where the input function varies only slowly with frequency and the panel-response curve is sharp, reference 1 shows that the expression for the mean-square stress σ^2 takes the simplified form

$$\sigma^2 \approx \frac{\pi}{4\delta}\omega_0 S_0{}^2 \Phi_N(\omega_0) \tag{1}$$

Thus, the stress in the panel is a function of four parameters, one associated with the input and three with the panel structural characteristics. These parameters are: (1) ω_0, the panel natural frequency, (2) $\Phi_N(\omega_0)$, the input noise at the panel natural frequency, (3) S_0, the static stress per unit pressure, and (4) the damping of the panel given as percent of critical damping and designated as δ. This damping factor δ is a measure of the sharpness of the response curve, denoted either by its width or by its height. Perhaps another form of this equation, utilizing panel dimensions, is of more practical interest. Thus, if the product $\omega_0 S_0{}^2$ is substituted for in terms of panel length (or width) a and panel thickness t, the proportionality relation

$$\sigma \propto \frac{\left[\Phi_N(\omega_0)\right]^{1/2} a}{\delta^{1/2} t^{3/2}} \tag{2}$$

is obtained from equation (1), where $\left[\Phi_N(\omega_0)\right]^{1/2}$ has the dimension of root-mean-square value of acoustic pressure per unit band width. The latter form is noted to apply to the root-mean-square stress.

RESULTS AND DISCUSSION

The rest of the present paper is devoted to an evaluation of how well equation (1) and expression (2) apply to stress predictions in panels exposed to jet noise. To make this evaluation, some flat panels measuring 11 inches by 13 inches were tested with a 4-inch air jet.

Noise Input

A typical spectrum of the jet noise is shown in figure 2. The main reasons for presenting these data are to point out again that, in the analysis, only the value of the input at the natural frequency of the panel is of interest and to indicate that these tests deal with overall noise levels of approximately 130 decibels. The frequency at which the spectrum is a maximum will vary with location in the noise field as well as with the size and velocity of the jet. Hence, the noise from a full-scale engine would probably be somewhat different from that for the example shown in figure 2. Therefore, no special significance is attached to the ordinate numbers in figure 2 except that they may be useful in checking the method of calculation of the present paper.

Panel Structural Characteristics

The means used to obtain the response and the damping characteristics of the test panels are illustrated in figure 3. The panels were exposed to the periodic noise from a laboratory siren which could be operated in such a manner as to vary both the fundamental frequency of the noise and its intensity. The siren output was not sinusoidal, as would be desired in the ideal case, but contained a few harmonics of relatively low intensity. This particular study was concerned only with the fundamental frequency, and the root-mean-square pressure P_1 of this component is used as a measure of the acoustic input to the panels. To avoid excitation from both sides, the panels were mounted on a rigid chamber which was acoustically insulated and vented so as to minimize the load on the back of the panel.

A sample frequency-response curve for a 0.040-inch aluminum-alloy panel is shown on the left side of figure 3, where the stress amplitude σ_{MAX} in pounds per square inch is plotted as a function of frequency for constant input pressure P_1. The panel is seen to have a very sharp response at its resonant frequency. The slight skewness of the curve is believed to be due to nonlinearities of the system. Experimental determination of the unstable part of the curve is somewhat difficult and, for that reason, this portion of the curve is shown as a broken line. Because of this difficulty, measurements of damping were based on the height of the response curves rather than on the width. By definition, δ is equal to the ratio $\frac{\sigma_{st}}{2\sigma_{MAX}}$ where σ_{st} is the stress amplitude at zero frequency. This static stress is obtained experimentally by evacuating the chamber to obtain a static differential pressure across the panel corresponding to $\sqrt{2}$ times the value P_1 of the dynamic tests.

Values of damping corresponding to the appropriate input pressures of the tests were determined with the aid of figure 3 for the calculations of this paper. It should also be noted that the damping values of figure 3 apply directly to the case of pure frequency excitation. Hence, for the case of excitation by the random spectrum of the jet, some account must be taken of the effective band width of the panel. This band width can be shown to be approximately equal to $2\delta\omega_0$.

By the means just discussed, evaluations of the three structural parameters necessary for a stress calculation were made. These parameters are: the resonant frequency ω_0, the panel damping δ, and the static stress per unit pressure S_0, which is equal to $\sigma_{st}/\sqrt{2}\,P_1$.

Panel Response

Before considering the measured stresses of the present tests, it is helpful to study some of the qualitative results. Characteristic time histories of the response of a panel to both periodic and random excitations are shown in figure 4. At the top of the figure is shown the panel response to a periodic excitation which in this case is the noise from a siren. It can be seen that the panel response is uniform and has a definite frequency which in this case corresponds to the fundamental frequency of the periodic input function.

At the bottom of figure 4 is shown the panel response to a random excitation which in this case is the noise from the 4-inch air jet. Again, the panel response has a definite frequency but, in this case, it is the first-mode resonant frequency of the panel. It will also be noted that pronounced beats are apparent in the response. This type of response is characteristic of a sharply tuned system with low damping and is an ample justification for studying only the first-mode response characteristic, as has been done in these tests.

The results of the siren tests of figures 3 and 4 would apply directly to the case of excitation by propeller noise; for this case, the panel would respond mainly to that noise frequency at or near its first mode of vibration. Because of the sharp panel-response curve, it can be anticipated that a small change in the propeller rotational speed would markedly change the panel stresses even though the noise-input level was unchanged.

COMPARISON OF THEORY AND EXPERIMENT

Some stress calculations by the simplified equation, equation (1), given in the analysis section of the paper have been made by using the values of S_0, ω_0, and δ determined in the experiments. In figures 5 and 6 these calculations are compared with measured stresses obtained when the panels were placed in the near-noise field of the 4-inch air jet. Figure 5 gives the results obtained from a 0.040-inch panel located at various axial and radial distances from the jet. In the plot on the lower left-hand side of figure 5, stress data are shown as a function of x/D, where x is the axial distance from the nozzle to the center of the panel and D is the jet diameter. The radial distance d is held constant at 1.75 diameters. It can be seen from this figure that the theory and experiment are in very good agreement. Likewise, good agreement is indicated in the plot on the lower right-hand side of figure 5, where radial distance was varied and the axial distance held constant at 6.4 diameters.

Figure 2 indicated that the overall noise levels for these tests were in the range of 130 decibels. Since jet-engine noise levels may be of the order of 150 decibels or perhaps even higher, it can be anticipated that the associated stresses will be considerably higher than those measured in the present tests. If the assumption is made that the shape of the spectrum from an engine is the same as that from the 4-inch air jet used in these tests, then an increase of 20 decibels in the overall noise level would result in an increase in the root-mean-square stresses by a factor of approximately 10.

Figure 6 gives results for the case where panel thickness is varied and input pressure is held constant. The root-mean-square stress is plotted as a function of panel thickness for a range of thickness from 0.020 inch to 0.081 inch. The panel natural frequencies ranged from 78 cycles per second for the 0.020-inch thickness to 250 cycles per second for the 0.081-inch thickness. According to expression (2), stress should vary inversely as the thickness to the 3/2 power, as is indicated by the dashed curve. The measured points tend to bear out this relation fairly well.

From the good agreement between theory and experiment shown in figures 5 and 6, it can be concluded that the method of generalized harmonic analysis is particularly well suited to this problem and that equation (1) applies well for the conditions of these tests.

CONCLUDING REMARKS

Some preliminary measurements of the response of aircraft-skin panels to a random acoustic excitation have been presented and were found to be in general agreement with the results of an approximate analysis. These panels were noted to have very low damping and to vibrate mainly in their first modes in response to a random-noise input. Root-mean-square stresses were noted to be proportional to the sound pressure per unit band width at the natural frequency of the panel and were higher for panels of less thickness.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., April 28, 1955.

REFERENCE

1. Miles, John W.: On Structural Fatigue Under Random Loading. Jour. Aero. Sci., vol. 21, no. 11, Nov. 1954, pp. 753-762.

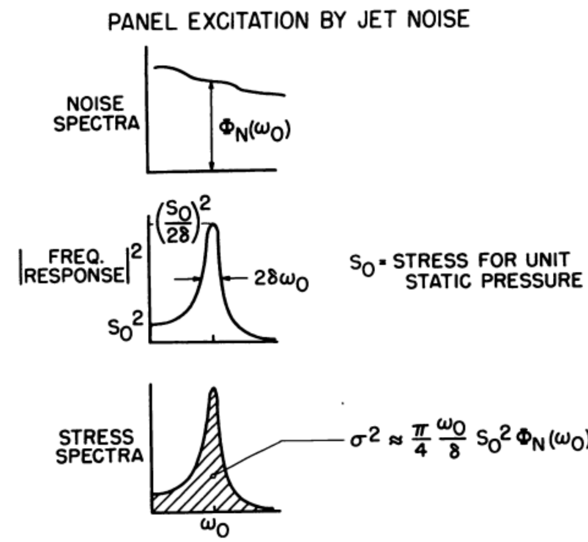

Figure 1

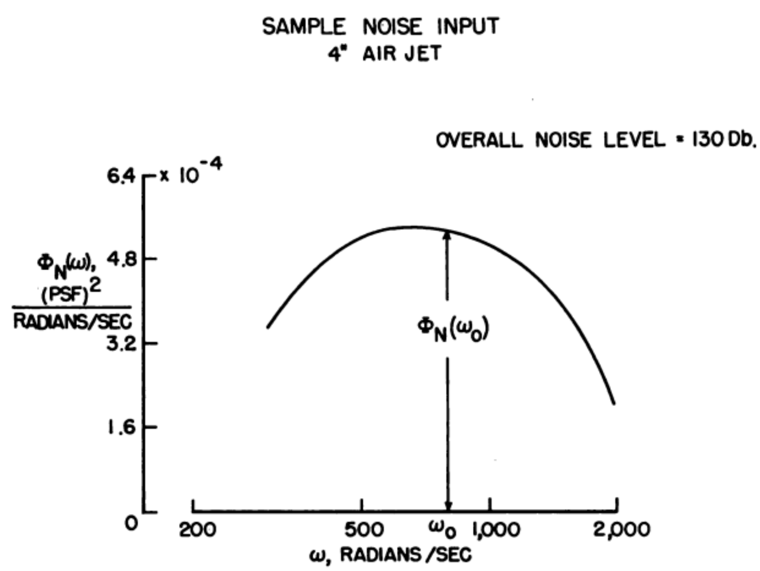

Figure 2

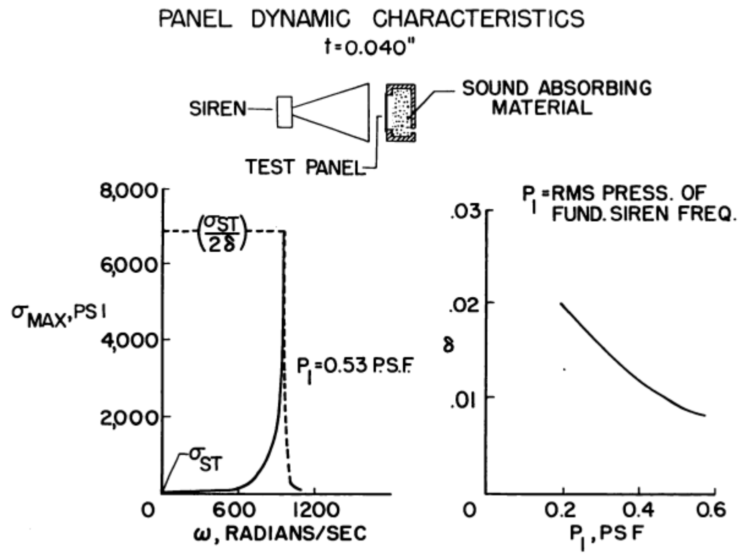

Figure 3

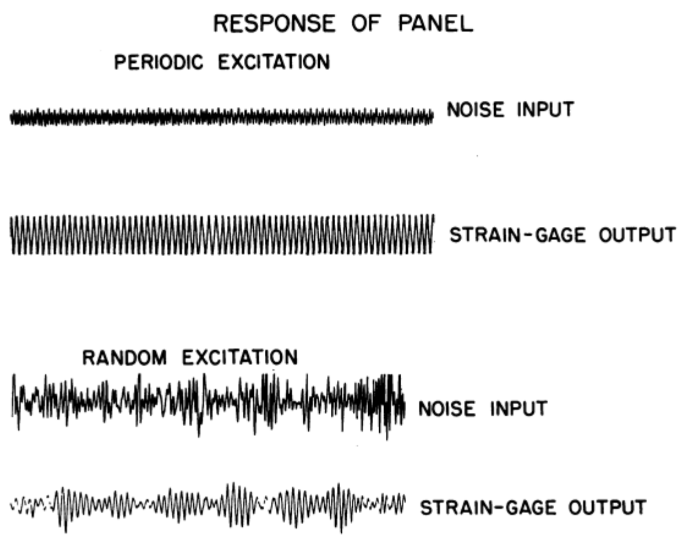

Figure 4

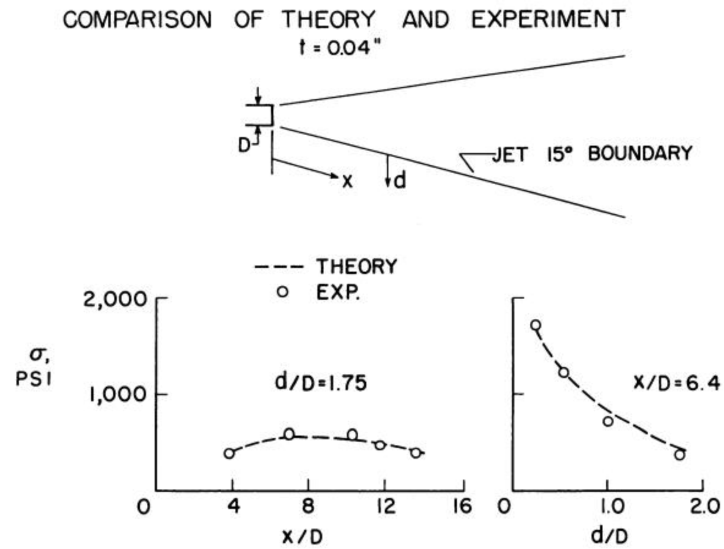

Figure 5

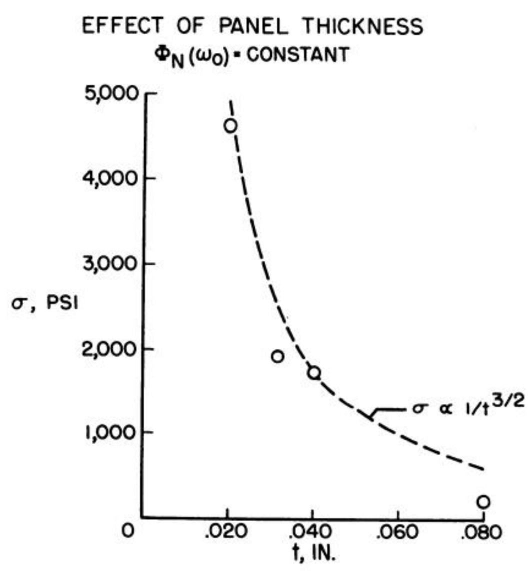

Figure 6