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research report

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AN EXPERIMENTAL PROGRAM FOR THE INVESTIGATION OF PANEL INSTABILITIES CAUSED BY SHOCK WAVES

by

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SUMMARY

This report outlines a series of experiments designed to examine panel instabilities induced by step and wedge generated shock waves. Two distinct modes of panel instability are discussed, namely conventional panel flutter involving the lower panel modes, and shock-panel coupling involving the higher panel modes. The experimental approach described here consists of driving the panel with an impedance head while monitoring force, acceleration and the phase lag between them; the magnitude of the phase lag gives a positive indication of panel instability. Following an introductory review of panel instability modes in Section 1.0, the experimental apparatus to be used for this investigation is described in Section 2.0. Finally the experimental procedure to be followed is outlined in Section 3.0.

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1.0 INTRODUCTION

During a previous experimental investigation of shock-panel coupling, impedance measurements were obtained from a flexible titanium panel mounted in the wall of a 7-inch wind tunnel (Reference 1). The tunnel was operated at a flow Mach number of 2.44, and the panel was excited mechanically by an impedance head, with and without a shock formed at the panel half-chord. It was found that, with the shock in position, an increase in impedance occurred when the panel was driven in its second mode. The increment in impedance was observed to be greater than the estimated contribution from the static pressure behind the shock and was tentatively ascribed to a shock-panel coupling mechanism.

Before discussing the present experimental program, it is convenient at this point to review some of the features of conventional panel flutter and to separate out shock-panel coupling and panel flutter as two distinct modes of instability. The flutter of a flexible panel excited by boundary layer turbulence is governed theoretically by the numerical value of its "shock-free" flutter parameter (or dynamic pressure parameter), λ, which is defined by the following relationship:

$$\lambda = \frac{\gamma p M^2}{\sqrt{M^2 - 1}} \cdot \frac{\ell^3}{D} \tag{1}$$

where

γ = specific heat ratio of the gas

p = surface pressure acting on the panel

M = flow Mach number

ℓ = panel length in the flow direction

D = flexural rigidity of the panel = $\dfrac{E h^3}{12(1 - \nu^2)}$

E = Young's modulus of elasticity

h = panel thickness

ν = Poisson's ratio.

Theory (References 2 and 3), shows that for a simply supported panel of infinite aspect ratio, the critical flutter parameter, λ_{crit}, has a numerical value of 343; for λ less than λ_{crit} the panel is regarded as being stable while for λ greater than λ_{crit} the panel is theoretically unstable.

In a linear theory, the critical flutter condition represents a sustained harmonic oscillation. On the unstable side of the flutter boundary, the amplitude of oscillation increases exponentially with increasing time, while on the stable side of the boundary it subsides exponentially. Experimentally, however, a different situation usually arises. The panel lies in a flow that is turbulent so that on the stable side a random oscillation of small amplitude is observed; on the unstable side, the growth in amplitude of oscillation is limited by the nonlinear effects of membrane tension so that a steady limit cycle oscillation is normally observed.

The critical flutter condition for a flat panel is illustrated in Figure 1, which shows typical amplitudes of panel oscillation as a function of increasing dynamic pressure for a flow Mach number of 2.81. These results, which were obtained by Anderson (Reference 4), show quite clearly the critical flutter condition, defined by the point where the sudden increase in the amplitude of oscillation occurs.

Furthermore, in a linear theory the time dependence of the solution is assumed to be of the form $e^{i\omega t}$; a panel vibrating in a vacuum has a real-valued frequency spectrum $(\omega_1, \omega_2, \omega_3, \ldots \omega_n)$. If aerodynamic damping is ignored the theoretical frequency spectrum changes continuously with increasing dynamic pressure of the flow, q, provided all other parameters are fixed. A critical dynamic pressure q_c is reached when two panel frequencies coalesce; at $q > q_c$, the pair of coalesced frequencies become a pair of complex conjugate numbers, one of which indicates a divergent oscillation.

If aerodynamic damping is considered, then the situation is similar except that before the frequency coalescence the ω_n are complex with positive imaginary parts so that the motion is convergent. This situation continues until q_c is reached, at which point the real parts of a pair of ω_n's coalesce, then when $q > q_c$ one of the roots ω_n will have a negative imaginary part, corresponding to a divergent oscillation (Reference 5). However, this frequency coalescence is not always observed in practice; this is illustrated in Figure 2, which shows the effects of increasing dynamic pressure on the resonant frequencies of typical panels. These results, which were obtained by Anderson (Reference 4) and Muhlstein (Reference 6), show that significant variations in the panel resonant frequencies occur as the dynamic pressure is increased. Anderson's results were obtained by both mechanical excitation of the panel and harmonic analysis of the random oscillations induced by the flow; at the onset of flutter the amplitude of the response of the fundamental panel mode completely submerged the response at the other frequencies, confirming a single mode type of flutter. Muhlstein's results were obtained by mechanical excitation throughout, using an impedance head; the impedance and phase angle between force and velocity were recorded continuously. The variation in the real part of the impedance was used as an indication of flutter; this component decreasing as flutter is approached, becoming zero at the flutter boundary, and assuming a negative value when the panel is fluttering. Muhlstein's results show that, for the panel investigated, flutter at a Mach number of 1.1 involved only the fundamental panel mode whereas panel flutter at a Mach number of 1.4 involved coalescence of the fundamental and first harmonic.

While the linear theory predicts frequency coalescence at flutter, it should be remembered that the theory is based on free vibration while experimental studies involve forced oscillations derived from wind tunnel turbulence. Hence, as q becomes larger than q_c, while one theoretical mode whose frequency ω_n becomes complex with a negative imaginary part passes into a limit cycle oscillation, other modes may still be excited by the random loading.

Turning attention now to the problem of shock waves impinging on the panel surface, the stability of panels in supersonic flow in the presence of regularly reflecting oblique shock waves has been examined theoretically by Ellen (Reference 2). Stability boundaries have been presented for a range of Mach numbers and panel flutter parameters and the results indicate that a destabilizing effect occurs, even with very weak shock waves, if they reflect near the upstream edge of the panel. A typical stability chart for a simply-supported panel in supersonic flow (M = 3.0) is shown in Figure 3; the critical shock position, $\xi_0 = x_0/\ell$, has been plotted as a function of the incident shock wave angle to form static and dynamic stability boundaries. The critical values of ξ_0 are those at which the panel eigenvalues ($k^2 = m\,\ell^4\,\omega^2/D$, where m = mass per unit panel area, ω = frequency of panel oscillation and ℓ and D are as defined by Equation 1) become zero or coalesce with another to become complex. With increasing shock strength, the flutter and divergence instability regions are seen to enlarge until the two regions join, which in this case takes place at wave angles greater than about 35 degrees.

The theoretical analysis is based on the static aerodynamic approximation for the pressures, and by linearizing the conditions across the foot of the shock wave the surface pressure increment and Mach number decrement are expressed in terms of the unknown local surface angle. It is implied here that the waves from the shock-expansion interaction have negligible effect on the surface pressure distribution as given by the linearized approximation. It is also assumed that the time scale of the flow disturbances is much smaller than that of the panel oscillation, thus the panel is responding dynamically to the aerodynamic static pressure distribution in the flow; equivalently, the wavelengths in the flow and surface are poorly matched. Since panel flutter occurs in the lower modes (one of the first three harmonics) for which the ratio of the time scales is large, this assumption is justified.

Because the previous argument has been restricted to the lower panel frequencies required by the flow pressure approximation, it is important to determine whether the higher panel frequencies, which are well-coupled acoustically, will generate an instability. In this regime, the panel flutter analysis is unsuitable for predicting the coupling behavior, however, it is possible to readily obtain a qualitative estimate by use of the results obtained by Moore, Ribner, and Lowson (References 7, 8 and 9).

The sound generation upstream of the shock wave produces, on interaction with the shock, a downstream wave system consisting of a steady vorticity wave pattern and sound waves which attenuate for a certain range of incidence angles. If the downstream waves are coupled to the surface vibration, it may be expected that a higher mode instability would occur of a completely different type to that occurring in conventional panel flutter. For a shock wave generated by separated flow, the incident sound waves are almost parallel to the shock and a radiating downstream pressure wave system is produced of somewhat larger amplitude than those upstream, the amplitude increasing with Mach number. The panel itself could now supply a feedback mechanism to amplify the upstream disturbances and thus generate an instability.

The boundary layer vorticity interaction with the shock wave will also generate downstream sound waves leading to further coupling through the panel motion. For the case of a shock reflection, the two being almost normal to each other, the sound wave interaction with the first shock will generate a decaying pressure field over a wide range of Mach numbers, and the main effect at the second shock will be from the vorticity, originating at the first shock, interacting with the second shock.

In order to differentiate between the two possible types of panel instability, namely conventional panel flutter and shock-panel coupling, it is desirable to employ a panel which is only marginally stable in the lower modes. This panel will be poorly coupled acoustically in the lower modes and highly coupled acoustically in the higher modes; therefore, the second type of instability would be expected to show up in the higher modes.

2.0 DESCRIPTION OF THE EXPERIMENTAL APPARATUS

The experimental arrangement for the present investigation is essentially the same as that used for the previous experiments reported in Reference 1. However, a smaller impedance head is to be used in order to obtain more accurate measurements of force and acceleration at very low excitation levels.

Two titanium test panels will be used in this study; one 6 1/2 in. by 4 1/2 in. by 0.016 in. thick and the other, 6 1/2 in. by 4 1/2 in. by 0.010 in. thick. The panels are designed to be recessed into the tunnel sidewall fixture such that their effective dimensions are 5 in. in the flow direction and 3 in. transverse to the flow. The method of assembling the test panels is illustrated in Figure 4; flush-mounting is achieved by using closely spaced screws, shim-stock and a nonhardening bonding material. Since the bonding failure observed in the previous experiments (Reference 1) was due wholly to the brittleness of the bonding material, the present method of panel mounting should lead to a significant improvement in the experimental setup. Providing flexibility at the panel edges in this way should give boundary conditions which approximate closely to a simply-supported condition. The calculated resonant frequencies of the test panels, for both simple supports and full edge fixity, are tabulated below:

Mode (m,n)*	PANEL NO. 1 (0.010 in. Thick)		PANEL NO. 2 (0.016 in. Thick)	
	Simple Supports (Hz)	Fixed Edges (Hz)	Simple Supports (Hz)	Fixed Edges (Hz)
(1,1)	139.8	270.5	223.6	433
(2,1)	251	390	402	624
(3,1)	436	595	698	955
(1,2)	450	681	720	1092

* m = number of half-waves in flow direction.
n = number of half-waves transverse to flow.

Calculations show that the maximum stresses in the 0.010 in. and 0.016 in. panels caused by the tunnel starting load (a differential pressure of about 1.8 lb/in.^{2}) are about 85,000 lb/in.^{2} and 33,000 lb/in.^{2}, respectively; the former figures represents about 70 percent of the yield stress for Titanium alloy. The calculated shock-free flutter parameters for the 0.010 in. and 0.016 in. panels are $\lambda = 302$ and $\lambda = 73.6$, respectively; thus it can be seen that the thinner panel is quite close to the critical flutter condition for the given flow conditions ($M = 2.44$ and $q = 3.94$ lb/in.^{2}).

Two types of shock-generating device will be investigated during the present study; a simple step attached to the tunnel wall, measuring 4 in. by 1 in. by 3/4 in., and a wedge shaped body inserted into the test section of the tunnel. The principal advantage in using the wedge shaped body is derived through the absence of separated flow over the panel surface. Wedges having included angles of 5, 10 and 20 degrees are available for this study and the design details and method of mounting a typical wedge are shown in Figure 5. The support strut is fixed to the floor of the wind tunnel and its vertical slots allow the adjustment of the wedge to any desired position. The shock can thus be positioned at any point of interest on the panel.

A block diagram of the electronic apparatus to be used for the investigation is shown in Figure 6.

3.0 EXPERIMENTAL PROCEDURE

The objective of the experimental program is to investigate in detail the stability characteristics of two-panel specimens immersed in supersonic flow and subjected to the combined effects of shock waves of varying strength and a forced sinusoidal oscillation. Panel instability is to be detected from observations of the magnitude of the real component of impedance, which represents the damping at resonance. This quantity decreases as the unstable region is approached, becomes zero at the stability boundary, and is negative when the panel becomes unstable. When driving the panel with an impedance head, it follows that the real component of impedance becomes negative when the phase lag between the force and acceleration lies between 180 degrees and 360 degrees, and is zero at the two extremes. Thus, the phase angle between force and acceleration gives a positive indication of panel instability.

While the 0.010 in. thick test panel has been designed to be marginally stable under shock-free flow conditions, the use of a wedge-shaped body to generate the shock waves is of particular value in this investigation since it allows, at least in part, experimental verification of the shock-panel flutter theory proposed by Ellen (Reference 2).

Referring to Figure 3, it is observed that at a Mach number of 3, a shock wave having an angle of about 20 degrees, if reflecting anywhere on the leading third of a simply-supported panel surface, will lower the critical flutter parameter to 300. Similarly, a shock wave having an angle of about 29 degrees, if reflecting anywhere on the leading quarter of the panel, will lower the critical flutter parameter to approximately 100.

A stability chart for a Mach number of 2.44 has not yet been compiled; however, it is anticipated that the present series of experiments, using 5, 10 and 20 degree wedges, (giving shock wave angles of 28, 32.5 and 43.7 degrees, respectively) will show up this destabilizing trend.

The experimental procedure can be divided into a number of steps as follows:

(1) Assemble the 0.016 in. panel in the tunnel sidewall fixture and drive the panel with the impedance head to determine the resonant frequencies, mode shapes, damping and linearity of response. Plot force, acceleration and phase lag continuously while performing a sinusoidal sweep from 10 to 1200 Hz.

(2) Assemble the panel and fixture in the wind tunnel wall and establish flow in the wind tunnel at a Mach number of 2.44 (without the step), while simultanteously sweeping with the impedance head to determine shifts in the resonant frequencies. Plot, force, acceleration and phase lag continuously and record on tape. Perform a second tunnel run and record the pressure fluctuations at the downstream edge of the panel while dwelling at the resonant frequencies of the first few modes. Plot force, acceleration and phase lag continuously and record on tape.

(3) Position the 3/4 in. step so as to place the shock at the downstream edge of the panel. Establish flow in the tunnel and perform resonant dwell tests in the first few modes, recording the pressure fluctuations at the downstream edge of the panel for each case. Plot force, acceleration and phase lag continuously and record on tape.

Reposition the shock at the panel half-chord and sweep with the impedance head to detect possible flutter in the lower modes and shock-panel coupling in the higher modes. Plot force, acceleration and phase lag continuously and record on tape.

(4) Assemble the shock-generating wedge in the tunnel and adjust to position the shock at the downstream panel edge. Perform a sinusoidal sweep with the impedance head and plot force, acceleration and phase lag continuously and record on tape. Record the pressure fluctuations at the downstream edge of the panel during resonant dwells in the first few modes.

(5) Position the wedge-generated shock at distances from the panel leading edge of 5/8 in. (L/8), 1 1/4 in. (L/4), 1 7/8 in. (3L/8), and 2 1/2 in. (L/2), and perform a sinusoidal sweep for each case. Plot force, acceleration and phase lag continuously and record on tape.

For each shock position, perform resonant dwell tests in the first few modes and record the pressure fluctuations at the downstream panel edge.

(6) Assemble the 0.01 in. panel in the tunnel sidewall and repeat steps (1) through (5).

Data from these experiments will be analyzed to reveal the principal contributing mechanisms. The sine sweep experimental data will be presented in the form of graphs of force versus frequency, acceleration versus frequency, and phase lag versus frequency. In addition, selected mechanical impedance data will also be presented. Experimental data from the dwell tests will be presented as continuous records of force, acceleration and phase lag versus dwell time. Pressure fluctuation data recorded on tape will be analyzed and reduced to determine overall levels and spectral characteristics.

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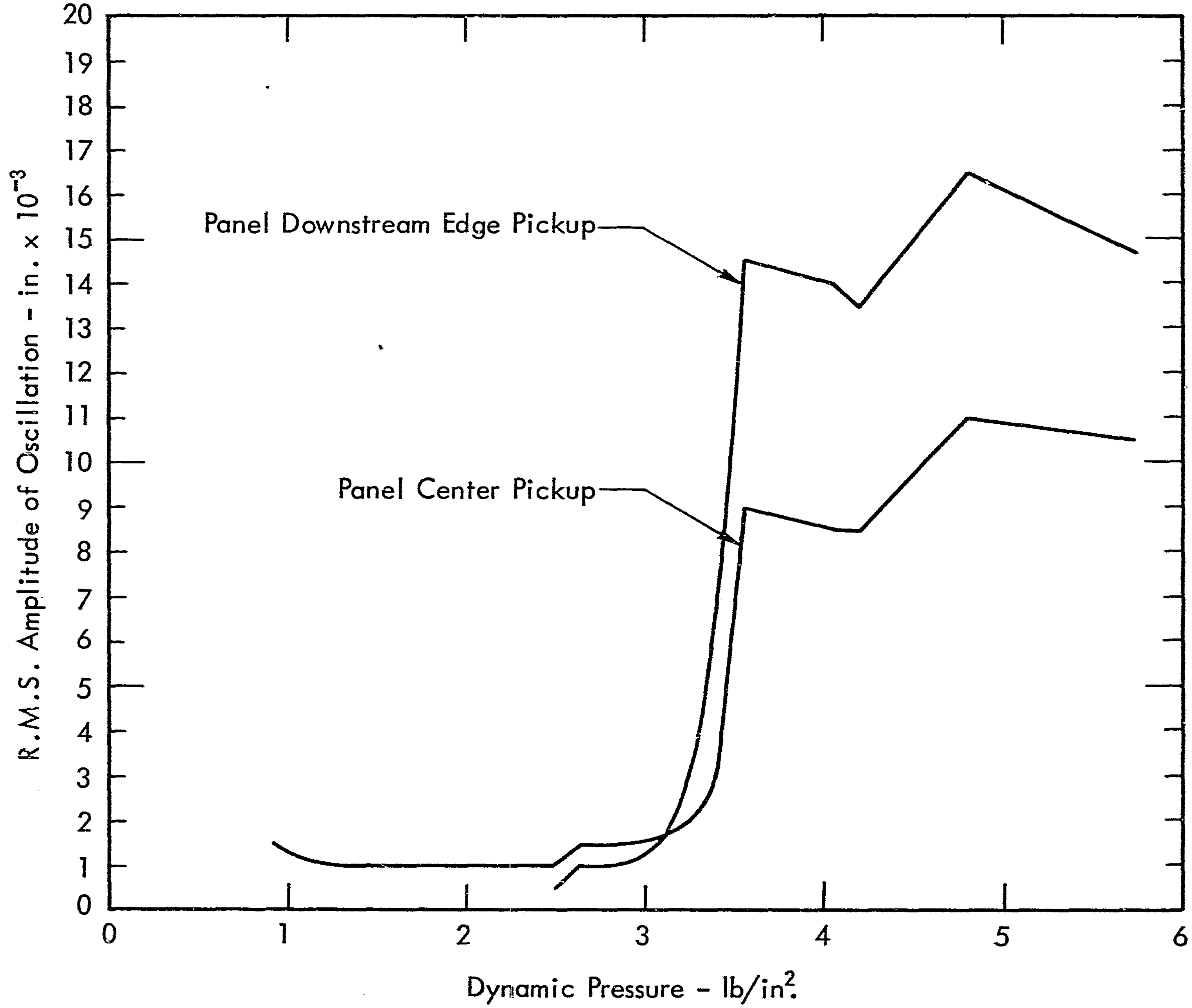

Figure 1. R.M.S. Amplitude of Panel Oscillation Versus Dynamic Pressure (Mach Number 2.81, Panel Thickness 0.015 in.) - After Anderson (Ref. 4)

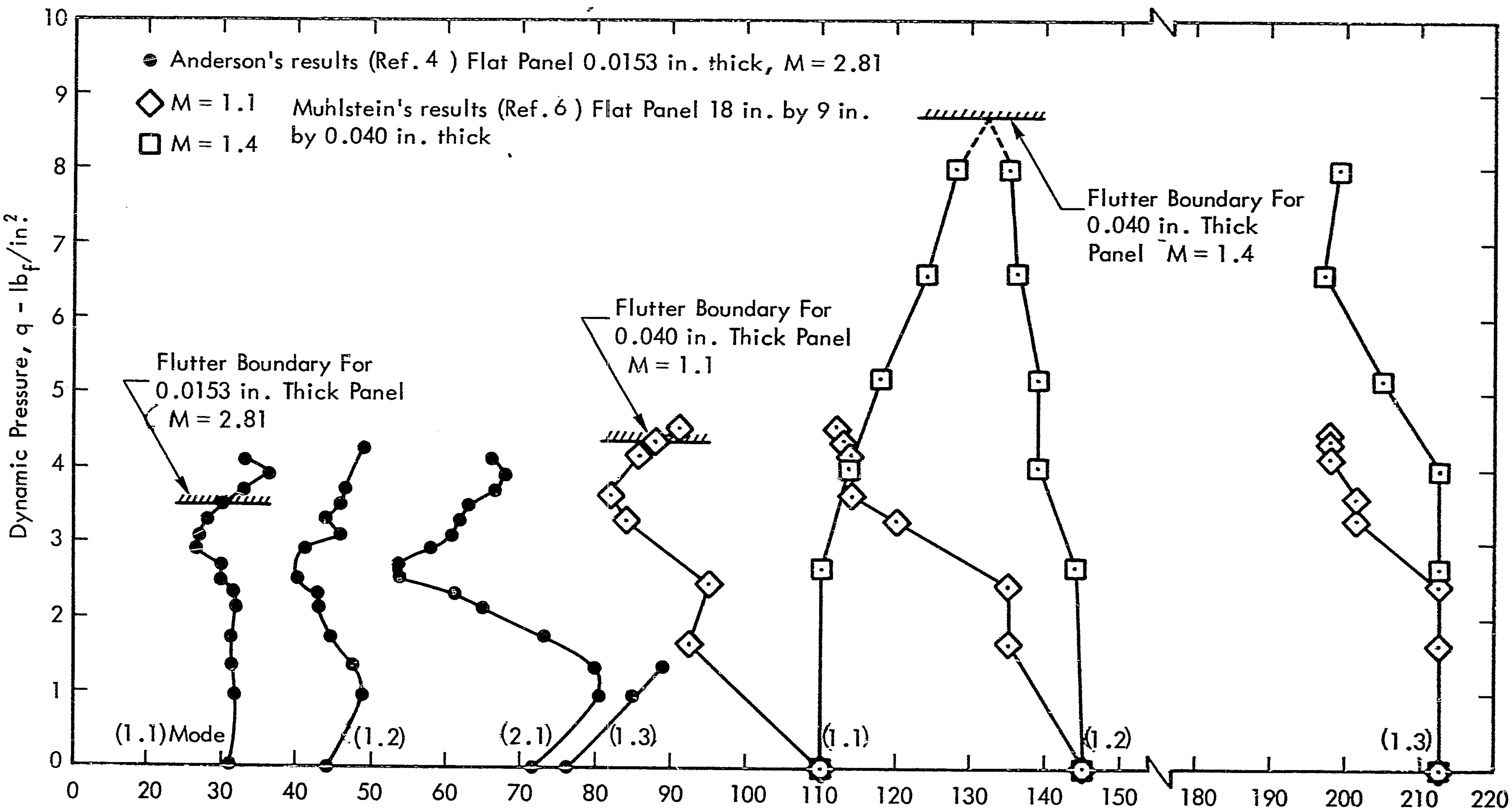

Figure 2. Effect of Dynamic Pressure on Panel Resonant Frequencies

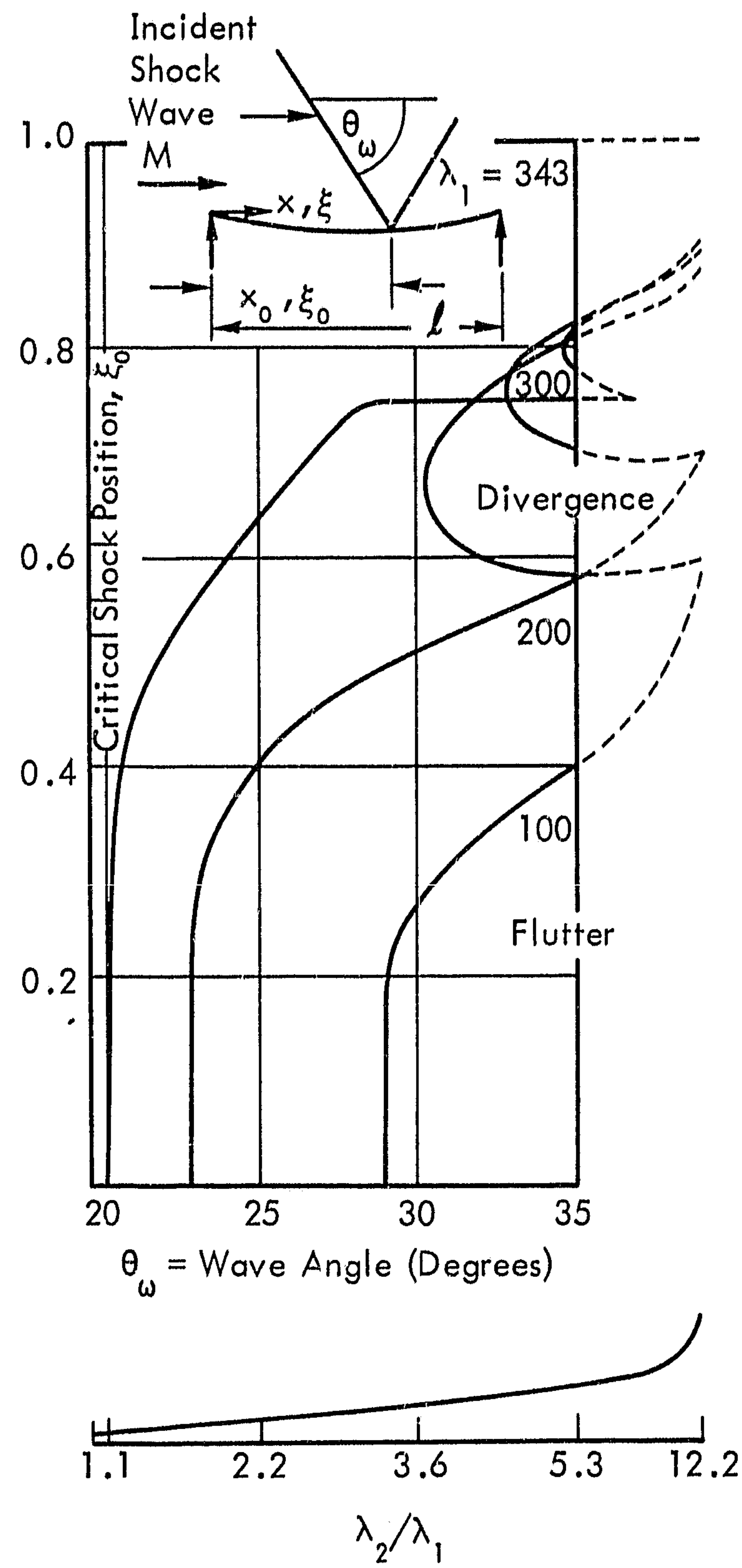

Figure 3. Simply Supported Panel Stability Boundaries ($M = 3$). After Ellen (Ref. 2)

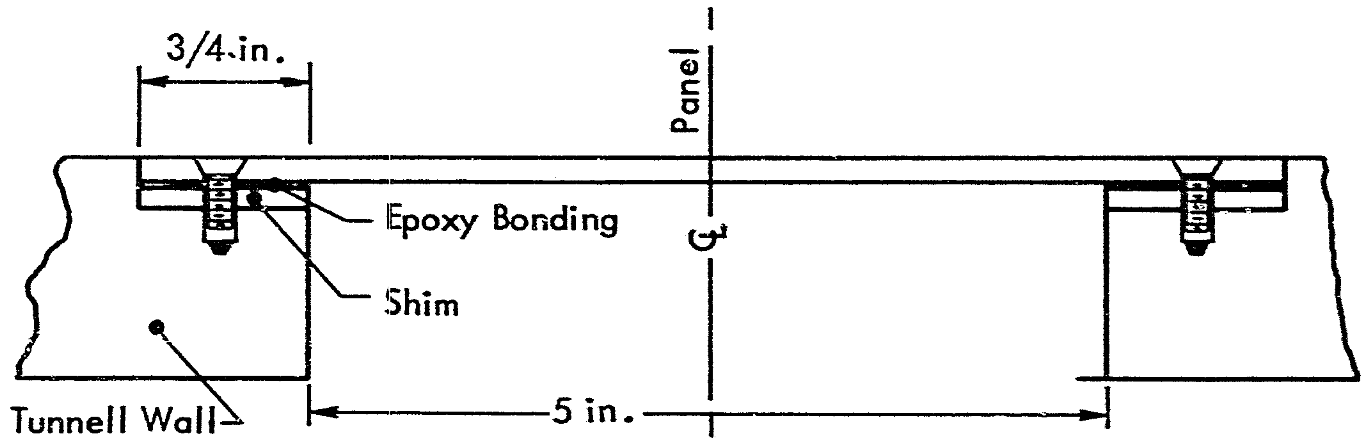

Figure 4. Panel Mounting Arrangement

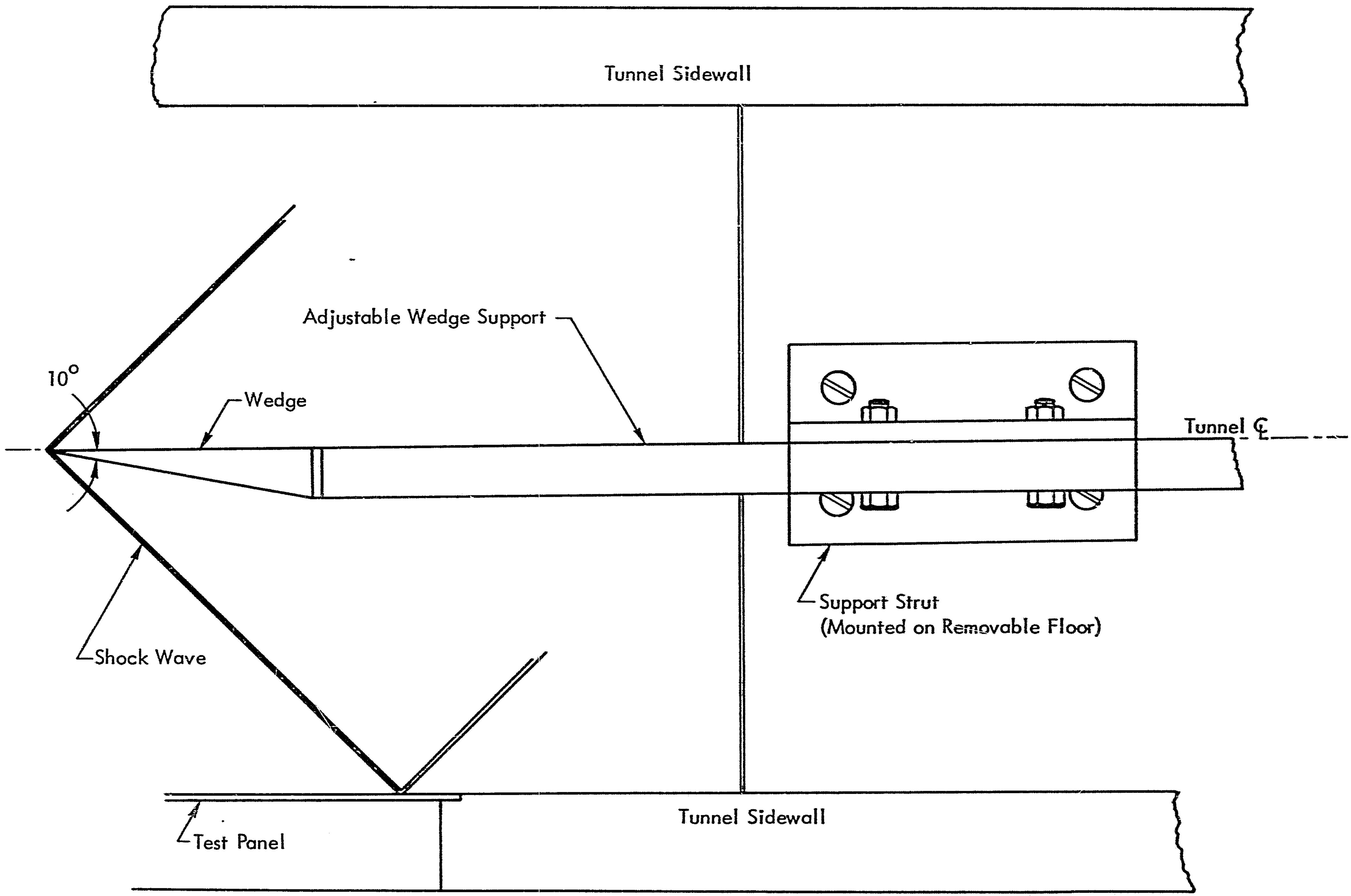

Figure 5. Details of the Shock Generating Wedge

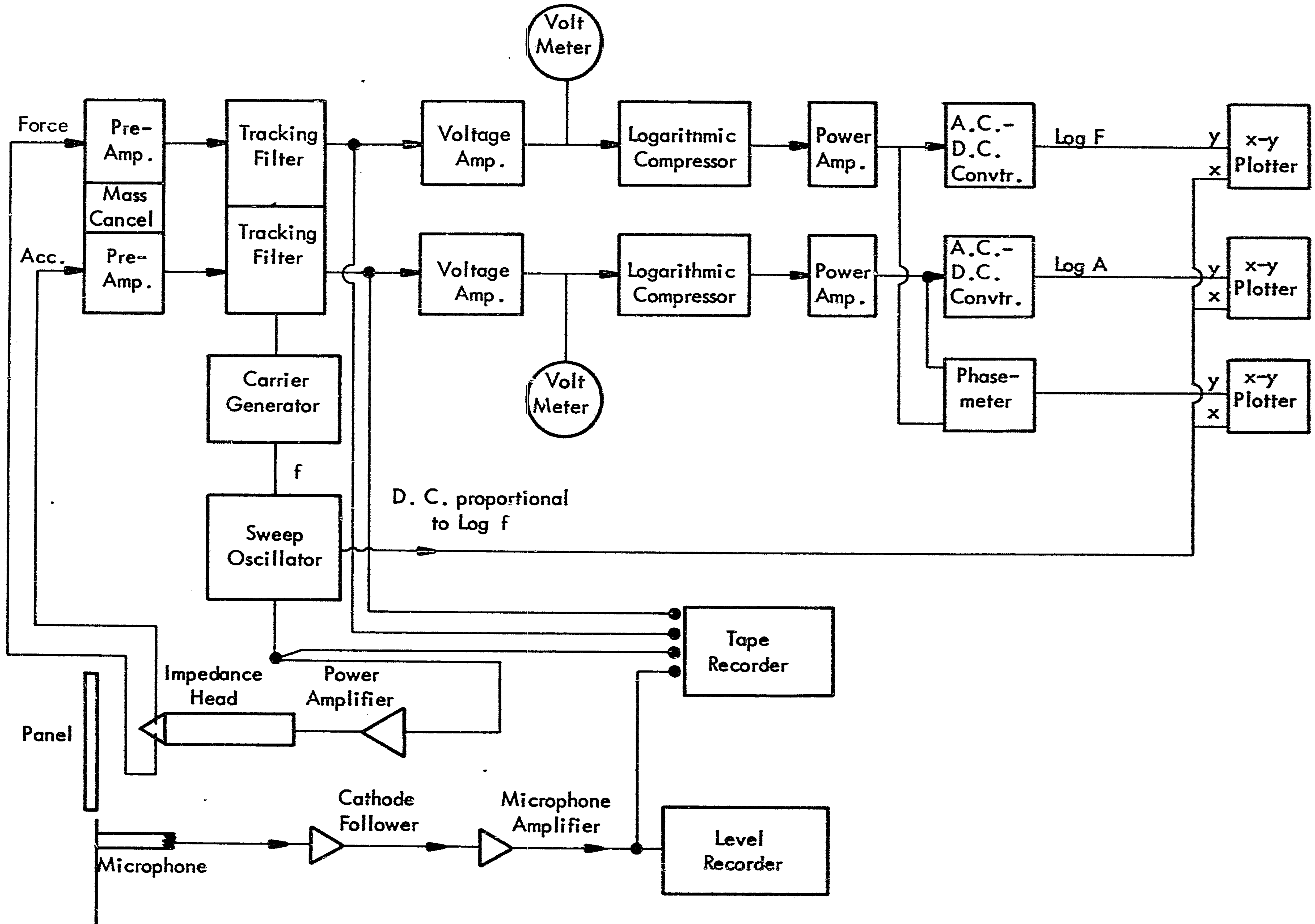

Figure 6. Block Diagram of the Electronic Apparatus for Panel Instability Experiments