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FLUCTUATING PRESSURE LOADS UNDER HIGH SPEED
BOUNDARY LAYERS

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SUMMARY

Aeroacoustic fatigue is anticipated to control the design of significant portions of the structures of advanced high-speed vehicles. This is due to contemplated long-duration flights at high dynamic pressures and Mach numbers with related high skin temperatures. Fluctuating pressure loads are comparatively small beneath attached turbulent boundary layers, but become important in regions of flow separation such as compression and expansion corners on elevons and rudders. The most intense loads are due to shock/boundary-layer interaction. These flows may occur in the engine-exhaust wall jet and in flows over control surfaces. A brief review is given of available research in these areas with a description of work underway at Langley.

INTRODUCTION

Aerothermal loads on hypersonic vehicles are largest during the low-altitude, turbulent flow portion of the flight.[1] These conditions are also the conditions which yield the largest aeroacoustic loads, that is, the fluctuating pressures under the turbulent layers. The combined effects of aerothermal and aeroacoustic loads are anticipated to elevate sonic fatigue to dominate the design of aft portions, such as elevons, rudders, and surfaces exposed to engine exhaust, of hypersonic aircraft. These areas of the structure may be exposed to separated flow and the interactions between shock waves and boundary layers, which are high loading sources.

This paper discusses some recent research on fluctuating pressure loads under boundary layers. It is not intended as a review, but only to highlight some of the more recent and significant research. The discussion here was prepared in support of a workshop on sonic fatigue and hypersonic loads which was held at the AIAA 11th Aeroacoustics Conference at Sunnyvale, California, October 19–21, 1987.

The discussion below first reviews some available prediction methods—the broad rules used by designers. Then some recent research is considered in light of its possible impact on these predition methods. Three general categories of boundary layer flow are used for the purpose of discussion. These are (first) the attached turbulent boundary layer, (second) the separating turbulent boundary layer, and (third) the turbulent boundary layer with shock interaction. The latter category may experience separation due to the concentrated pressure gradient in the vicinity of the shock foot. The discussion concludes with a description of work in progress at Langley Research Center.

PREDICTION METHODS

Attached Turbulent Layer

Reviews of available prediction methods[2,3,4,5] indicate that the attached turbulent layer has relatively small pressure fluctuations at the wall. The dynamic pressure q at the edge of the boundary layer is frequently used as a scaling parameter for the overall root-mean-square pressure p_{rms}. Lowson's formula[6] gives an estimate of the rms pressure as shown in Fig. 1. The subsonic condition, where Mach number is small, gives $p_{rms} = 0.006q$. In the supersonic range ($M < 5$) the rms pressure decreases with the square of Mach number.

Flow Description		rms Pressure	Power Spectrum, Φ
Attached Turbulent Layer		$\frac{0.006}{1+0.14\,M^2}\,q$	$p_{ms}\,\frac{\delta^*}{U}\,\frac{2/\pi}{1+(\omega\delta^*/U)^2}$
Separated Turbulent Layers			
	Compression Corner	$0.022\,q$	$p_{ms}\,\frac{\delta_e}{0.17U}\left[1+\left(\frac{f\delta_e}{0.17U}\right)^{.83}\right]^{-2.15}$
	Expansion Corner	$\frac{0.045}{1+M_e^2}\,q$	
	Shock/B-L Interaction	$\frac{0.14}{1+0.5\,M^2}\,q$	

Fig. 1. Prediction methods for fluctuating pressures under high speed boundary layers.

The power spectrum for the attached boundary layer is often given as a function of circular frequency ω, scaled with the boundary layer displacement thickness δ^* and velocity at the edge of the boundary layer U. The magnitude of the power spectrum may be scaled with the mean-squared pressure p_{ms}, δ^*, and U. The Houbolt formula[7] shown in Fig. 1 is a relatively accurate example of this scaling. This formula is readily integrated to demonstrate its conformance with the usual constraint on power spectrum

$$p_{ms} = \int_0^\infty \Phi(\omega)\, d\omega$$

The Houbolt formula has the properties that the power spectrum is flat at low frequencies and decays as frequency-squared at high frequencies. Other empirical formulas have similar, but not the same, characteristics. It will be seen later that recent data indicate that the high- and low-frequency character of power spectra remains an open question.

Separated Turbulent Layers

Boundary layer separation may result from adverse streamwise pressure gradients. A typical experimental flow where this occurs is the compression corner at a ramp or a cylinder-cone transition. The fluctuating pressure is about 2% of the dynamic pressure in a compression corner. This value varies with Mach number, but the variation has not been clearly defined with empirical formulas. The power spectrum developed by Robertson[8] gives a fair representation for this load. This formula is similar to the Houbolt equation except for the use of boundary layer thickness δ instead of δ^* and the fractional exponents. Note that these exponents give an $f^{-1.83}$ variation instead of an f^{-2} variation at high frequency.

Separation at an expansion corner is a source of significantly higher loads than in the compression corner. As indicated in Fig. 1, these are 4–5% of the local dynamic pressure at low Mach numbers. The M^2 term reduces these loads at higher Mach numbers. The power spectrum for these loads is the same as for the compression corner loads.

Turbulent Layers with Shock Interaction

The fluctuating pressures resulting from separation due to shock/boundary-layer interaction produce the largest loads of these flows. Chaump, Martellucci, and Montfort[2,3] developed the formula shown in Fig. 1 which gives a fluctuating rms presssure near 10% of the dynamic pressure in the transonic range above Mach 1. The power spectrum for SBLI loads is complicated by the fact that there are two contributing sources of the loads. These are the oscillations of the shock wave about its mean position and the sources due turbulence passing through the steep pressure gradients near the foot of the shock. Recent research on these loads and their spectra is discussed below. The point here is that they are the largest loads and, as mentioned earlier, that they occur in conjunction with high aerothermal loads. This conjunction suggests that high-temperature acoustic fatigue may be a significant design limitation for hypersonic aircraft.

ATTACHED BOUNDARY LAYERS

Fluctuating pressures under attached turbulent layers depend on the wall temperature as well as on the Mach number. Laganelli, Martellucci, and Shaw[10] have developed a prediction method which incorporates this effect through a compressibility factor ϵ_T. As shown in Fig. 2, the compressibility factor depends on both the Mach number and the ratio of the wall temperature T_w to the adiabatic wall temperature T_{aw}. It also includes an effect of the velocity profile, through the power law exponent n, and an effect of boundary layer growth rate, through the viscous exponent m. In Fig. 2, n is 7, for the $(y/\delta)^{1/7}$ velocity profile, and m is 4/5, for the flat plate turbulent layer growth rate. The power spectra in Fig. 2 are Houbolt curves scaled with the compressibility factor. The authors of this

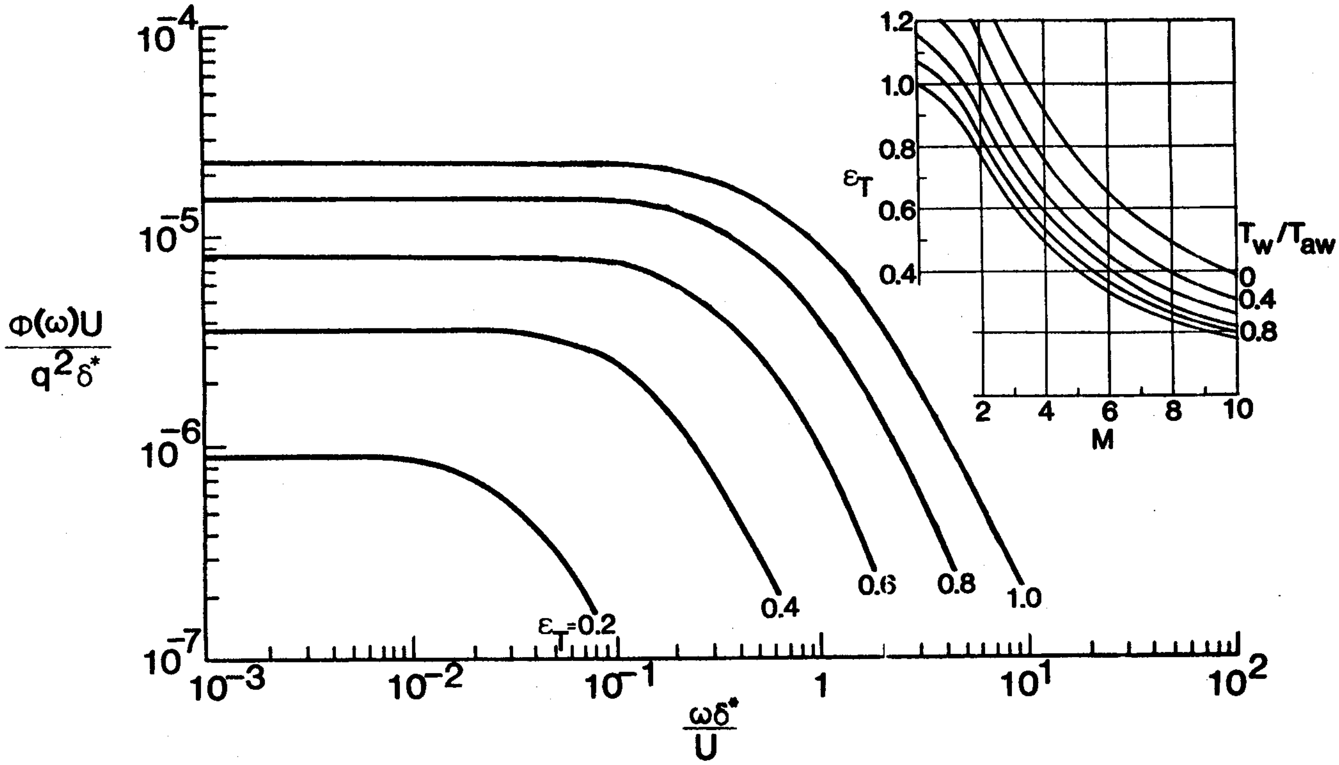

Fig. 2. Power spectrum of pressure fluctuation under an attached boundary layer. From Laganelli, Martellucci, and Shaw, AIAA J., Vol. 21, No. 4, 1983, pp 495–502.

method have shown good correlations with the data of Laganelli[9] and with the data of several earlier investigators.

Instrumentation for fluctuating pressure measurement is still inadequate, as was shown recently by Schewe.[11] Measurements were made in a very low speed flow, $U = 6\text{m/s}$, in a nearly disturbance-free tunnel. Schewe developed a special 1mm diameter transducer and compared measurements to larger transducers having diameters of 2.1, 4, 9, and 18mm. It was found that only the smallest transducer with dimensionless diameter $(dU_\tau/\nu) = 19$, based on friction velocity and kinematic viscosity, could resolve the pressure fluctuations. Extrapolation to an infinitesimal transducer size indicated that the resolution was within 4% of the actual value, which was slightly over $0.01q$, nearly twice the "accepted" value of $0.006q$ shown in Fig. 1. This careful study of a comparatively simple turbulent layer illustrates the need for further instrumentation development and the need to repeat many of the older experiments with modern instrumentation and data processing techniques.

Schewes results for power spectrum show an $f^{-7/3}$ high-frequency variation in accordance with isotropic turbulence theory. Since the model equation of Laganelli, Martellucci, and Shaw[10] is integrable with arbitrary exponents, the isotropic characteristic could be incorporated into their method. Of course, it would be necessary to support this change with data from both subsonic and supersonic boundary layers.

SEPARATED BOUNDARY LAYERS

Simpson, Ghodbane, and McGrath[14] use the maximum shear stress within the boundary layer to scale wall pressure fluctuations in the vicinity of boundary layer separation. Recall that wall shear stress τ_w has been used as a scaling parameter for the fluctuating pressure under attached boundary layers. The maximum shear stress scaling introduces the effect of boundary layer profile, as represented by displacement thickness δ^* and momentum thickness θ, by estimating the maximum stress by

$$\tau_M = \rho U^2 \left[\frac{1}{6.55}\left(1 - \frac{\theta}{\delta^*}\right)\right]^2$$

where ρ is the density. Data from 10 experiments indicate that

$$p_{rms} = (1.5 \pm .07)\tau_M$$

where the above variation is within one standard deviation. Correlations of the same data based on wall shear stress show significantly larger variation than on the maximum shear stress.

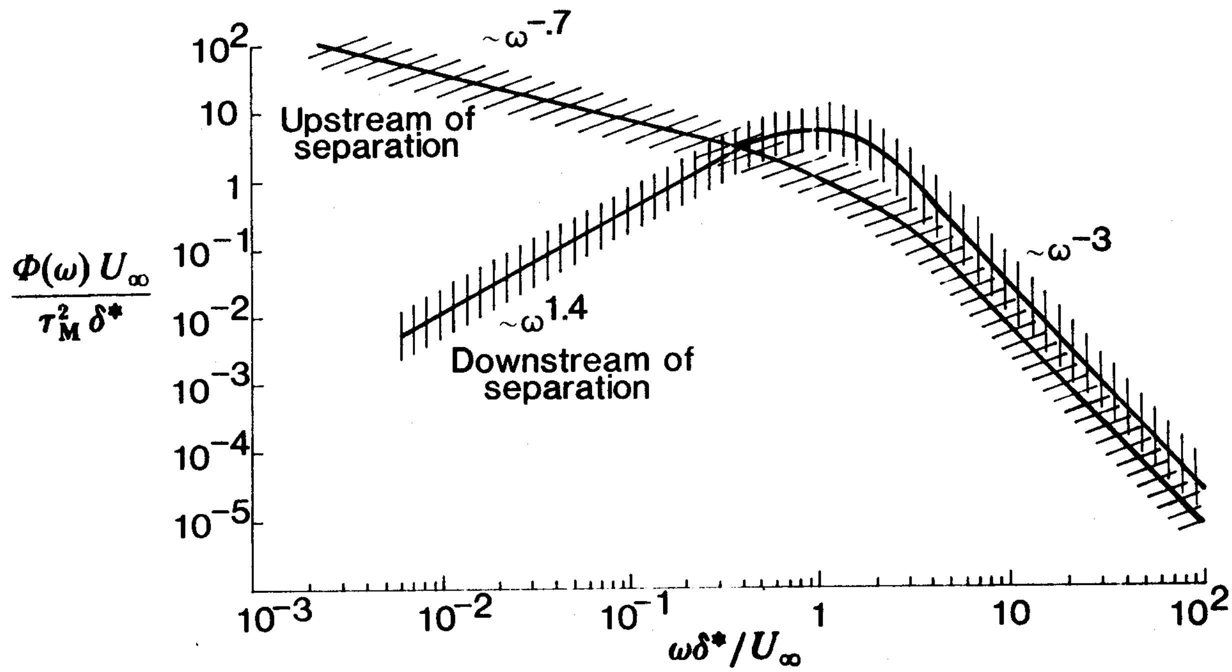

Fig. 3. Power spectrum of pressure fluctuation under a separating subsonic boundary layer. From Simpson, Ghodbane, and McGrath, J. Fluid Mech., Vol. 177, 1987, pp 167–186.

The power spectrum upstream and downstream of separation is distinctly different, as shown in Fig. 3. Upstream, where the attached layer has an adverse pressure gradient, the low-frequency spectrum varies as $\omega^{-0.7}$ and the high-frequency spectrum varies as $\omega^{-3.0}$. Downstream, the low-frequency variation is $\omega^{1.4}$. This is followed by a sprectrum peak and the same high-frequency roll off. These data were taken by microphones under 0.74mm diameter pinholes, $(dU_\tau/\nu) < 43$. The difference in transducer design and size may account partially for the discrepancy between this high-frequency spectrum and the data of Schewe.[11] Since the flows are physically distinct, the spectral differences are more probably real differences.

SHOCK/BOUNDARY-LAYER INTERACTION

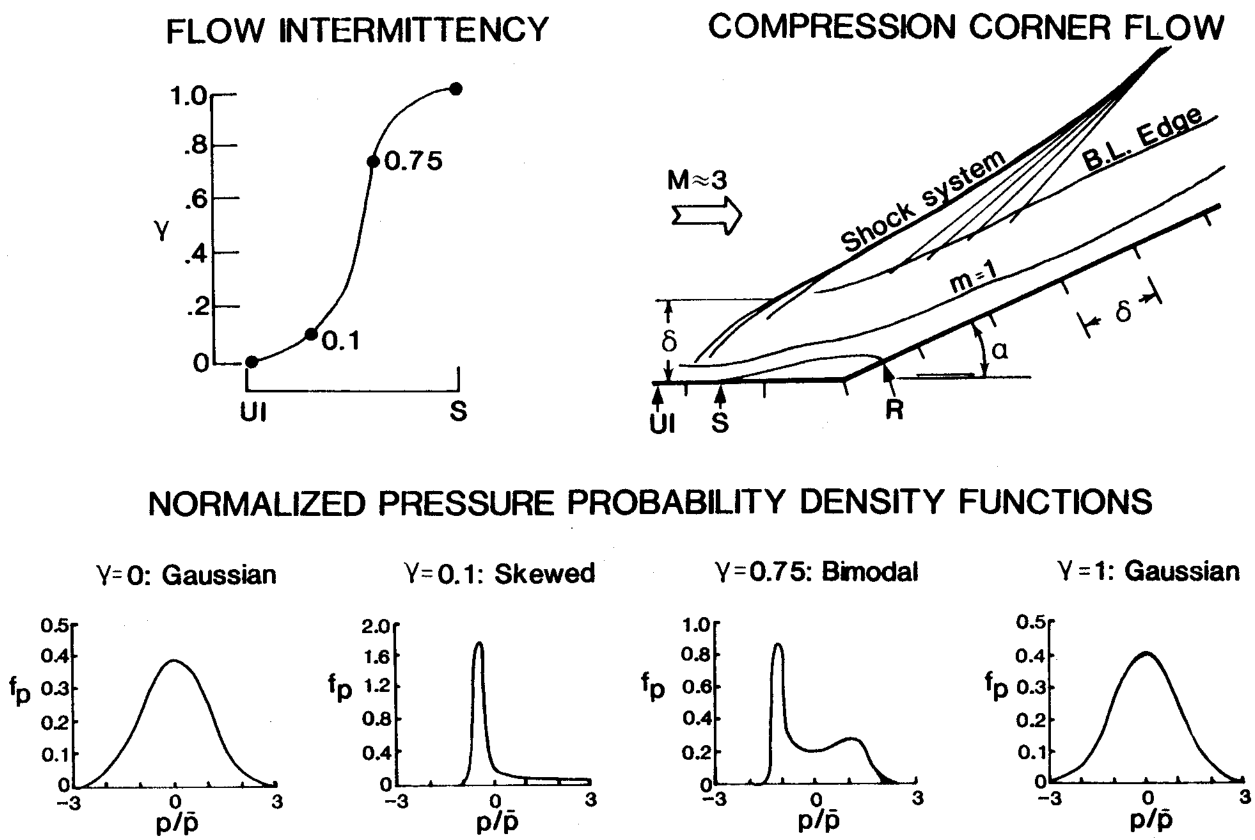

Fig. 4. Pressure fluctuation under shock/boundary-layer interaction with flow separation. From Dolling and Murphy, AIAA J., Vol. 21, No. 12, 1983, pp 1628–1634, and from Dolling and Or, Experiments in Fluids, Vol. 3, 1985, pp 24–32.

Shock/boundary-layer interactions may occur in supersonic flow in compression corners. Fig. 4 shows a supersonic ramp flow studied by Settles.[15] The Mach number is slightly less than 3 and shocks are observed with ramp angles between 12 and 24 degrees. The point of separation S is 1 or 2 upstream boundary layer thicknesses from the corner. Dolling and Murphy[16] have investigated the intermittent flow and shock unsteadiness near this corner. They define the intermittency as the probability that the wall pressure will exceed the upstream average pressure by three standard deviations. The upstream influence point UI is then defined by $\gamma = 0.04$, which is essentially where the intermittent flow begins. This intermittent region extends to about the point of separation. Dolling and Or[17] measured the probability distribution of the fluctuating pressures at points within this intermittent region. At the upstream influence point, the probability density is approximately Gaussian. Within the intermittent region, where γ is fairly small, the density function is skewed to the left because the fluctuations are dominated by upstream characteristics. Further downstream, the density function becomes bimodal, representing the intermittent upstream and downstream characteristics. In this bimodal regime,the point of peak rms pressure is between $\gamma = 0.7$ and $\gamma = 0.8$. Beyond the separation point, the density function returns to a near-Gaussian form. These changes all occur within about one boundary layer thickness.

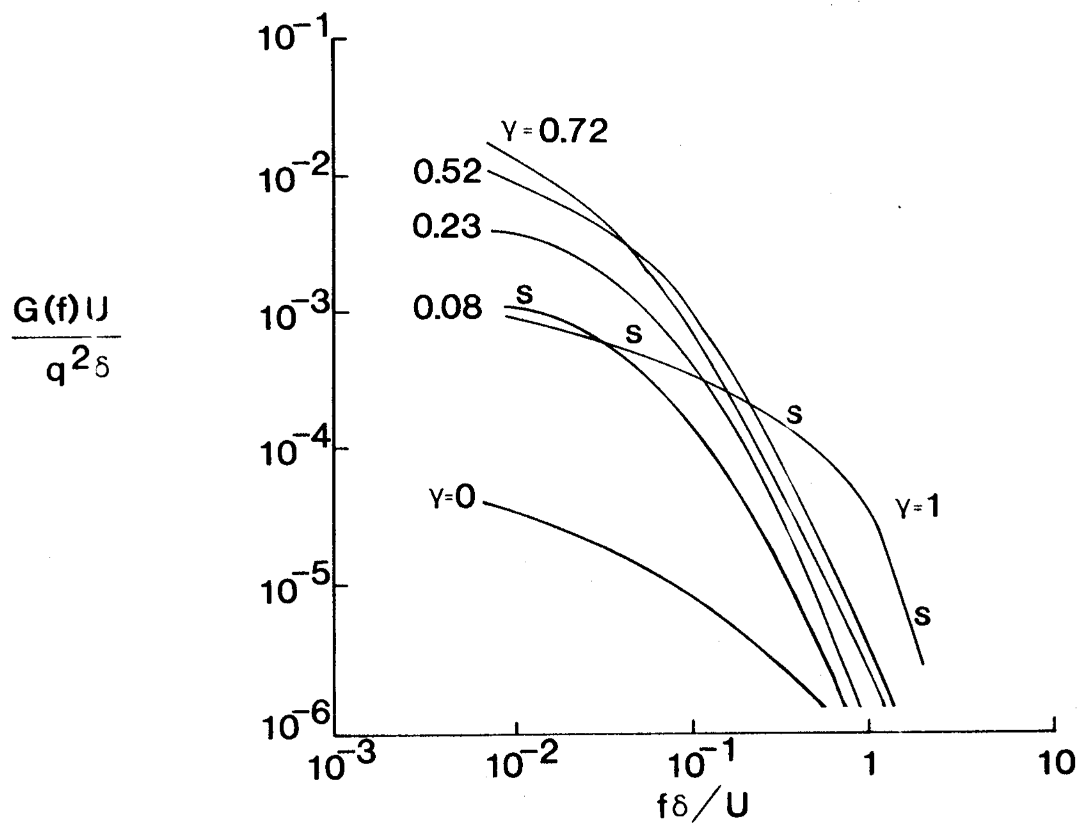

Fig. 5. Power spectrum of pressure fluctuation under shock/boundary-layer interaction. From Dolling and Or, Experiments in Fluids, Vol. 3, 1985, pp 24–32.

The power spectrum varies rapidly with streamwise position within the intermittent region. Fig. 5 shows the spectrum measured by Dolling and Or[17] with the pressure intermittency as a parameter. The upstream spectrum, $\gamma = 0$, is relatively small in magnitude since it represented the attached boundary layer. The spectrum level rises rapidly with increasing γ to a peak rms pressure at $\gamma = 0.72$ and then to the separated flow spectrum at $\gamma = 1$. The curves of Fig. 5 do not show a frequency of peak pressure fluctuations. If a peak exists, it might lie near $f\delta/U = 10^{-4}$, a small value when compared to the subsonic measurements of Simpson, Ghodbane, and McGrath.[14] The implication for sonic fatigue is that the structure will be excited at low frequencies with high pressure levels spread over a small surface area. There are comparatively few studies of fluctuating pressures under three-dimensional shock/boundary-layer interactions. A recent exception is the work of Tan, Tran, and Bogdonoff[18] on the interaction between a sharp-fin-generated shock and a turbulent boundary layer. Pressure fluctuations under this interaction were relatively small— roughly twice those in the upstream attached boundary layer. The shock was more steady than those in the compression corner, although some unsteadiness was observed.

LANGLEY HIGH-TEMPERATURE TUNNEL EXPERIMENT

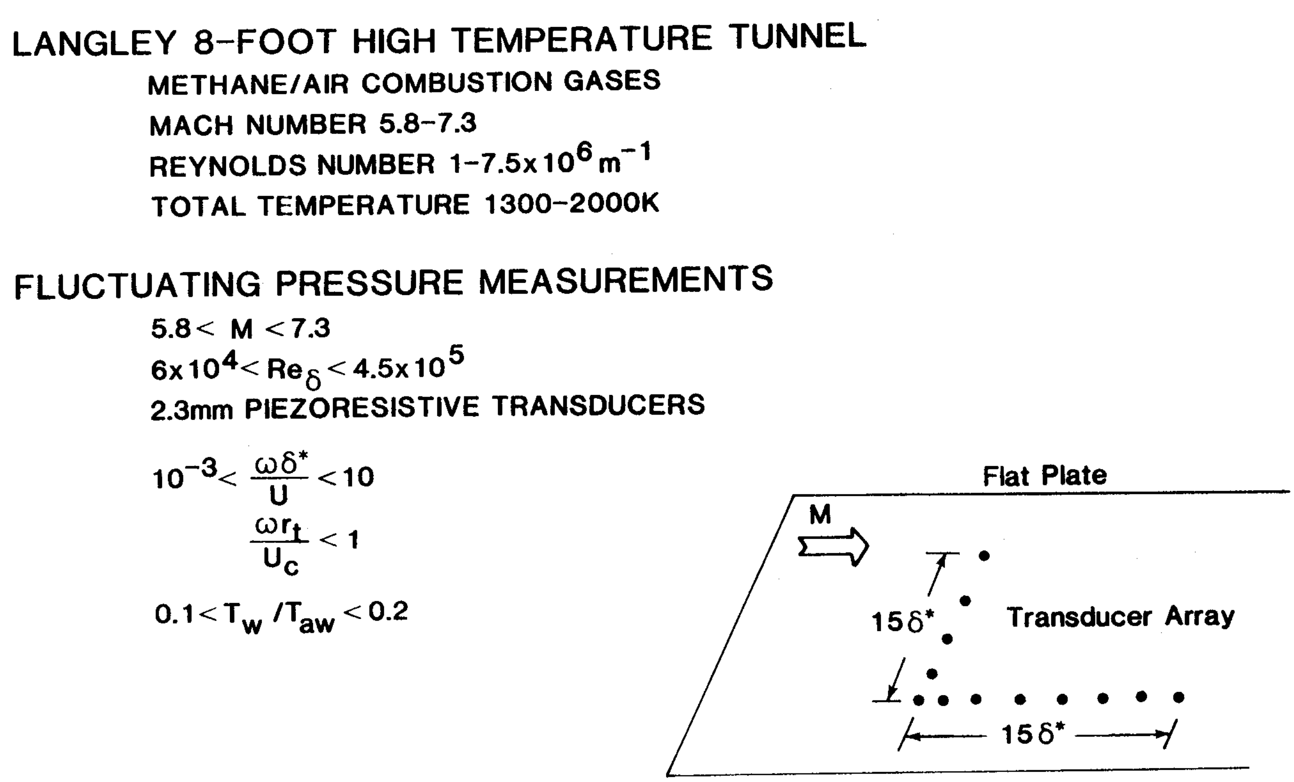

Fig. 6. Flat Plate boundary layer survey. Langley High Temperature Structures Tunnel C. W. Albertson, aerothermal investigator, and T. L. Parrott, aeroacoustic investigator.

The emerging possibility of hypersonic flight has stimulated interest in boundary layer loads research at Langley. One facility which will be used for this research is the 8-Foot High Temperature Structures Tunnel. This tunnel has a methane/air combustion chamber which exhausts through a supersonic nozzle to an open-jet test section. Some test section parameters are listed in Fig. 6. The Mach number is around 6 and the total temperature may be varied from 1300K to 2000K. A flat plate experiment is planned as a tunnel calibration. Boundary layer flow data, aerothermal loads, and aeroacoustic loads will be measured over the range of tunnel operating conditions. These data will be compared to available theory and data from other investigations. Raman[19,20] has measured the fluctuating pressures on a plate with hypersonic flow speeds and reservoir temperatures between 700 and 1200K. These data were taken with 0.5mm transducers and should provide a good comparison for the planned experiment.

The fluctuating pressure measurements will be made with commercial 2.3mm piezoresistive transducers in linear streamwise and cross-stream arrays. These transducers will be recessed $75\mu m$ and the recess will be filled with an silicone rubber insulator to form a flush surface. The installed transducers will be calibrated relative to a condenser microphone by placing a small waveguide over each array. A broadband noise source will send waves past the condenser microphone and the installed transducers to an anechoic termination, providing the calibration. Fluctuating pressure data will be acquired with digital instruments and data reduction systems. Fluctuating pressure statistics and correlations will be developed from the digital data.

Combustion noise radiated downstream is a concern, due to direct contamination of the aeroacoustic data and due to its possible influence on the boundary layer. Another problem with working in this environment is the temperature variation on the plate. After the tunnel is started, the plate is inserted into the jet. There is a large heat flux at this time and the plate temperature rises quickly. The plate is withdrawn before the transducers are destroyed. The data are taken during the time of high heat flux and variable surface temperature. This procedure, while unavoidable, complicates comparisons to data taken under stable conditions.

CONCLUSIONS

High-temperature sonic fatigue may control the design of many structural components of hypersonic aircraft. These aircraft will fly at high dynamic pressures and skin temperatures for long periods of time, which may lead to fatigue limits within a few missions. Aeroacoustic loads are largest where aerothermal loads are large—in regions of shock/boundary-layer interaction. There are no prediction methods for these loads that have been shown to be valid for the planned flight conditions. They must be defined by small-scale model experiments for each structural component where high loads are likely.

Instrumentation for measuring fluctuating pressure loads is inadequate. Transducers less than 1mm diameter are needed which can operate at temperatures up to 2000K. Because of the small model sizes, it will be necessary to take digital data at speeds up to 200kHz.

Most older experiments should be repeated with more modern instrumentation and data reduction methods in order to provide an improved basis for parametric studies of flow effects on loads. There is a need for many experiments with an orderly variation of flow parameters. These parametric studies can be expected to lead to more general prediction methods.

Separating flows and shock/boundary-layer interaction are the source of the largest fluctuating pressure loads. Aerodynamic designs should therefore attempt to avoid these flows, especially flows that are essentially two-dimensional or axisymmetric. Where they are unavoidable, detailed model studies should be conducted to define both the aerothermal and aeroacoustic loads. High-temperature fatigue resistant structures must be developed for these environments.

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