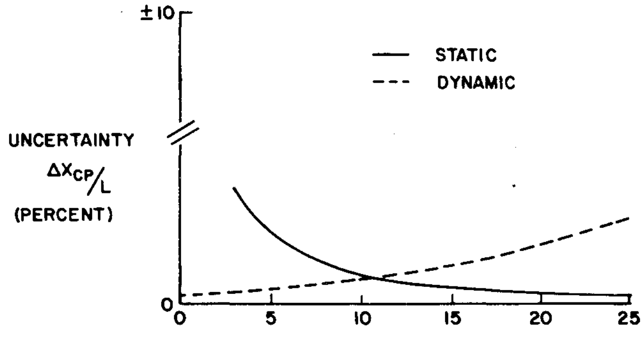

Fig. 6 Uncertainty in center of pressure vs angle of attack.

the model centerline and the tunnel flow at $\alpha = 0$ was not considered. Also, since the tests were conducted in the same facility with models of the same size, tunnel calibration uncertainties and flow conditions peculiar to this wind tunnel were not included in the uncertainty estimates. The intent here was only to illustrate the relative values and trends for the two techniques used. The uncertainty in X_{CP}/L for the static measurements increases with decreasing angle of attack. Since this characteristic depends primarily upon the balance calibration uncertainties, smaller uncertainty in the measurement of X_{CP}/L could be obtained by using a more sensitive balance. For example, this approach was used recently by Adams and Griffith,[6] who used a special force balance to investigate static stability characteristics of a sharp 5 deg cone at angles of attack up to 4 deg. For the dynamic measurements, the uncertainty in X_{CP}/L is least at $\alpha = 0$ and increases gradually with increasing angle of attack. This increase in uncertainty is possibly due to unsteady flowfield effects. The increasing "noise" observed in the θ vs t plots for the larger angles of attack supports this speculation.

In deciding which experimental technique to use to obtain static aerodynamic data, the primary consideration must be the purpose of the test. If accurate results in the small angle of attack range are not required, then a general purpose force balance is the better choice. If the emphasis is in the low angle of attack regime one has the option of using the dynamic testing technique or using a special, very sensitive, force balance. It should be noted that the damping derivatives are also obtained from the dynamic tests.[3,4]

The static balance is relatively simple to use, allows rapid data reduction, and yields data of low uncertainty for moderate to large angles of attack. Dynamic testing is more intricate and data reduction is more involved and time consuming; however, this technique provides a method of obtaining static data with low uncertainty at small angles of attack, including $\alpha = 0$.

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Flat Plate Turbulent Boundary Layers Subject to Large Pressure Fluctuations

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Nomenclature

C_f = skin friction coefficient = τ_w/q
e = hot wire output voltage
f = frequency, Hz
$F(n)$ = contribution to $\tilde{p}^2/q^2$ in frequency band Δn
$\sqrt{nF(n)}$ = $p/q(\epsilon)^{1/2}$
M = Mach number
n = frequency parameter = fw/U
p = pressure
q = freestream kinetic pressure = $\frac{1}{2}\rho U^2$
R = Reynolds number based on wire diameter = Ud/ν
R_θ = momentum thickness Reynolds number = $U\theta/\nu$
r = overheat parameter
$S_\rho, S_u, S_{T_0}, S_{\rho u}$ = hot wire sensitivity to density, velocity, total temperature, and mass flow fluctuations
T_0 = total temperature
U = freestream velocity
u = local velocity
w = width of tunnel
x = distance from the start of the slot(s)
ρu = mass flow
β = pressure gradient parameter = $\delta^* \dfrac{\mathrm{d}p/\mathrm{d}x}{\tau_w}$
δ = boundary-layer thickness at $u = 0.99U$
δ^* = boundary-layer displacement thickness

$$= \int_0^\infty \left(1 - \frac{\rho u}{\rho_1 U_1}\right) \mathrm{d}y$$

θ = boundary-layer momentum thickness

$$= \int_0^\infty \frac{\rho u}{\rho_1 U_1} \left(1 - \frac{\rho u}{\rho_1 U_1}\right) \mathrm{d}y$$

ϵ = analyzer bandwidth ratio = $\Delta f/f$
ρ = density
τ_w = wall shear stress
ν = kinematic viscosity

Superscripts

$(\ )'$ = instantaneous values of the fluctuating quantities
$(\tilde{\ })$ = root mean square value of the fluctuating quantities

Subscript

1 = freestream value

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Introduction

FLOW unsteadiness is one of the primary variables that describe the flow environment in wind tunnels. The importance of flow unsteadiness to transonic mesurements has been highlighted by recent reports.[1,2] There is evidence to suggest that the flow unsteadiness may influence the turbulent boundary layer, boundary-layer transition, bubble flows, separations, intake flows, shock interactions, wing buffeting, an flutter. It has been observed that for a subsonic turbulent boundary layer, the characteristics of the boundary layer are strongly influenced by the freestream turbulence (vorticity).[3,4] The major source of disturbance in transonic tunnels, however, is acoustic in nature, although vorticity and entropy modes of disturbances can also be present. The influence of any of the these disturbances on the behavior of the transonic turbulent boundary layer is uncertain. A recent investigation by Weeks and Hodges[5] into the effects of acoustic disturbances on a turbulent boundary layer proved somewhat inconclusive. Tests carried out by Ross and Rohne[6] ona supercritical airfoil with noise levels of $\tilde{p}/q = 0.35\%$ and 0.6% showed that the change in noise level did not affect the trailing edge and shock separation but did cause transition to move forward. This Note presents some data showing the influence of pressure fluctuations on attached subsonic and transonic turbulent boundary layers in small pressure gradients.

Experiment

The intermittent 0.101×0.101 m transonic tunnel used has a running time of 15 s, which is considered adequate for unsteady transonic testing. Only the floor of the test section was slotted. The Mach number in the test section was adjusted using wedge shaped chokes placed downstream of the test section. A single slotted liner with perforated screens and a double slotted liner were used to change the level of pressure fluctuations in the test section. The open area ratios of both the single slotted liner with screens and the double sloted liner, based on the total area of the four walls of the test section, were 4.8% and 6%, respectively. The boundary layer on the roof of the test section was chosen as the model boundary layer. Boundary-layer traverses were carried out along the roof centerline at four stations corresponding to values of $x/w = 0.25$, 3.75, 5.0, and 6.75. The boundary-layer profiles were measured using a pitot tube. A CTA anemometer, DISA 55M, was used to measure the flow unsteadiness in the boundary layer. A single 5 μm platinum tungsten probe, DISA P14, was used for all hot wire measurements. Wall pressure fluctuations were measured by Kistler miniature pressure transducers, Type 6031, mounted flush on the roof of the test section at the boundary-layer traversing stations. A Brüel and Kjaer tape recorder, Type 7003, was used to record all the signals. The direct recording channels of the tape recorder had a frequency response of 100 Hz to 50 kHz. The root mean square values and frequency spectra of the unsteady signals were obtained by a ⅓ octave band analyzer, Rion Model SA-59, an a level recorder, Rion Model LR-04.

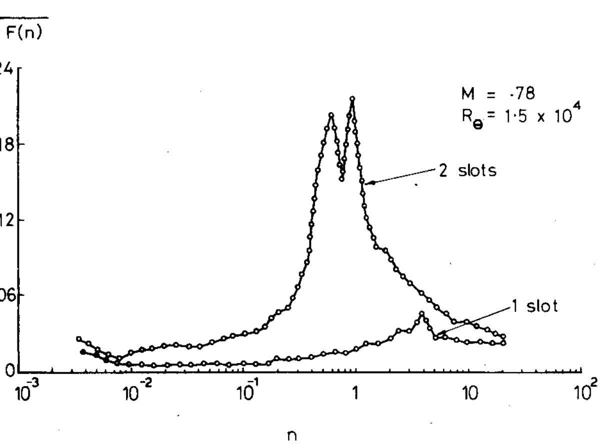

Fig. 1 Pressure fluctuation spectra.

A hot wire immersed in a fluid normal to the flow is sensitive to velocity, density, and total temperature fluctuations, so that[7]

$$\frac{e'}{e} = S_\rho \frac{\rho'}{\rho} + S_u \frac{u'}{u} - S_{T_0} \frac{T_0'}{T_0}$$

Recent correlations[8] in transonic flow show that both S_u and S_ρ are sensitive to overheat ratio r, sensor Reynolds number R, and Mach number M for $r > 0.5$ and $R > 20$. In fact, it is shown that $S_u/S_\rho \simeq 1$ and is independent of the length-to-diameter ratio of the wire for values greater than 50.

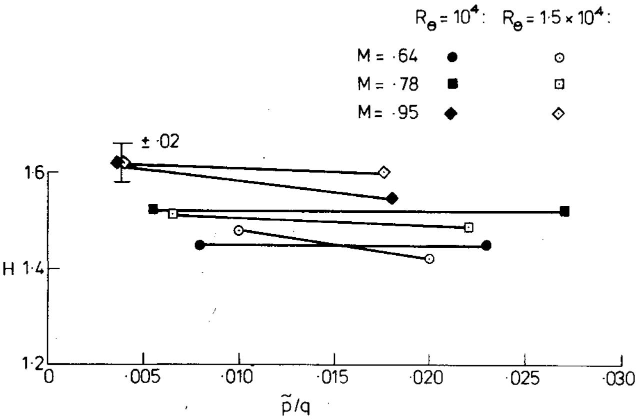

Fig. 2 Variation of boundary-layer shape factor with pressure fluctuation level.

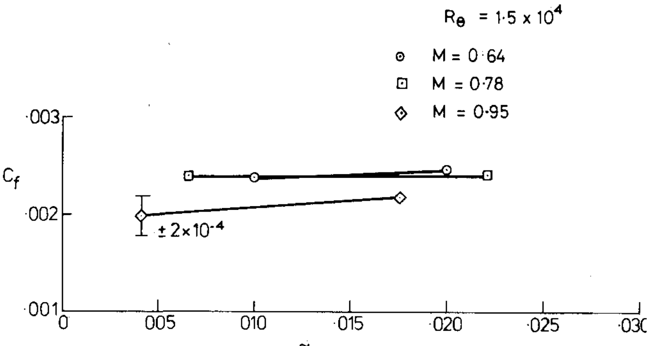

Fig. 3 Variation of skin friction coefficient with pressure fluctuation level.

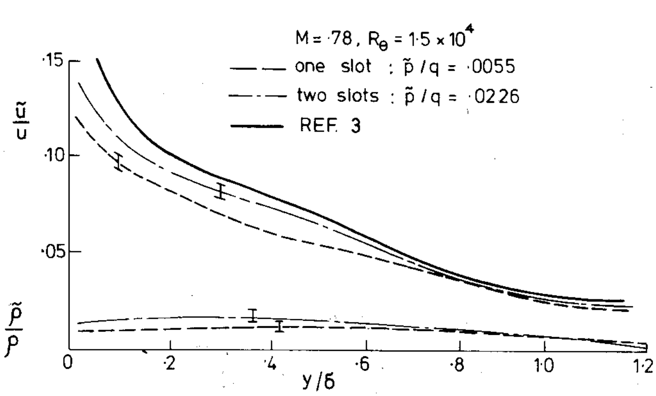

Fig. 4 Influence of $\tilde{p}/q$ on the velocity and density fluctuations within the boundary layer.

For an adiabatic turbulent boundary layer, the total temperature gradients are negligible and we may neglect total temperature fluctuations. Hence the hot wire sensitivity becomes,

$$\tilde{e}/e = S_{\rho u}\ (\tilde{\rho u}/\rho u)$$

Results

The momentum thickness Reynolds number for the test section roof was measured to be between 0.3×10^4 and 1.6×10^4. This compares with a typical value of $R_\theta \simeq 10^4$ measured at the trailing edge of models in larger test facilities. The pressure gradient parameter $\beta = \delta^*(\mathrm{d}p/\mathrm{d}x)_w$ was within ± 0.015. The values of $\tilde{p}/q$ ranged from 0.003, comparable with the noise levels in well designed closed tunnels,[2] to 0.026, representing the noisy environment typical of many large ventilated tunnels.

Using the notation adopted by Mabey,[2] some typical pressure fluctuation spectra are presented in Fig. 1 for a typical test Mach number of 0.78 and one value of $R_\theta = 1.5\times10^4$. For the double slotted liner, the peak amplitude varies from $\sqrt{nF(n)} = 0.20$ at $n = 0.6$ to $\sqrt{nF(n)} = 0.22$ at $n = 0.9$, while for the single slotted liner with the perforated screens, the peak amplitude varies from $\sqrt{nF(n)} = 0.03$ at $n = 3.2$ to $\sqrt{nF(n)} = 0.08$ at $n = 4$.

Figure 2 shows the small influence of pressure fluctuations on the boundary-layer shape factor H, and Fig. 3 shows the small influence on the skin friction coefficient C_f for a typical value of R_θ. The skin friction coefficient was obtained from the log of the wall plot for compressible turbulent boundary layers. These changes in R_θ, C_f, and H due to pressure fluctuations are an order less than those due to pure velocity fluctuations observed in low speed flows.

Figure 4 shows the typical variation of velocity and density fluctuations in the boundary layer with changes in the pressure fluctuation levels. It can be seen that a large increase in $\tilde{p}/q$ from 0.0055 to 0.0226 produces a significant increase in $\tilde{u}/u$ and $\tilde{\rho}/\rho$ over a large portion of the boundary layer. The general distribution of $\tilde{u}/u$ in the boundary layer is similar to that measured in a low speed boundary layer ($U = 10$ m/s) with a freestream turbulence level of 2.4%.[3] The temperature or density fluctuations are an order less than the velocity fluctuations and do not show the same degree of variation as $\tilde{u}/u$ but tend to have a nearly constant value through the boundary layer.

Thus, for small pressure gradients and attached flow, the turbulence within the turbulent boundary layer is significantly influenced by the external pressure fluctuations, but the corresponding changes in the mean properties of the boundary layer are fairly small.

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Numerical Solutions of Hypersonic Viscous Shock-Layer Equations

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Nomenclature

- n = coordinate measured normal to the body, nondimensionalized by the body nose radius
- n_s = shock standoff distance
- s = nondimensional surface distance coordinate
- u = nondimensional velocity component tangent to the body surface
- v = nondimensional velocity component normal to the body surface
- η = transformed normal coordinate, normalized with value behind the shock
- ξ = transformed surface coordinate, $\xi = s$

I. Introduction

THERE have been several efforts in the past to seek numerical solutions of hypersonic viscous shock-layer equations for reentry blunt-body applications.[1-3] Continued interest is motivated by the several advantages of using these equations[1] as compared to others and their wide domain of applicability; that includes the Mach numbers and Reynolds numbers commonly encountered during re-entry conditions.[3] However, most shock-layer methods encounter numerical difficulties whenever shock-layer thickness becomes large.[1] This difficulty usually manifests itself as a divergent behavior in the relaxation scheme used to update the shock shape.[4] In an attempt to overcome this problem, the results of a new approach are presented here.

The viscous shock-layer equations as applicable to blunt bodies are parabolic-hyperbolic in nature;[1] however, the effect of the unknown shock shape is to introduce an elliptic-type behavior. When the shock-layer thickness is small, the blunt-body problem is apparently of weak elliptic nature leading to the success of the past shock-layer methods.[1] For thick shock layers, a numerical technique that properly accounts for the global effect of the shock shape on a blunt body must be adopted. An efficient way to achieve this would be to employ a technique normally applied to elliptic partial differential equations with a specified downstream boundary condition. A modified version of the alternating direction implicit (ADI) finite-difference scheme is used in the present technique in order to provide the global influence of the shock shape during the numerical integration scheme.[4]

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