Supersonic Turbulent Boundary Layer in Adverse Pressure Gradient. Part I: The Experiment

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Experimental measurements of the mean profile characteristics of the supersonic turbulent boundary layer in a region of moderate adverse pressure gradient along the curved surface of an isentropic ramp model are reported. Detailed surveys of impact pressure, static pressure and total temperature were made through the boundary layer and local values of wall shear stress were obtained using the Preston tube technique. The measurements were made for a nominal tunnel nozzle setting of Mach 3.5, closely adiabatic wall, and momentum thickness Reynolds number of 1.9 to 4.2 × 10^4. In addition to the mean profile data, fluctuation data were obtained using constant temperature hotwire anemometry at one station in the region of zero pressure gradient and at one station in the region of adverse pressure gradient.

Nomenclature

c_f = skin-friction coefficient, $2\tau_w/\rho_\infty u_\infty^2$
M = Mach number
p_o = tunnel total pressure, measured in the supply header, psia
p_t = local total pressure, psia
R = local radius of longitudinal curvature of the ramp model, inch
Re_θ = momentum thickness Reynolds number, $\rho_\infty u_\infty \theta/\mu_\infty$
T = local static temperature, °R
T_o = tunnel total temperature, measured in the supply header, °R
T_t = local total temperature, °R
v = voltage proportional to electric current thru hot wire probe
x = streamwise distance, referenced to the tunnel nozzle exit, in.
y = distance normal to the local surface, in.
θ = boundary-layer momentum thickness, in.
τ = shear stress, psi or psf

Subscripts

aw = property evaluated at the theoretical adiabatic wall condition assuming a recovery factor of 0.88
w = property evaluated at the wall
δ = property evaluated at $y = \delta$
∞ = reference condition, property evaluated external to the boundary layer for $dp/dx = 0$ and at $y = \delta$ for $dp/dx > 0$

Introduction

RECENT application of numerical techniques to obtain solutions to the boundary-layer equations has focused attention on the need for accurate, detailed measurements of boundary-layer characteristics. The proposed numerical procedures for the compressible turbulent boundary layer cannot be properly evaluated because the data available are not sufficiently accurate, detailed or comprehensive. For example, the data obtained by Winter, Smith, and Rotta[1] on a waisted body of revolution have been frequently used to evaluate the calculation procedures. For these data, neither total temperature nor the static pressure normal to the model surface was measured.

Data existing in the literature for the supersonic turbulent boundary layer in an adverse pressure gradient are particularly poor. This is due to difficulty experienced in obtaining reliable test conditions and due to the fact that the measurements reported have been incomplete. Measurements of wall shear stress have not been reported for the configuration of this experiment and measurements of the static pressure profile through the boundary layer have been either omitted or of insufficient detail and accuracy. References 2–6 report measurements of the supersonic turbulent boundary-layer characteristics along a surface with longitudinal concave curvature. This flow configuration is of interest for design of supersonic engine inlets and flare stabilized missiles.

This experiment has been conducted with the objective of obtaining a good quality flow that could be measured accurately and would enable a complete set of measurements to be taken in sufficient detail to reveal the physics of the mean flow. Beyond the experimental objective, the intent is to use the data to further basic understanding of the supersonic turbulent boundary layer.

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The Experiment

Test Facility

The experiment was performed in Supersonic Wind Tunnel No. 2 of the Ballistic Research Laboratories, Aberdeen Proving Ground, Md. This is a continuous operating, asymmetric, flexible nozzle research tunnel. The test section size is 6 in. × 6 in. and extends for 22 in. beyond the nozzle exit. The test section boundary layer is a fully developed turbulent boundary layer approximately 1 in. thick that has developed naturally along a smooth flat surface.

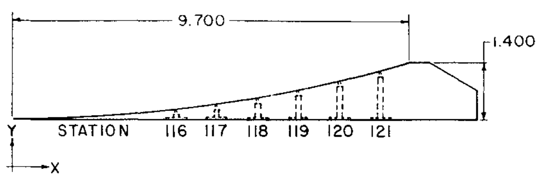


Fig. 1 Isentropic ramp model.

Model

An isentropic ramp model, shown to scale in Fig. 1, was used to create the region of adverse pressure gradient. The coordinates of the model surface and local turning angle at the test stations are tabulated in Table 1. The model was designed to create a streamwise pressure gradient severe enough to measure accurately but not severe enough to cause the formation of a shock wave in the vicinity of the measuring stations. The contour was calculated using Prandtl-Meyer turning angles. No attempt was made to adjust the model contour for boundary-layer growth.

Table 1 Model coordinates for the isentropic ramp model[a]

Model Coordinates					
X	Y	X	Y	X	Y
0	0.010	3.750	0.224	6.655	0.673
0.908	0.023	3.994	0.252	6.895	0.722
1.135	0.030	4.237	0.282	7.135	0.772
1.362	0.038	4.480	0.313	7.374	0.824
1.588	0.049	4.723	0.346	7.613	0.877
1.815	0.061	4.965	0.381	7.852	0.932
2.042	0.074	5.207	0.418	8.091	0.989
2.286	0.090	5.449	0.455	8.362	1.055
2.531	0.108	5.691	0.496	8.632	1.124
2.775	0.128	5.932	0.538	8.902	1.194
3.019	0.149	6.174	0.581	9.172	1.266
3.263	0.172	6.414	0.627	9.441	1.341
3.507	0.197				

Coordinates of test stations and local turning angle of the surface[b]

Station	X	Y	ϕ
116	3.994	0.242	6.75
117	5.013	0.378	8.13
118	6.029	0.545	9.85
119	7.015	0.737	11.48
120	8.019	0.962	13.24
121	9.010	1.213	14.65

[a] Dimensions are in inches.
[b] Turning angle in degrees.

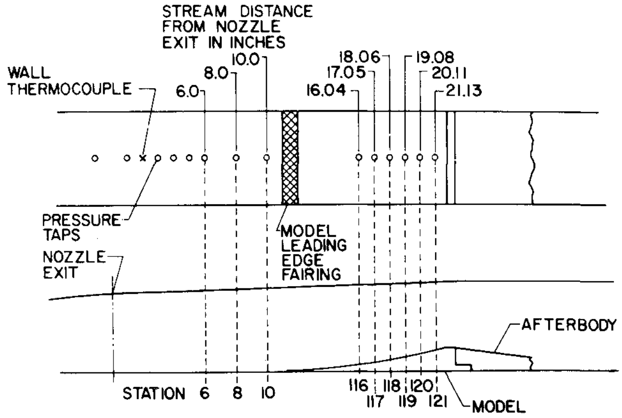


Fig. 2 Location of model relative to the nozzle exit and test station identification.

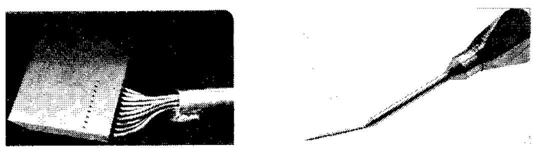


Fig. 3 Static pressure probes.

The front edge of the model had a thickness of 0.010 in. When mounted in the test section, the front edge was glued to the floor of the tunnel and lacquer putty used to form a smooth transition from the tunnel floor to the surface of the model.

The model was instrumented with six 0.040-in.-diam static pressure holes drilled normal to the local surface at 1-in. intervals along the centerline. In addition, eleven 0.025-in. diam static pressure taps were located off centerline at the last three stations on the model. The position of the model relative to the nozzle exit is illustrated in Fig. 2. Also shown are the location and designation of the test stations and the wall thermocouple which is positioned 0.050 in. below the test surface of the tunnel.

Instrumentation

General

All pressures were read using strain gage transducers that are calibrated within $\pm 0.25\%$ of their individual full-scale range. Temperatures were measured using iron-constantan thermocouples. The thermocouples were connected to an electric cold junction and the output of the cold junction was read on an electronic voltmeter. The recovery temperature probe was sensitive to the temperature fluctuations within the turbulent boundary layer. To overcome these fluctuations, the output of the thermocouple was processed through a simple RC filter before being read on the electronic voltmeter.

Impact Probe

Several impact probes with different tip openings were tried. The probe used had a tip opening that was 0.060 in. $\times$ 0.003 in. with a lip thickness of approximately 0.001 in.

Static Pressure Probes

Static pressure probes were constructed in two configurations, flat plate and cone-cylinder. These probes are shown in Fig. 3. The flat plate probe was a 0.125-in. thick steel plate with a 20° razor-sharp leading edge. Ten 0.040-in. diam holes were located 1.125 in. from the leading edge and spaced 0.125 in. apart. The bottom edge of the probe was curved to enable it to seat flush with the model surface. This configuration was considered desirable due to the angularity of the flow over the curved surface of the model.

The cone-cylinder probe was constructed with a 0.050-in.-diam cylindrical body. The half-angle of the cone portion was 10°. Two static holes of 0.0135 in. diam were located 10 diam downstream of the start of the cylindrical portion and 5 diam upstream of the bend where the tube connects to the probe support. The static holes were located on the sides of the cylindrical body. This probe was calibrated at angle of attack from 0° to 10° and was found to be very sensitive to angle of attack. The deviation of the measured pressure from the reference pressure ranged from -1.58% at zero angle of attack to -26% at 10° angle of attack. Within the boundary layer, the angle of attack experienced by this probe is expected to be less than 3°. The maximum deviation for this condition was found to be -2.3%.

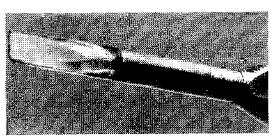
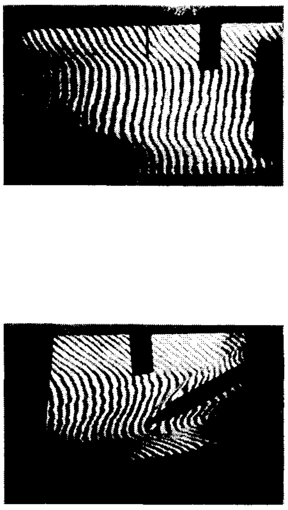


Fig. 4 Recovery temperature probe.

Recovery Temperature Probe

The probe used to measure the total temperature through the boundary layer is shown in Fig. 4. This probe was constructed of lexan plastic and had a 0.005-in. diam iron-constantan thermocouple located at the center of the wedge tip. The lead wires were placed in a groove machined along the front edge and sides of the plastic wedge and covered with epoxy cement. Only the spot welded junction of the thermocouple was exposed to the flow. The recovery factor for this probe was established by a separate test in Supersonic Wind Tunnel No. 1 of the Ballistic Research Laboratories for a Mach number range from 3.5 to 1.5. The recovery factor was extrapolated from Mach 1.5 to 0 according to the trend of the data indicated in Fig. 11a of Ref. 7.



Zero pressure gradient turbulent boundary layer.

Adverse pressure gradient turbulent boundary layer.

Fig. 5 Single plate laser interferograms.

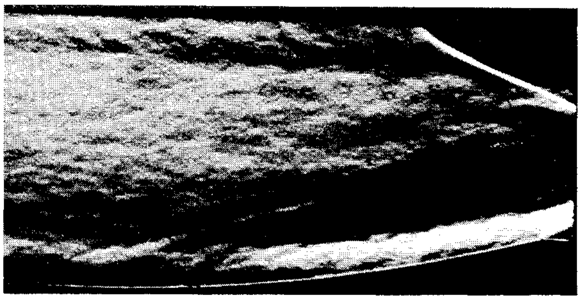


Fig. 6 Schlieren of flow over ramp model.

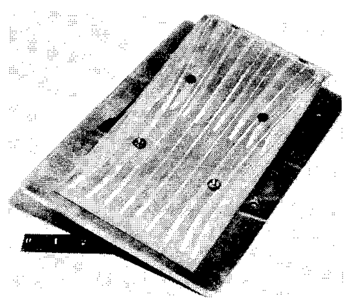


Fig. 7 Oil flow test.

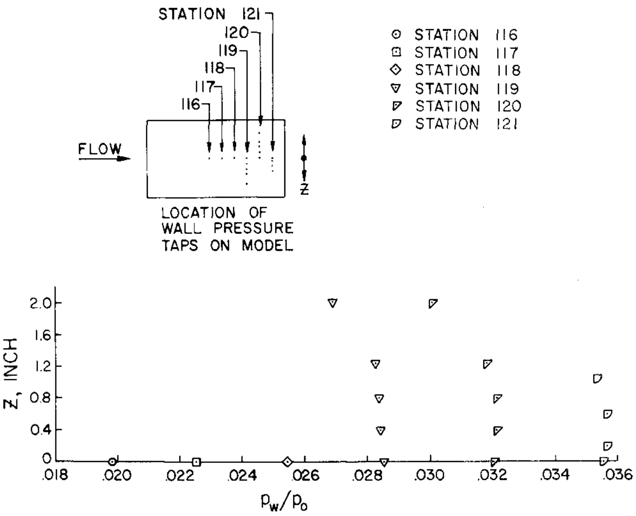


Fig. 8 Wall static pressure off centerline.

Flow Visualization

The sidewalls of the test facility are glass permitting observation of the flow from the nozzle throat to the end of the test section. Flow visualization has been obtained using single plate laser interferometry and schlieren photography. Figure 5 shows interferograms of the flow upstream of the ramp model and midway along the surface of the ramp model. Figure 6 shows a schlieren picture of the flow over the ramp model. The weak shock formed at the leading edge of the ramp model is visible as is the turbid structure within the boundary layer and the irregular outer edge. Also apparent is the decreasing boundary-layer thickness as the flow is compressed by the increasing static pressure.

Two-Dimensionality

The two-dimensionality of the flow is of great concern in a facility of the type used for these tests. This concern is intensified when a pressure gradient is imposed on the flow. Two tests were conducted in an effort to evaluate the departure from two-dimensionality for the conditions of this test: 1) oil flow visualization, and 2) wall pressure measurements off centerline at the last three stations on the ramp model.

A picture showing the result of oil flow development over the ramp model is shown in Fig. 7. Streamlines along the surface of the model near the centerline do not diverge appreciably; however, considerable divergence of streamlines near the side walls does take place.

Wall pressure measurements on a line normal to the centerline at the last three stations on the ramp model are shown in Fig. 8. These measurements establish that a region exists in the flow over the ramp model in which the lateral pressure gradient is very small compared to the streamwise pressure gradient.

Procedure

The profile surveys of impact pressure, static pressure and recovery temperature were made during separate test runs since the survey mechanism could accommodate only one probe at a time. The survey mechanism was designed to traverse the boundary layer at an angle perpendicular to the local surface. The probe position was measured using a cathetometer out to a distance of $\simeq 0.30$ in. from the wall; from there on the position was established by turns of the mechanical drive mechanism.

Data were obtained for three values of tunnel total pressure at a tunnel total temperature of 560°R. Tunnel total pressure variation was less than $\pm 0.25\%$ and the tunnel total temperature was controlled within ± 1°F or $\pm 0.18\%$. Assuming a wall recovery factor of 0.88, the ratio $(T_w - T_{aw})/T_{aw} \simeq 0.021$ shows that the wall was closely adiabatic.

Adjustment of the Nozzle Contour

A great deal of care was taken in the adjustment of the nozzle to produce a uniform flow in the test section. A flow with maximum variation of ± 0.01 Mach number was finally achieved. The Mach number distribution as determined from measurements of wall static pressure and tunnel total pressure without the model in the test section is shown in Fig. 9.

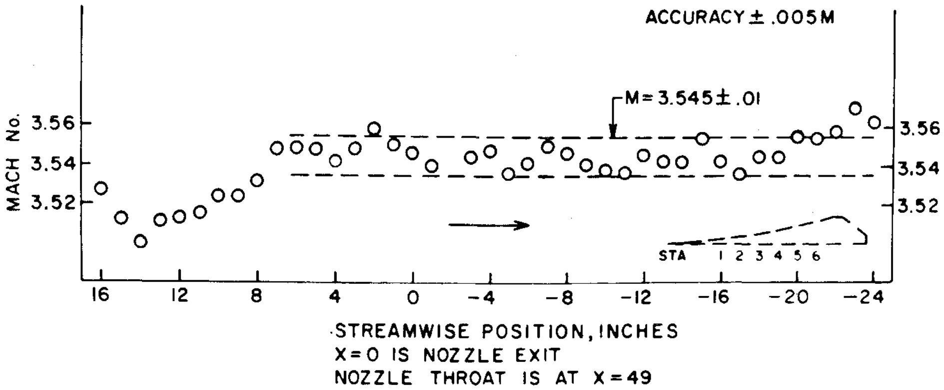


Fig. 9 Test section Mach number distribution.

Boundary-Layer Profiles

Profile Calculations

The profile characteristics were calculated from the experimental data using perfect gas relations and the Rayleigh Pitot formula. The calculations were performed on the BRL digital computer. Tables of experimental measurements of impact pressure, static pressure and recovery temperature vs distance normal to the surface and recovery factor vs Mach number were introduced into the computation routine. Calculations were performed at the value of y corresponding to each measurement of impact pressure.

Integral properties of the boundary layer were calculated using a trapezoidal numerical integration routine. The integral properties calculated were displacement thickness, momentum thickness, energy thickness, enthalpy thickness and velocity thickness.

Experimental Profiles

Examples of the impact pressure profiles obtained are shown in Fig. 10. The profile at Station 8, $dp/dx = 0$, increases monotonically to a constant freestream value while the profiles over the ramp model increase sharply to a maximum and then decrease almost linearly with y beyond the boundary-layer region.

The static pressure measurements were rather troublesome to resolve. This was caused by probe-wall interference and angle of attack sensitivity for the cone-cylinder probe; and interference

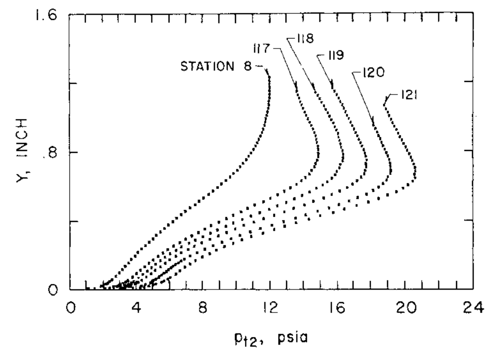


Fig. 10 Impact pressure profiles.

with the weak shock propagating from the leading edge of the model and yaw deflection for the flat plate probe.

The data obtained using the flat plate probe are considered to be of qualitative value in establishing the trend of the static pressure profile in the vicinity of the wall. The data obtained with the cone-cylinder probe corrected by +1.58% are considered to be accurate beyond the region of probe-wall interference out to the edge of the boundary layer.

Preliminary profile calculations using uncorrected cone-cylinder static pressure data revealed a Mach number greater than the tunnel nozzle setting beyond the region of the boundary layer for the first station on the ramp model. This led to calculating the

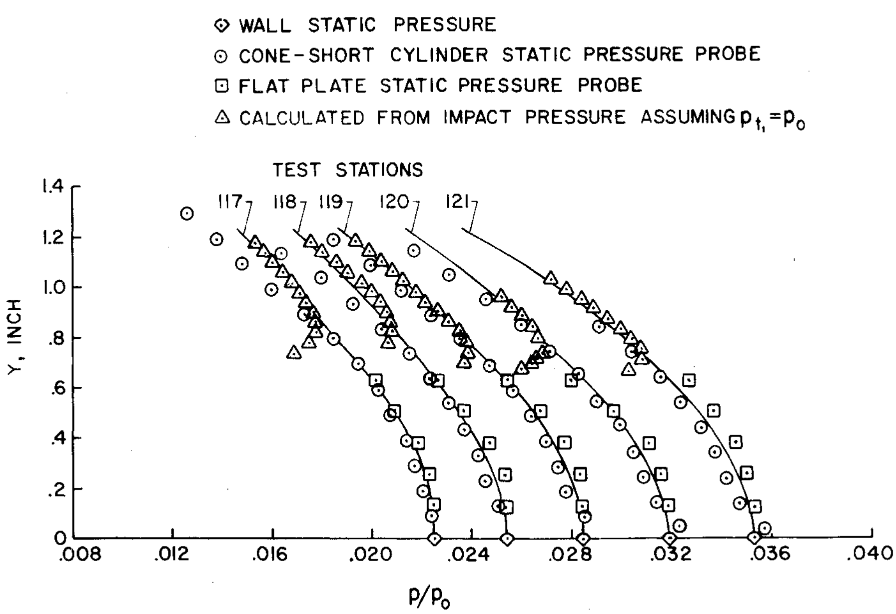


Fig. 11 Static pressure profiles.

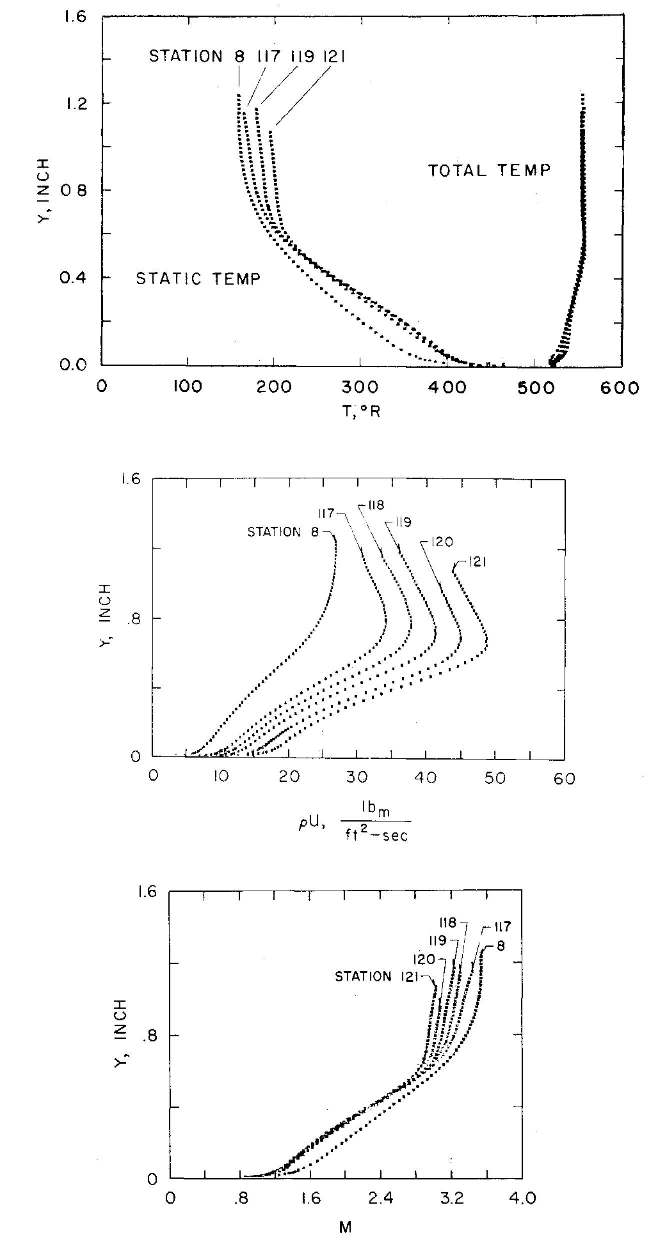


Fig. 12 Temperature, mass flux, and Mach number profiles.

static pressure from impact pressure measurements assuming a constant total pressure equal to the tunnel total pressure. The profile resulting from this calculation is shown in Fig. 11 along with corrected cone-cylinder and flat plate static pressure probe data. The solid line represents a fairing of what is felt to be the best data.

Examples of temperature, mass flux and Mach number profiles are shown in Fig. 12. The static and total temperature profiles indicate that the total temperature profiles are much less affected by the different flow configurations than the static temperature profiles. The static temperature profiles have a linear variation within the boundary layer for the flow over the ramp model as opposed to the more full profile for the zero pressure gradient flow.

The mass flux profiles indicate that the mass flux increases to a maximum near the edge of the boundary layer then decreases linearly with y beyond the boundary layer for the flow over the ramp model. This is in sharp contrast to the flow upstream of the ramp model in which the mass flux increases monotonically to a constant freestream value.

The Mach number profiles show that the Mach number increases monotonically for both regions of flow, $dp/dx = 0$ and $dp/dx > 0$. A freestream region of constant Mach number is indicated for the zero pressure gradient flow; however, for the flow over the ramp model, a region in which the Mach number increases uniformly exists beyond the extent of the boundary layer.

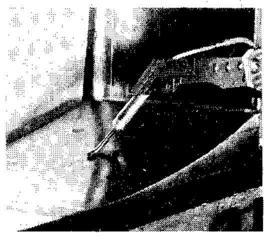


Fig. 13 Preston tube mounted in position over the ramp model.

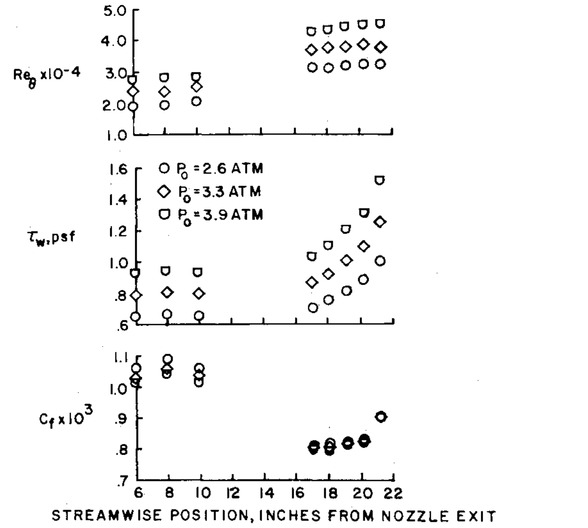


Fig. 14 Skin friction and momentum thickness Reynolds number.

Skin-Friction Measurements

Wall shear stress has been measured using the Preston tube technique. A Preston tube is a circular impact tube mounted flush with the model surface and sized to lie within the logarithmic portion of the law of the wall velocity profile. The tube used here was 0.125-in. in diameter and is shown in position over the ramp model in Fig. 13.

Tests have been conducted by various researchers[8–11] for the purpose of evaluating the Preston tube as a means of measuring local skin friction. These tests have been conducted over a range of Mach numbers from subsonic to hypersonic and have included the effects of heat transfer and both favorable and unfavorable pressure gradient. These tests indicate that the Preston tube can be expected to yield measurements of wall shear stress within an accuracy of $\pm 15\%$ for the conditions of this experiment.

The value of wall shear stress was calculated using Eq. (2b) of Ref. 10 because it was established from data obtained in a test facility similar to the one used for this experiment. Also, τ_w is calculated from measurements of wall properties so no uncertainty is introduced into the wall shear stress due to the determination of the freestream conditions. This is particularly desirable due to the uncertainty in defining the freestream conditions for the flow over the ramp model. A summary of the wall shear stress data is shown in Fig. 14. The data indicate that the wall shear stress increases in the streamwise direction in the region of adverse pressure gradient. The value of the skin-friction coefficient for the flow over the ramp model is 20% less than that for the zero pressure gradient flow upstream.

Hot Wire Measurements

Measurements were made at two stations in the test section using constant temperature hot wire anemometry: in the region of zero pressure gradient at station 8 and in the region of adverse pressure gradient at station 119. A tungsten wire coated with platinum of 0.00025 in. diam and 0.10 in. length was used.

An example of the fluctuation and mean data is shown in Fig. 15. Sufficient data have not been accumulated to properly obtain turbulence intensities so the data are plotted as the ratio of the local value of the measurement to a reference value of that measurement. The same value of v_0 and $\langle v_0'^2 \rangle^{1/2}$ was used to normalize the data for both sets of measurements. The values used were 0.15 and 0.004 v, respectively, and were chosen arbitrarily. Coincidentally, the mass flux at the edge of the boundary layer was closely the same for the two profiles measured.

The data for $dp/dx = 0$ are compared to turbulence intensity measurements reported by Kistler[12] that were taken in the same tunnel for the same Mach number and tunnel total pressure but for a different value of tunnel total temperature. A similarity in the shape of the profiles is indicated.

The data for $dp/dx > 0$ indicate a fluctuating signal strength comparable to that obtained at station 8 near the wall and at the edge of the boundary layer. Within the boundary layer, the strength of the fluctuating signal increases to a maximum near the edge of the boundary layer before decreasing sharply to the low freestream value. This is quite unlike the almost constant value of the fluctuating signal through the boundary layer for the data obtained in the region of $dp/dx = 0$. In both cases the edge of the boundary layer is distinct and a uniform low-signal level is obtained outside the boundary layer.

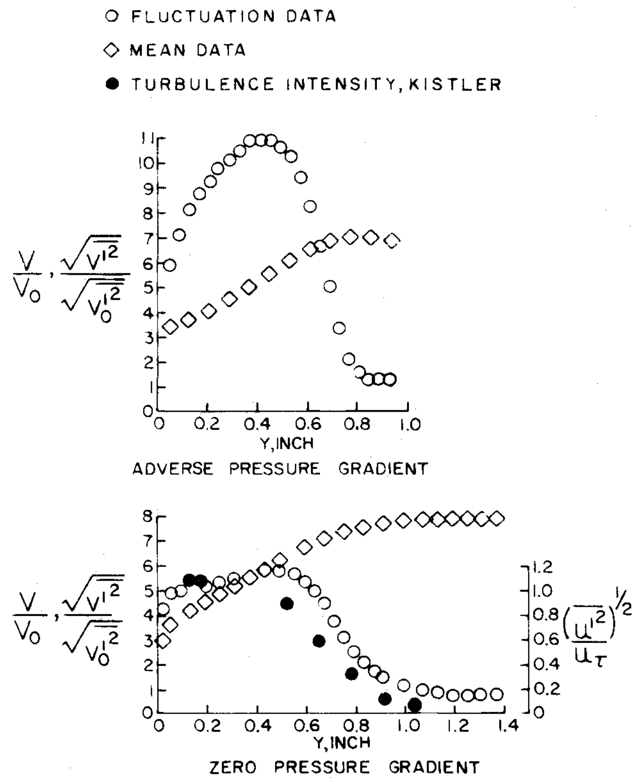


Fig. 15 Hot wire measurements.

Comments on the Experimental Data

The tunnel total temperature was adjusted to 560°R and maintained within ±1°F during all test runs. The freestream total temperature in the test section was found to range from 551 to 554.6°R. This discrepancy is accounted for by a 1% uncertainty in the recovery factor for the total temperature probe. This is likely a result of determining the recovery factor in a different test facility than that used to obtain the profile data.

The uncertainty in calculating the profile characteristics from the experimental data has been estimated and is shown in Table 2. No attempt has been made to include the uncertainty in the measured quantities caused by the turbulent fluctuations. A more detailed description of the experiment as well as a complete data tabulation are available in Refs. 13 and 14.

Table 2 Estimated accuracy of the profile characteristics

Profile parameter	Estimated error, %
Static pressure	+0.25, −1.25
Mach number	+1.00, −0.50
Total temperature	+0.25, −1.25
Static temperature	+0.75, −2.25
Velocity	+1.40, −1.60
Density	+2.50, −2.00
Mass flux	+3.90, −3.60
Wall shear stress	±15

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