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Shock impingement near mild hypersonic expansion corners

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The University of Texas at Arlington, 1992
SHOCK IMPINGEMENT NEAR MILD HYPERSONIC EXPANSION CORNERS

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SHOCK IMPINGEMENT NEAR MILD HYPERSONIC EXPANSION CORNERS

by

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Presented to the Faculty of the Graduate School of The University of Texas at Arlington in Partial Fulfillment of the Requirements for the Degree of DOCTOR OF PHILOSOPHY

THE UNIVERSITY OF TEXAS AT ARLINGTON
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This dissertation is dedicated to my mother, Mrs. Ya-Shu Fu Chung, for her love and encouragement.

November 18, 1992
ABSTRACT

SHOCK IMPINGEMENT NEAR MILD HYPersonic EXPANSION CORNERS

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Kung-Ming Chung, Ph.D.
The University of Texas at Arlington, 1992

Supervising Professor: Frank K. Lu

Experiments were performed in the UTA shock tunnel at Mach 8 and a Reynolds number of 10.2 million/m. A cold-wall, turbulent boundary layer developed along a long flat plate installed in the test section. An expansion corner of 2.5° or 4.25° was located at 0.77 m (30.25 in) from the leading edge of the flat plate. The test surface was instrumented for surface pressure measurement using flush-mounted, fast-response pressure transducers. Interactions were generated by full-span external shock generators in the form of a 2° or 4° degree sharp wedge.

A hypersonic similarity parameter was identified which scaled the downstream influence of an expansion corner, indicating that the flow can be treated as primarily rotational and inviscid. The surface pressure fluctuations were severely damped by the expansion. The pressure distributions with shock impingement were strongly influenced by the presence of the expansion corners. The upstream influence scale decreased as the shock impinged further downstream of the corner. The unsteadiness of the shock wave/boundary layer interactions was characterized by an intermittent
region together with a local rms pressure peak near the upstream influence line. Also, the peak rms pressure fluctuations increased with a larger overall inviscid pressure ratio.
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NOMENCLATURE

$A, B$  
van Driest parameters

$a$  
speed of sound

$C$  
law-of-wall constant

$C_v$  
constant volume specific heat

$\bar{D}_c$  
normalized convection distance, $\tau U_c/\delta_o$

$d$  
diameter of transducer's sensing element

$f$  
frequency

$K$  
hypersonic similarity parameter

$L$  
distance between transducers

$M$  
Mach number

$\dot{m}$  
mass flow rate

$N$  
total number of data points

$p$  
pressure

$p_F$  
final inviscid pressure

$q_{oo}$  
dynamic pressure

$r$  
radius of pressure transducer

$Re$  
Reynolds number

$R_{pp}$  
space-time correlation
<table>
<thead>
<tr>
<th>Symbol</th>
<th>Definition</th>
</tr>
</thead>
<tbody>
<tr>
<td>rms</td>
<td>root-mean-square pressure</td>
</tr>
<tr>
<td>$T$</td>
<td>temperature</td>
</tr>
<tr>
<td>$t$</td>
<td>time</td>
</tr>
<tr>
<td>$U, u$</td>
<td>velocity</td>
</tr>
<tr>
<td>$U_c$</td>
<td>convection velocity</td>
</tr>
<tr>
<td>$U^*$</td>
<td>van Driest generalized velocity</td>
</tr>
<tr>
<td>$W$</td>
<td>interval width</td>
</tr>
<tr>
<td>$W(y/\delta)$</td>
<td>Coles universal wake function</td>
</tr>
<tr>
<td>$x$</td>
<td>streamwise distance along expansion corner</td>
</tr>
<tr>
<td>$x_{sh}$</td>
<td>shock impinging position</td>
</tr>
<tr>
<td>$\bar{x}_{sh}$</td>
<td>normalized shock impinging position, $x_{sh}/\delta_o$</td>
</tr>
<tr>
<td>$x_u$</td>
<td>upstream influence scale</td>
</tr>
<tr>
<td>$\bar{x}_u$</td>
<td>normalized upstream influence scale, $x_u/\delta^*$</td>
</tr>
<tr>
<td>$y$</td>
<td>vertical distance measured from test surface</td>
</tr>
<tr>
<td>$y^+$</td>
<td>wall coordinate, $yu_r/\nu_w$</td>
</tr>
<tr>
<td>$\alpha$</td>
<td>expansion corner angle</td>
</tr>
<tr>
<td>$\beta$</td>
<td>shock generator angle of attack</td>
</tr>
<tr>
<td>$\tilde{x}$</td>
<td>hypersonic viscous similarity parameter</td>
</tr>
<tr>
<td>$\Delta U^+$</td>
<td>wake component</td>
</tr>
<tr>
<td>$\Delta p$</td>
<td>pressure drop across Prandtl-Meyer expansion</td>
</tr>
</tbody>
</table>
$\Delta t$ sampling period

$\delta$ boundary layer thickness

$\delta^*$ displacement thickness

$\gamma$ intermittent factor or specific heat ratio

$\kappa$ von Karman constant

$\mu$ dynamic viscosity

$\nu$ kinematic viscosity

$\phi$ shock angle

$\Pi$ wake strength parameter

$\sigma_p$ standard deviation of pressure fluctuations

$\tau$ time delay

$\tau_o$ wall shear stress at the corner

$\theta$ momentum thickness

$\varrho$ density

$\xi$ separation distance between transducers

$z$ spanwise distance

$Z$ standardized variable, $(p_w - \bar{p}_w)/\sigma_p$
Subscripts

1  driven tube, upstream of interaction
4  driver tube
aw  adibatic wall
D  downstream influence
e, δ  boundary-layer edge
i  number of data points within each interval
le  leading edge of flat plate
max  maximum
min  minimum
o  stagnation conditions, upstream condition
pit  Pitot probe
ref  reference
sat  saturation
s  constant in Sutherland’s law, = 110 K
tr  transition
w  wall condition
∞  incoming freestream conditions
Superscripts

\((\cdot)')\quad \text{fluctuating value}
CHAPTER 1

INTRODUCTION

1.1 General Introduction

Shock wave/boundary layer interactions have been the subject of considerable research, both fundamentally and practically. Since the pioneering work by Ferri in 1939, numerous configurations at various conditions have been investigated [22]. Primarily, the renewed development of supersonic and hypersonic flight vehicles has brought about a need for a detailed understanding of the interactions and necessitates accurate prediction of the characteristics of the flowfield.

Examples of shock wave/boundary layer interactions can be found in flows past transonic airfoils, engine inlets, flaps, wing-body junctions, and in high-speed turbomachinery. A typical feature on a transonic airfoil is that associated with a "quasi-normal" shock on the upper surface. Once separation occurs, the sudden development of a large separation bubble dramatically changes the flow structure and can cause a large shift in the pitching moment. The interaction between an oblique shock wave and the boundary layer in a supersonic engine inlet affects the flowfield and engine performance. Also on a deflected flap, the interactions can result in a loss of control effectiveness, while the impinging shock generated from by the fuselage creates high heating levels further aft.

The interaction of a shock wave and a boundary layer promotes flow separation when the adverse pressure gradient exceeds a certain strength. Shock-induced
separation generally alters the pressure field and the subsequent flow reattachment gives rise to a high heating rate. The alteration of the pressure field can significantly reduce both internal and external aerodynamic performance while the local high heating values can cause catastrophic failure [47]. Therefore, detailed information on the interactions must be obtained to assess aerodynamic performance and minimize adverse shock-induced effects.

So far the research has yielded a good insight into the structure of the interactions. However, many physical aspects remain unanswered. For instance, the physics of the interactions based on the mean measurements is not completely understood due to the apparent unsteady nature of the interactions. Such unsteadiness arising from the large-scale oscillations of the shock can cause large pressure fluctuations, whereby intense vibration may result in fatigue of the aircraft's structure. The understanding of this unsteadiness is important in improving the prediction of aerodynamic performance and in the development of numerical modeling.

Further, with the development of advanced high-speed aircraft, shock wave and boundary layer interactions have recently been the focus in both experimental and computational fluid dynamics (CFD). Basic experimental research is required to understand the fluid dynamics of the phenomena and to validate CFD codes. Once validated, CFD can be very useful in providing great insights into the nature of flowfield, especially when it is difficult to carry out the measurements due to the limitations of wind-tunnel facilities. Thus, progress is still expected from both experimental and computational research.
1.2 Present Investigation

In the present study, the research program is aimed toward enhancing the understanding of shock wave/boundary layer interactions with emphasis on hypersonic inlet design. Fundamental experiments, namely impinging shock on expansion corners, are performed. This research is in support of scramjet engine inlet design pertaining to hypersonic vehicles such as the National AeroSpace Plane (NASP). The highly complex inlet flowfield needs to be properly understood before the performance of the rest of NASP scramjet propulsion system can be predicted.

The experiments were performed in the UTA shock tunnel facility. A Mach 8 flow was generated at a Reynolds number of 10.2 million per meter. A cold-wall turbulent boundary layer was naturally developed along a long flat plate installed in the test section. Expansion corners of 2.5° and 4.25° were located at 0.77 m (30.25 in.) from the leading edge of the plate, where the undisturbed boundary layer thickness $\delta_0$ was about 1.27 cm (0.5 in.). A shock was impinged by a 2° or 4° sharp wedge. Also, the shock impingement position was adjusted to be at the expansion corner, or one boundary layer thickness upstream and downstream of the corner. Surface pressure measurements using flush-mounted, fast-response pressure transducers and Pitot pressure surveys were performed. The effects of expansion corners on the surface pressure distribution and boundary layer development are examined. The unsteadiness of the interactions was also analyzed through measurements of surface pressure fluctuations.

This dissertation is organized as follows. A brief review of the literature is given in Chapter 2. Chapter 3 describes the test facility, data reduction and the incoming undisturbed boundary layer. Experimental results are discussed in Chapter 4, followed by conclusions in Chapter 5.
CHAPTER 2

LITERATURE REVIEW

2.1 Surface Pressure Fluctuations

Surface pressure fluctuations in turbulent boundary layers are of fundamental importance in fluid mechanics and other applications, e.g. flow-induced vibration [17]. Generally speaking, the majority of surface pressure fluctuations is generated by local, near-wall motions. Intermittent ejections of the viscous sublayer and sweep of high-speed fluid play a dominant role in turbulent energy production and dissipation [59]. Therefore, the measurement of surface pressure fluctuations can provide some information about the dynamics of the boundary layer [60]. Also, fluctuating pressure fields, as a forcing function to the structural model, are essentially required for the modeling of sound generation and flow-induced vibration.

There are extensive investigations of surface pressure fluctuations of incompressible as well as compressible flows [10, 17, 57]. Some of these studies are concerned with the prediction of the pressure fluctuation level. Raman’s study at Mach 5.2, 7.4 and 10.4 for a flat plate flow shows that the rms pressure is a function of the dynamic pressure $q_\infty$ and Mach number $M_\infty$. Raman found that $\sigma_p/q_\infty$ varies from 0.4 to 0.1 percent for a dynamic pressure range of 6.8 to 66 kPa. Further, $\sigma_p/q_\infty$ decreases with increasing Mach number and is a weak function of $Re_\delta^+$. For a given Mach number, $\sigma_p/q_\infty$ varies as $Re_\delta^{-0.3}$ which implies that the rms pressure can be represented in terms of the wall shear stress. Based on the above observation, Laganelli
and Martellucci [49] examined the available surface pressure fluctuation data in both incompressible and compressible flows. They noted that there is a large scatter in the data as the Mach number increases. They thought that the shear stress and wall temperature may play a important role at high Mach number. Laganelli and Martellucci [49] gave semi-empirical formulas for the surface pressure fluctuations as:

\[
\frac{\sigma_P}{q_\infty} = \frac{0.006}{0.5 + T_w/T_{aw}(0.5 + 0.09M_e^2) + 0.04M_e^{2.164}}
\]  \hspace{2cm} (2.1)

or

\[
\frac{\sigma_P}{p_w} = \frac{0.003\gamma M_e^2}{0.5 + (T_w/T_{aw})(0.5 + 0.09M_e^2) + 0.04M_e^{2.164}}
\]  \hspace{2cm} (2.2)

Equation (2.1) is plotted in Fig. 2.1 together with Raman’s [57] data. At high Mach number, the effect of wall temperature \(T_w/T_{aw}\) appears to be significant. Also for a given \(T_w/T_{aw}\), the pressure fluctuation \(\sigma_P/p_w\) is a function of \(M_e\) as shown in Equation (2.2), and, in fact, for \(T_w/T_{aw} = 0\), \(\sigma_P/p_w \propto M_e^2\). In other words, the intensive pressure fluctuations in a hypersonic flow may affect the integrity of the vehicle’s structure.

Further, it is thought that the surface pressure fluctuations are due to large-scale structures in the turbulent flow. For a “frozen” turbulent boundary layer, Taylor’s hypothesis suggests that the turbulent eddies are convected downstream at a “convection velocity” \(U_c\) which can be obtained from the cross-correlation of surface pressure fluctuations. An early study by Bull [10] at Mach 0.3 and 0.6 shows that the convection velocity \(U_c/U_\infty\) is a function of the separation distance between transducers as shown in Fig. 2.2a. Also, the study of Speaker and Ailman [67] at supersonic flow indicates the same trend. This trend implies that the high-frequency eddies prevailing near the wall travel downstream slower than the larger, low-frequency eddies. The small eddies which are near the wall move slowly and quickly damp out. Therefore, small eddies can only be detected for a small separation distance. Large eddies convect at higher velocity and can be detected further downstream. A later study by
$\frac{\sigma_p}{q_s}$ vs $M_e$

- $T_w/T_{aw} \ll 1$
- $T_w/T_{aw} = 0.43$
- $T_w = T_{aw}$

- Raman (1974)

Laganelli (1983)

---

Figure 2.1: Surface pressure fluctuations, Laganelli and Martelluci (1983)
Figure 2.2: Convection velocity
Kistler and Chen [46] shows that the convection velocity is Mach number dependent. As shown in Fig. 2.2b, $U_c/U_\infty$ decreases from a value of 0.85 at subsonic flow to 0.65 at supersonic and hypersonic flows.

2.2 Expansion Corner Flows

The Prandtl-Meyer solution for the flow around an expansion corner is well known. Consider a supersonic flow that is being turned around by a convex wall. The turn results in an expansion fan, with increasing Mach number and decreasing pressure through the fan. For Prandtl-Meyer flow, the expansion process is concentrated at corner point $O$, Fig. 2.3, and the flow is isentropic. The flow properties depend upon the angle of turning $\alpha$ and the incoming Mach number $M_\infty$. However, Adamson [1] indicated that the transport properties in the expansion process of a real flow have strong effects on the flow properties. For an expansion flow, the velocity and streamline patterns change due to the transport properties and the pressure drop at a given point along the $x$ direction decreases as the magnitude of velocity decreases. The flow would reach the final equilibrium condition only after some finite distance downstream downstream of a corner [1]. Fig 2.4 shows Goldfeld’s [33] measurements of surface pressure around a corner. It can be seen that the expansion process is not centered at the corner. The distance marked in figure as $\tilde{x}_D$ can be called a downstream influence of the corner, it being analogous to the upstream influence of shock wave/boundary layer interactions. It is defined as the distance from the corner to the intersection of the tangent through the downstream pressure data with the inviscid downstream pressure. In Goldfeld’s experiments, the surface pressure approached the inviscid pressure about $10\delta_0$ downstream of the corner, where $\delta_0$ is the undisturbed boundary layer thickness. Also, other previous studies show that the downstream
$M_\infty$

--- leading & trailing Mach waves of inviscid expansion

Figure 2.3: Expansion corner flow
Figure 2.4: Surface pressure distributions, Goldfeld (1983)
influence depends on the strength of the expansive flow [8, 16] in which the downstream influence increases as the deflection angle increases. More discussion of the downstream influence will be given in Section 4.2.

Goldfeld argued that a reverse transition of a turbulent boundary layer, or relaminarization, may occur if there exists a region with large favorable pressure gradient. In a highly accelerated flow, the pressure force is considered to dominate over the slowly responding Reynolds stress in the outer region and a new viscous sublayer is generated [54]. The flow is no longer in equilibrium. For Goldfeld’s experiments, the velocity profiles downstream of the corner show a distinguishing feature from the wall-wake law, indicating the presence of a non-equilibrium turbulent boundary layer. Within the expansion fan, the logarithmic region disappears or is subjected to a strong distortion. The thicknesses of the viscous sublayer and buffer region increase, and the wake component $\Delta U^+$ vanishes. A complete retransition to turbulent boundary layer is not obtained until $x > 20\delta_0$.

Further, the study of Narasimha and Sreenivasan [54] indicates that the turbulent fluctuations, or Reynolds stress, is not necessarily zero in a relaminarized flow but rather is negligibly important to mean fluid dynamics. For instance, when a turbulent boundary layer is subjected to a large favorable pressure gradient, the re-transition is rather pronounced due to the dominance of the pressure force over the Reynolds stress. Based on this observation, a relaminarization criterion for boundary layer flows between Mach 1.3 and 2.2, is suggested by Narasimha and Viswanath [55] as:

$$\frac{\Delta p}{\tau_0} > 75$$

(2.3)

where $\Delta p$ is the magnitude of the pressure drop across the Prandtl-Meyer expansion and $\tau_0$ is the wall shear stress at the corner. In other words, relaminarization occurs if
pressure forces are sufficiently large compared with the Reynolds stress. In the study of hypersonic expansion flow by Bloy [8], the heat transfer distributions for $\alpha = 10^\circ, 15^\circ$ are not predicted by either laminar or turbulent theories which indicates the presence of either a relaminarized flow or a turbulent flow out of equilibrium, although it is not classified as a reverse flow by Narasimha and Viswanath [55]. Therefore, the Narashima-Viswanath criterion, based on low Mach number studies, requires to be further examined for strongly expansive flows.

Detailed measurements of the interactions of a turbulent boundary layer past an expansion corner have shown that there is a reduction of wall heat transfer, skin friction and turbulence intensity, even if the flows do not relaminarize. The wall heat transfer distributions of Bloy's study shows that the reduction of turbulence mixing due to the convex surface subsequently decreases the heat transfer downstream of the corner. Chew [16] and Goldfeld [33] further observes that the skin friction distribution exhibits a similar trend as the wall heat transfer. The skin friction increases behind the expansion corner at first, and then smoothly decreases. Further downstream, the increase of skin friction indicates a retransitioning flow and the boundary layer begins to recover to a new equilibrium state. The study of Dussauge and Gaviglio [26] also indicates that velocity fluctuations $\overline{u'^2}$ decreases through the expansion corner. Thus the turbulence is damped and a new internal layer is created following a changing of the properties of the whole boundary layer.

The expansion process can also be characterized by boundary layer lengthscales. Examples from Chew [16] at Mach 2.5 are shown in Fig. 2.5. It can be seen that the boundary layer thickness $\delta$ and the displacement thickness $\delta^*$ increase across the expansion corner. However, Chew indicates that the mass flow rate ($\dot{m} = \rho U(\delta - \delta^*)$) remains almost constant in spite of large increase of $\delta$. Therefore, the increase of $\delta$ is
Figure 2.5: Boundary layer length scales, Chew (1979)
considered not due to the entrainment of additional fluid [16]. Also, Smith and Smits shows that $\langle au \rangle'$ did not decrease [65].

2.3 Two-Dimensional Shock Waves/Turbulent Boundary Layer Interactions

Shock wave/boundary layer interactions have been studied for over fifty years. Some comprehensive reviews of two-dimensional studies were given by Adamson and Messiter [2], Délier and Marvin [22], Green [38], and Stanewsky [68]. The understanding of two-dimensional interactions from past research work is considered to be sufficient to predict and model this complex flow phenomena reasonably well. However, data are still needed for the analytical methods and calibration of computational schemes particularly in the hypersonic regimes [40, 61]. In this section, a brief review of two-dimensional interactions is given to provide a general background to the present study.

The wave patterns involving impinging shock wave/boundary layer interactions are shown in Fig 2.6a–c. For an inviscid flow, the impinging shock is reflected as a single wave with the same deflection angle, Fig. 2.6a. When the shock is weak, the general flow structure of the interaction does not substantially differ from the inviscid flow case, Fig. 2.6b. However, the pressure rise associated with the shock wave is partially transmitted upstream through the inner subsonic portion of turbulent boundary layer. As a result, thickening of the streamtube near the wall causes the outer supersonic portion to be deflected away from the wall. Consequently, the streamline deflection generates compression waves. As long as the shock wave is weak, the interaction is a localized phenomenon which depends on the properties of the incoming boundary layer and the local flowfield. The interaction region is typically limited to two or three times the undisturbed boundary layer thickness [37]. Also,
a. Inviscid flow.

b. Unseparated viscous interaction.

c. Separated viscous interaction.

Figure 2.6: Oblique shock wave/boundary layer interactions, Lu (1988)
the boundary layer thickness decreases downstream of the interaction. Further, at a critical shock strength, the boundary layer separates, forming a region of reverse flow. The disturbance creates by the shock wave propagates further upstream through a larger subsonic region. The interaction affects the outer flow pattern drastically and the surface pressure changes appreciably from the inviscid model, Fig. 2.6c. The wave system is generally characterized by an initial compression, an expansion fan behind the reflected shock and a recompression during boundary layer reattachment. Also, a typical pressure distribution associated with separated flow has three distinct regions, including a steep pressure rise near the separation region due to the thickening of the boundary layer followed by a reduced pressure gradient region which is an indication of separated flow and a sharp pressure increase during the reattachment process. Further, the pressure rise to separation is independent of the total pressure rise or the configuration inducing the interaction if the separation is strong enough; this is known as “free interaction” [12].

The upstream influence \( x_u \), associated with the propagation of disturbance upstream through the subsonic layer, can be defined as the distance between the shock position in an inviscid flow model and the point where the unsteady shock motion is first detected [22]. The upstream influence scale is important for understanding the interaction and it is usually determined experimentally from either surface pressure distribution or surface flow-visualization [51]. From the surface pressure distribution, the upstream influence scale is usually determined as the intercept of the tangent to the maximum pressure gradient with the upstream surface pressure as shown in Fig. 2.7a.

The extent of the interaction is dominated by the following parameters, namely, the upstream Mach number \( M_\infty \), the Reynolds number \( Re_{\kappa} \), the shock strength
Figure 2.7: Upstream influence of two-dimensional interaction
the upstream Mach number \( M_\infty \), the Reynolds number \( Re_\infty \), the shock strength \( p_F/p_1 \), wall temperature condition \( T_w/T_{aw} \) and the incoming boundary layer state [37]. At a fixed Reynolds number, the upstream influence increases with shock strength for a given incoming Mach number \( M_\infty \) and increases even more rapidly once the flow is separated [58], Fig. 2.7b. Further, the upstream influence decreases with Mach number for the same shock strength. The Reynolds number effect on the upstream influence is more uncertain. At a fixed incoming Mach number and shock strength, the upstream influence increases with the Reynolds number at the low or moderate range [28]. At high Reynolds number, a reversal is observed. An empirical formula proposed by Settles et al. [61] for a Mach 3 ramp flow is given as

\[
\hat{x}_u Re_\infty^{1/3} = 0.9 \exp(0.23\beta)
\]  

(2.4)

where \( \beta \) is the ramp angle and \( 10^5 \leq Re_\infty \leq 10^7 \). For fixed ramp angle \( \beta \), the length scale \( \hat{x}_u \) is proportional to \( Re_\infty^{-1/3} \), that is, the upstream influence scale decreases with increasing Reynolds number. The reversal is basically related to the characteristics of the incoming turbulent boundary layer at different Reynolds number. The interaction is dominated by viscous effects at low Reynolds number due to a relatively thick viscous sublayer. At high Reynolds number, the viscous sublayer becomes sufficiently thin and pressure propagation becomes an inviscid phenomenon [22]. Further, wall temperature effect on the upstream influence was investigated by Spaid and Frishett [66]. Cooling the wall \( (T_w/T_{aw} < 1) \) results in a thinning of the subsonic layer and the upstream influence scale is considerably reduced.

Another important feature associated with shock wave/boundary layer interactions is unsteadiness, e.g., as manifested in surface pressure fluctuations in both attached and separated flows. An early study by Kistler [45] at Mach 3 and 4.5 using a forward facing step reveals that the time trace of pressure fluctuations shows an
intermittent behavior near the interaction start. Also, a peak rms pressure, $\sigma_p/\sigma_{p,o} \approx 20$, occurs between the upstream-influence and the separation line. The substantial increase of pressure fluctuations causes significant problems in high-speed flight, namely, structural vibration or fatigue, associated peak heating, and aerodynamic noise.

Following the work of Kistler, tremendous efforts have been undertaken to understand the unsteady nature of the interactions for both two- and three-dimensional configurations [25, 32, 53, 67, 71]. Discussion here is limited to compression ramp interactions. In the study of Dolling and Or [25], with a ramp angle from 12° to 24° at a freestream Mach number of 2.9, the surface pressure fluctuations show intermittent behavior, as reported by Kistler [45], between the upstream influence and separation. Fig. 2.8a shows the intermittency factor $\gamma$ with $\tilde{x}$, where $\gamma$ is defined as [25]

$$\gamma = \frac{\text{time}[p_w > (p_{w,o} + 3\sigma_{p,o})]}{\text{total time}} \quad (2.5)$$

All the distributions have the same shape and the downstream flow (where $\gamma = 1$) is located near the separation line or close to the corner in the attached flow case ($\beta = 12^\circ$). Within the intermittent region, the unsteadiness of the interactions results in a peak rms pressure. Dolling and Or’s results are replotted in Fig. 2.8b. The figure shows that the peak rms pressure is 10–20 percent of the local surface pressure $p_w$ and $\sigma_{p,max}$ increases with the shock strength or ramp angle. Moreover, recent work of Gramann and Dolling [36] at Mach 5 shows that the unsteadiness of the interactions is a global phenomena for a separated flow. The interaction region extends from the upstream influence line to where the surface pressure reaches the downstream inviscid value. The rms pressure fluctuation increases due to a low-frequency, large amplitude component caused by the global unsteadiness. Also, the rms generated by a high-frequency, small-scale turbulence increases throughout the interaction. The
Figure 2.8: Surface pressure fluctuations, Dolling and Or (1985)
rms distribution is similar to the Mach 3 results of Dolling and Or. However, the magnitudes at Mach 5 are substantially larger which show that the intense surface pressure fluctuations are of critical importance in the hypersonic flow regime [35].

The study of Muck et al. [53] with similar test configurations as Dolling and Or further examines the interaction unsteadiness through the autocorrelation function $R_{pp}(\tau)$ and the convection velocity $U_c$ from the surface pressure fluctuations. They observed that the turbulence microscales and integral scales obtained from the autocorrelation inside the separation region increase with decreasing ramp angle. These time scales simply indicate a shift in the shock frequency as the shock strength decreases. Also, the convection velocity $U_c$ is significantly reduced in the separation region and it decreases with ramp angle ($U_c/U_\infty = 0.60, 0.48$ for ramp angles of $24^\circ$ and $20^\circ$ respectively). According to Muck et al., the pressure fluctuations are basically dominated by the high-frequency oscillatory shock wave motion. The energy-containing eddies outside the flow recirculation zone propagating downstream appear to be the main sources of wall pressure fluctuations in the separation region.

Finally, previous investigations concerning the mutual interactions of shock waves and expansion fans are very scarce. The only experimental program known to the author is the study of Chew [16]. In his study, with $4^\circ$ to $8^\circ$ deflection angles at Mach number of 1.8 and 2.5, and a $6^\circ$ expansion corner, the Prandtl-Meyer expansion tends to "neutralize" the reflected shock and reduces the extent of the interaction. The turbulence intensity increases due to the unsteady shock motion but drops downstream of the expansion corner. The mutual interference also depends on the shock impingement position. When the shock impinges upstream of the corner, the mutual influences increase with shock strength while the mutual influences reduce when the shock impinges downstream of the corner.
CHAPTER 3

EXPERIMENTAL PROGRAM

An experimental study of oblique shock wave/boundary layer interactions in the presence of expansion corners was performed in the University of Texas at Arlington (UTA) hypersonic shock tunnel located in the Aerodynamics Research Center (ARC). The test conditions were an incoming flow at Mach 8 and a unit Reynolds number of $10.2 \times 10^6$ per meter. The expansion corner angle $\alpha$ was 2.5° or 4.25° while the oblique shock generator angle $\beta$ was 2° or 4°. The shock impingement position $x_{sh}$ was near the expansion corner (Fig. 3.1). The experimental program included the design and fabrication of test surfaces and external shock generators, wind tunnel testing and data analysis. The tests were conducted with the boundary layer developed naturally over a flat plate. Wall pressure data were gathered for the flat plate and expansion corners with or without shock impingement. Boundary layer profiles were also obtained for the undisturbed and expansive flows. Limited Pitot pressure surveys for the shock impingement cases were performed.

3.1 Test Facility

3.1.1 Shock Tunnel

Shock tunnels have been used for many years to study high-speed flows, and they offer one way of producing both high enthalpy and total pressure levels. The run time is short compared to supersonic blowdown tunnels but is usually long enough
(a) Shock impingement ahead of expansion corner.

(b) Shock impingement at expansion corner.

(c) Shock impingement behind expansion corner.

Figure 3.1: Test configuration
to establish steady flows over most model configurations [11]. Detailed description of the shock tunnel used in this experiments was given in Ref. [69] and only a brief description is given below.

A schematic diagram of the UTA shock tunnel is shown in Fig. 3.2. The tunnel is of conventional design and consists of a shock tube connected to a nozzle downstream of which are the test section, diffuser and dump tank. The driver tube is 3 m (10 ft) long with a 15.24 cm (6 in.) I. D. and is connected to the driven tube via a double-diaphragm section. The driven tube, which consists of three segments, is 8.22 m (27 ft) long with the same diameter as the driver tube. Flow is initiated by rupturing the double-diaphragm which initially separates the driver and driven gases. The double-diaphragm arrangement provides precise control of the driver and driven pressure which in turn ensures repeatable stagnation conditions and unit Reynolds number.

Also, a secondary diaphragm of thin Mylar or aluminum foil is used to separate the driven tube and the nozzle. Once the secondary diaphragm is ruptured, the test gas in the driven tube is expanded by a conical nozzle with a 7.5° half-angle expansion. Interchangeable throat inserts provide variable test section Mach number capability from 5 through 16. The test section is a semi free-jet design and it is 53.6 cm (21 in.) long and 44 cm (12 in.) in diameter. Following the test section is a diffuser and a 4.25 m³ (150 ft³) upright, cylindrical dump tank.

The test model consisted of a flat plate mounted in the test section. The relative position of the flat plate inside the wind tunnel is shown in Fig. 3.3. The plate, with a 15° sharp leading edge, was 203 mm (8 in.) wide by 0.96 m (37.75 in.) long. The long plate length and high Reynolds number of the test flow were considered to be necessary to obtain a natural turbulent boundary layer without a tripping device. The plate was supported by a single foot and was mounted 50 mm (2 in.) below the tunnel.
centerline to avoid wave focusing that exists along the centerline of axisymmetric test sections. Also, side fences under the plate were used to prevent crossflows. Due to the limited size of the test section, parts of the test surface protruded into the nozzle and diffuser. Hence a favorable pressure gradient was imposed initially on the flow. Further details of the undisturbed boundary layer are discussed in Section 3.3.5.

3.1.2 Instrumentation Plates

The top of the flat plate was comprised of four 200 mm by 200 mm (8 in. by 8 in.) interchangeable stainless steel plates which were butted tightly together with O-ring material sealing the endfaces. For surface pressure measurements, one of the plates was replaced by the instrumentation plates as shown in Fig. 3.4a–c. In Fig. 3.4a, the instrumentation plate with four static pressure taps was used to measure the surface pressure distribution of the undisturbed boundary layer. These pressure taps of 2.5 mm (0.098 in.) diameter were drilled perpendicularly to the test surface. Also, these pressure taps were spaced 50.8 mm (2 in.) apart and 3.18 mm (0.125 in.) off-centerline. For surface pressure measurements of the flat plate with shock impingement, an instrumentation plate (Fig. 3.4b) with 14 pressure taps spaced 6.35 mm (0.25 in.) apart was used to improve the spatial resolution. Further, two instrumentation plates (Fig. 3.4c) with 2.5° and 4.25° ± 0.1° deflection angles respectively were fabricated for surface pressure measurements of expansion corner flows. Two rows of orifices offset from both sides of the centerline by 3.18 mm (0.125 in.) were drilled perpendicularly to the test surface from 38.1 mm (1.5 in.) upstream of the corner to 60.3 mm (2.375 in.) downstream. One of the rows was for mounting pressure transducers while the other row was for coaxial surface thermocouples. The orifices were spaced 6.35 mm (0.25 in.) or approximately 0.5 δo apart, δo being the
Figure 3.4: Instrumentation plates
incoming boundary layer thickness.

In designing the expansion corner model, it was hypothesized that hypersonic similarity may possibly exist between corner flows at different Mach numbers for given values of

$$K = M_\infty \alpha$$  \hspace{1cm} (3.1)

where $\alpha$ is the corner angle in radians, this being analogous to the hypersonic similarity parameter of slender bodies [5]. Thus, it was thought that the proposed hypersonic experiments would be more revealing if they were performed with values of $K$ comparable to those found in more numerous supersonic experiments. For the present, two corner angles of 2.5° and 4.25° were chosen which gave $K = 0.35$ and 0.59 respectively. A combined supersonic-hypersonic similarity parameter $\alpha \sqrt{M_\infty^2 - 1}$ proposed by Van Dyke [73] was also examined but this parameter was found to be applicable to a restricted range of $\alpha$ only.

The pressure ratio across the Prandtl-Meyer expansion $p_2/p_1$ against $K$ for the experimental conditions is plotted in Fig. 3.5, where the boldline is for $M_\infty \rightarrow \infty$. In addition to plotting the present conditions, those of some previous investigations are also displayed. It can be seen that the present values of $\alpha$ are within the range of previous supersonic experiments. More details of the appropriateness of $K$ as a scaling parameter will be discussed in section 4.2.1.

### 3.1.3 Shock Generator

The shock generator consisted of a plate and an angle-of-attack adapter, Fig. 3.6. The plate was made of aluminum and was 12.7 mm (0.5 in.) thick, 180 mm (7 in.) wide and 130 mm (5.25 in.) long. Its leading edge was cut at 15°. The plate was designed to span the test surface, and its length ensured that expansion waves from
Figure 3.5: The parameter $k$ for scaling expansion corner flows.
Figure 3.6: External shock generator
the trailing edge impinged the boundary layer downstream of the region of interest. The plate was tightened to an adapter to obtain the desired shock generator angle of 2 or 4 ± 0.1°. The adapters were made of aluminum and they had 15° leading edges. The shock generator assembly was mounted and supported by a sting. The specially designed sting mount allowed the shock generator to traverse vertically and horizontally. Further, before running the tunnel, the position of the shock generator was adjusted so that the shock impinging position based on an inviscid shock calculation was set to ± 1.3 mm (0.05 in.).

3.1.4 Pitot Probe and Boundary Layer Rake

The Pitot probes, Fig. 3.7a, were made of 12 and 15 gauge stainless steel tubes. The tubes were soldered together by 45 percent silver solder to provide a solid joint. Kulite pressure transducers snuggled inside the probe at 18 mm (0.7 in.) from the tips to ensure a fast response. The flattened intake was 1.9 mm (0.075 in.) wide by 0.25 mm (0.01 in.) high to minimize displacement effects of the Pitot probe and to improve the spatial resolution [4]. For the Pitot pressure survey, four Pitot probes were installed inside a boundary-layer rake, Fig. 3.7b. The housing, with a double-face leading edge, consisted of a 130 mm (5.125 in.) long, 57 mm (2.25 in.) high by 22 mm (0.875 in.) wide hollow pocket. The probes protruded through a slot and faced the incoming flow perpendicularly. These probes could be moved in small variable steps, accurate to 0.05 mm (0.002 in.), to build up a fairly detailed boundary-layer profile in about 4–5 runs.
(a) Pitot Probe

(b) Boundary-Layer Rake

Figure 3.7: Pitot probe and boundary layer rake
3.2 Measurement Techniques

3.2.1 Data Acquisition System

A LeCroy high-speed data acquisition system was used for the experiments. Two four-channel, twelve-bit waveform recorders (Model 6810) with a digitizing and sampling rate up to 5 Megasamples/sec each were available. Programmable amplifiers allowed each channel to read from 400 mV to 100 V full scale. In this study, the output range of each channel was adjusted to optimize the resolution of the different output ranges of the pressure transducers and thermocouples. All input channels were triggered simultaneously, either externally or by using an input channel as a trigger source. Fig. 3.8 shows a block diagram of the data acquisition system. All the transducers were connected to a Tektronix Model CPS250 15 V DC power supply. Transducer signals were digitized and stored in the model 6810 high-speed modules while the Everex Step 286 host computer with CATALYST software controlled the setup of the model 6810 modules through a LeCroy Model 8901A Interface and a pair of National Instruments Model GPIB–110 bus extenders. The latter was necessary because the data acquisition system was located over 30 m (100 ft) from the host computer. Also, external instrumentation amplifiers (Leyh Model 29) were used. The frequency response of the amplifier is shown in Fig. 3.9. It can be seen that the amplifier also performed as a low-pass filter. With a gain of 500, the roll-off frequency is about 100 kHz. The sampling rate used in the experiments was 1 Megasamples/sec (the highest sampling rate for four channel operation).

3.2.2 Surface Pressure Measurements

The pressure transducers used in the measurements initially were Kulite Model XCS–093–50A. These pressure transducers have an outside diameter of 2.36 mm
Figure 3.8: Data acquisition system
Figure 3.9: Frequency characteristics of amplifier
(0.093 in.) and a pressure sensitive sensor of 0.97 mm (0.038 in.) in diameter. The natural frequency of the transducers is 250 kHz as quoted by the manufacturer, and the sensitivity is 0.73–1.03 mV/kPa (5–7 mV/psia) with a 15 V DC excitation. It was found that the static calibration differs by only a few percent from a dynamic calibration using a shock tube [14]. The pressure transducers were therefore calibrated statically only.

The 300 kPa (50 psia) pressure transducers were used in order to survive the final equilibrium pressure of 200 kPa (30 psia) in the tunnel. When using these pressure transducers to measure low pressures of 0.3–2.8 kPa (0.04–0.40 psia), Leyh Model 29 amplifier-filters were used to improve the signal-to-noise ratio. Later, a relief valve was installed to reduce the final equilibrium pressure of the tunnel. A pressure trace in the test section with the relief valve installed is shown in Fig. 3.10. The peak pressure was only about 3.5 kPa (0.5 psi) above atmospheric pressure. Therefore Kulite XCS-093-5A pressure transducers with a 0–172 kPa (0–25 psia) range were employed to obtain better signal-to-noise ratios in latter surface pressure measurements. The natural frequency of these transducers is quoted by the manufacturer as 100 kHz, and the sensitivity is 10–12 mV/kPa (70–80 mV/psia) with 15 V DC excitation.

For surface pressure measurements, the pressure transducers were flush mounted and potted using silicone rubber sealant. Flush mounting is required to obtain dynamic data. In this study, the flushness of the transducers was checked by a machinist's block to be better than 0.005 δo in order to minimize interference with the flow [17]. Further, resolution of pressure fluctuations is limited by the finite size of the pressure transducer, i.e., there is high frequency damping due to the transducer size. Therefore, even though the sampling rate was set as 1 Megasample/sec in the present studies, the high frequencies were not well represented. According to Corcos'
Figure 3.10: Surface pressure trace with relief valve installed
criterion [19], the maximum measurable frequency is given by

\[ f_{\text{max}} = \frac{U_c}{2\pi r} \] (3.2)

where \( r \) is the radius of the pressure transducer and \( U_c \) is the convection velocity. In the present measurements, \( U_c \) (see Sec. 4.1.2) was about 850 m/sec (2800 ft/sec) and \( r = 0.48 \text{ mm} \) (0.019 in.). Thus \( f_{\text{max}} = 280 \text{ kHz} \). In addition, a nondimensional transducer diameter \( d^+ = \frac{U_c d}{\nu_w} \) was used for examining the effects of transducer size on the spatial resolution, \( d \) being the transducer diameter. In the present experiments, \( d^+ \approx 200 \) which gave an rms value of only about 60 percent of an ideal transducer \( (d^+ = 20) \) [60]. Further, the raw data through the amplifier was processed using a digital filter which cut off the upper frequency of the signal to 100 kHz. The 100 kHz data bandwidth severely limited the upper nondimensional frequency \( f_\delta/U_\infty \) to about 1.0, which was on the lower end compared to other results summarized by Dolling and Dussauge [24]. The reduced frequency \( f\nu_w/U_\infty^2 \) which was about 0.008 for the present experiments was also comparatively low. Based on the above discussion, it appears that the resolution of the present data is limited more by the transducer’s resonant frequency and narrow bandwidth characteristics than by the transducer size.

As mentioned above, even though the transducers were used for dynamic measurements, a static calibration sufficed in determining the calibration coefficients [14, 57]. But the transducers’ drift and hysteresis, which were quoted by the manufacturer as a percentage of the fullscale range, could be significant, especially when the transducers were used to measure low pressure [32]. To reduce the effects of drift, the transducers were calibrated against an MKS Baratron Model 127A vacuum gauge while the test section was being evacuated. This vacuum gauge is a capacitance-type manometer accurate to \( \pm 7 \text{ Pa} \) (\( \pm 0.001 \text{ psia} \)) and is used as a secondary standard. Least-square linear fits were made to the calibration data from which the transduc-
ers' sensitivities were obtained. Thirty minutes or more elapsed between a run and calibration during which significant zero shift was encountered. The drift problem was overcome by re-adjusting the transducers' calibrations through comparing the transducers' output against the vacuum gauge prior to tunnel firing. Subsequently, the acquired data were converted into engineering units using the linear fits, with the sensitivities obtained from calibration and off-sets obtained from the final zeroing adjustment.

In addition, the overall electronic noise of the transducer and the data-acquisition chain expressed as a signal-to-noise ratio S/N in dB was estimated as follows:

\[ S/N = 20 \log_{10} \left( \frac{\langle p_s \rangle}{\langle p_n \rangle} \right) \]  \hspace{1cm} (3.3)

where \( \langle p_s \rangle \) is the mean signal signal and \( \langle p_n \rangle \) is the rms of the noise. Fig. 3.11 shows the S/N ratio with 100 kHz cutoff frequency decreases with decreasing pressure. The S/N ratio was about 20 dB (10:1) for \( p < 0.05 \) psia. However, the magnitude of the pressure signal increased substantially with shock impingement. Therefore, the S/N ratio improved significantly downstream of shock location.

3.2.3 Data Reduction

The data reduction procedure for the mean surface pressure measurements followed standard practice. Further data analysis and processing of the fluctuating pressure data were also performed to extract time-domain statistics. Convection velocities were calculated for some test cases. All the data processing discussed in the following sections was carried out using the ASYSTANT software package.

The basic statistical properties of importance for describing a single stationary random record [7] consist of the autocorrelation \( R_{pp}(\tau) \), the probability density function PDF and the root-mean-square (rms). The rms \( \langle p'_{w} \rangle \) is the average value
Figure 3.11: Signal-to-noise characteristics of transducers
indicating the standard deviation of a signal. If the mean is substracted, the rms is equal to the standard deviation $\sigma_p$. The rms can be computed as:

$$\text{rms} = < p'_w > = \left( \frac{1}{N} \sum_{i=1}^{N} ( p_w(t_i) - \bar{p}_w )^2 \right)^{1/2}$$

(3.4)

where $\bar{p}_w = \sum_{i=1}^{N} p_w(t_i)/N$ and $N$ is the total number of data points.

The PDF represents the rate of change of probability with data value and it is also another indicator of the amplitude property of the fluctuating signals. To compute the PDF, it is usual to substract the mean value from the raw data. Then the probabilities of all the data within each interval are evaluated. The probabilities against the amplitude of the fluctuating signal is the PDF. It is useful to note that the PDF is Gaussian when the signal is random. The normalized PDF is expressed as:

$$\text{Normalized PDF} = \frac{N_i}{NW}$$

(3.5)

where $W$ is the interval width and $N_i$ is the number of data points that fall within each interval.

The autocorrelation function $R_{pp}(\tau)$ is useful for determining the microscale of the pressure fluctuations. For digitized pressure signals with time delay $\tau = r\Delta t$

$$R_{pp}(\tau) = \frac{1}{(N-r)} < p'_w > \sum_{i=1}^{N-r} p'_w(t_i) p'_w(t_i + r\Delta t)$$

(3.6)

where $r$ is the lag number.

The correlation function of interest between two recorded surface pressure fluctuations in the present study is the two-point space-time correlation. The space-time correlation or cross-correlation function can be defined as:

$$R_{pp}(\xi, \tau) = \frac{1}{N} \sum_{i=1}^{N} p'_{w1}(t_i) p'_{w2}(t_i + \tau)$$

$$< p'_{w1} > < p'_{w2} >$$

(3.7)
where $\xi$ is the transducer spacing.

For a nearly frozen turbulent flow, Taylor's hypothesis suggests that the slowly distorting eddies are convected downstream by mean flow at a steady, convection speed $U_c$ [78]. There are different definitions of convection velocity in the literature [34]. In the present study, the convection velocity based on Taylor's hypothesis is used. If the time delay between upstream and downstream signal peaks, i.e. $R_{pp,\text{max}}(\xi_1, \tau_1)$ and $R_{pp,\text{max}}(\xi_2, \tau_2)$, of the transducers is $\tau^* (= \tau_2 - \tau_1)$, the convection velocity $U_c$ can be estimated as:

$$U_c = \frac{|\xi_2 - \xi_1|}{\tau^*} \quad (3.8)$$

The accuracy of $U_c$ depends on the resolution of the correlation peaks. In the present data, a microsecond error caused an error of ten percent to the estimate of $U_c$ and similar error bands could be inferred from previous investigations [24].

3.2.4 Pitot Survey and Data Analysis

The Kulite Model XCS–093–50A pressure transducers were originally installed inside Pitot probes for surveying of the undisturbed boundary layer. It was observed that several pressure transducers with type "M" screens were damaged during the runs. Therefore, new pressure transducers, Kulite Model XCS–093–15A with type "B" screens, were used. These type "B" screens afforded more protection from impacting particles than type "M" screens.

Pitot surveys of the undisturbed boundary layer were made from 0.74 m (29.13 in.) to 0.816 m (32.13 in.) from the flat-plate leading edge. The Pitot probes were at 3.2 mm (0.125 in.) off-centerline. During the survey, $p_{\text{pit}}$ and $y$ were measured, while the stagnation pressure $p_o$ and stagnation temperature $T_o$ were obtained from shock
speed measurements. The surface temperature \( T_w \) was assumed to be room temperature. Also, the Pitot surveys were performed for both expansion corner flows, without shock impingement, from 9.5 mm (0.38 in.) to 35 mm (1.38 in.) downstream of the corner. Further, surveys were performed on a number of cases with shock impingement to provide some understanding of the structure of the disturbed boundary layer. From the data, boundary layer profiles were obtained as discussed next.

The usual assumption of constant static pressure normal to the surface is generally acceptable for the boundary layer developing over a flat plate even at Mach 8 [3]. The local Mach number profile in the boundary layer can be calculated from the Pitot pressure profile and the known local static pressure. For supersonic flows, the well-known Rayleigh Pitot-pressure equation is given as

\[
\frac{p_{\text{pit}}}{p_{\infty}} = \left[ \frac{(\gamma + 1)M^2}{2} \right]^{\frac{\gamma - 1}{2}} \left[ \frac{\gamma + 1}{2\gamma M^2 - (\gamma - 1)} \right]^{\frac{1}{\gamma - 1}}
\]  

(3.9)

where \( \gamma \) (= 1.4) is the specific heat ratio. The local Mach number is obtained iteratively from the above equation. In the measurements, no subsonic flow was sensed by the probe even at stations closest to the wall.

For simplicity and lack of a better procedure in obtaining the velocity profile, the following steps were performed. The total temperature \( T_e \) was assumed to be constant through the boundary layer [30]. For a zero-pressure gradient flow \((dp/dx = 0)\) and an isothermal wall condition \((T_w = \text{constant})\), the modified Crocco temperature-velocity relation was applied to obtain the local temperature profiles [6].

\[
\frac{T}{T_e} = 1 + r \frac{\gamma - 1}{2} M_e^2 \left[ 1 - \left( \frac{U}{U_e} \right)^2 \right] + \frac{T_w - T_r}{T_e} \left( 1 - \frac{U}{U_e} \right)
\]

(3.10)

where

\[
\frac{T_r}{T_e} = 1 + r \frac{\gamma - 1}{2} M_e^2
\]

(3.11)
and the recovery factor $r = 0.89$.

The local velocity profile was reduced directly from the Mach number and local temperature ratios as

$$\frac{U}{U_e} = \frac{M}{M_e} \sqrt{\frac{T}{T_e}}$$

(3.12)

The boundary layer thickness was obtained directly from the Pitot pressure profiles. For a compressible boundary layer, the boundary layer thickness can be defined as:

$$y = \delta, \quad \text{at} \quad p_{pit} = 0.99 \ p_{pit,\infty}$$

(3.13)

the criterion recommended at high Mach numbers [30]. With the perfect gas assumption, the density ratio was obtained from

$$\frac{\rho}{\rho_e} = \frac{T_e}{T}$$

(3.14)

Thus the displacement thickness $\delta^*$, momentum thickness $\theta$ and shape factor $H$ were calculated from the local density and velocity profiles.

Beside the experimental boundary layer profiles, the profiles were further analyzed using the Sun-Child curvefit [70], which is based on the Van Driest generalized velocity and Coles' Law-of-the-Wake [18]. For an incompressible, turbulent boundary layer, the velocity profile outside the viscous sublayer and the buffer layer can be written as

$$\frac{U}{U_r} = \frac{1}{\kappa} \ln \left( \frac{yU_r}{\nu} \right) + C + \frac{\Pi}{\kappa} W\left( \frac{y}{\delta} \right)$$

(3.15)

where $\kappa$ is the von Karman constant (usually 0.4), $C$ is the law-of-wall constant (usually 5.1) and $U_r = \sqrt{T_w/\rho_w}$. The function $W(y/\delta)$ is Coles' universal wake function, and the wake parameter $\Pi$ provides an indication of the equilibrium behavior of boundary layer.
For the mean characteristics of compressible turbulent boundary layers, the Van Driest transformation has been shown to give closest agreement with Eqn. (3.15) for a wide range of Mach numbers for both adiabatic and cold walls [30]. The Van Driest transformation provides a generalized velocity $U^*$ defined as

$$
\frac{U^*}{U_e} = \frac{1}{A} \sin^{-1} \left( \frac{2A^2(U/U_e) - B}{\sqrt{4A^2 + B^2}} \right)
$$

(3.16)

where

$$
A = \sqrt{\left( \frac{T_e}{T_w} \right) \left( \frac{\gamma - 1}{2} \right) M_e^2}
$$

(3.17)

$$
B = \frac{T_e}{T_w} + A^2 - 1
$$

(3.18)

A data reduction program was written using the SIGMAPLOT software package to obtain the transformed experimental and the wallwake profiles in wall coordinate.

### 3.3 Test Conditions

#### 3.3.1 Stagnation Conditions

Shock tube performance based on inviscid, perfect gas conditions is discussed in Ref. [14]. However, experiments [41, 50] showed that the shock was attenuated due to boundary layer growth and finite duration for the diaphragm to burst. Thus, in practice, stagnation conditions must be determined by measurements.

In the present program, the driver tube was charged to 24 MPa (3,500 psia) ± 1.5 percent and the double-diaphragm section to about 12 MPa (1,750 psia). The driven tube was charged to 280 kPa (40 psia) ± 1.3 percent after first being evacuated to remove moist ambient air. The test section, diffuser, and the dump tank were evacuated to less than 0.32 kPa (0.05 psia). The gas used throughout the tunnel was dried air.
Breaking the two diaphragms by venting the double-diaphragm section started the tunnel whereby a shock propagated into the driven tube and an unsteady expansion propagated into the driver tube. The actual shock speed had to be known to estimate the test conditions. The shock strength was determined by the elapsed time of the shock wave as it passed two sensors. A PCB piezoelectric pressure transducer and an E-type coaxial surface thermocouple separated by 0.55 m (21.75 in.) were used to detect the shock-wave passageas shown in Fig. 3.12. The shock Mach number was computed as

$$M_s = \frac{(L/\tau)}{a_1}$$  \hspace{1cm} (3.19)

where $a_1$ is the speed of sound of the driven gas, $\tau$ is the lapsed time of shock wave, and $L$ is the separation distance of transducers.

The stagnation conditions, $p_o$ and $T_o$, were obtained by the perfect gas relations [29]:

$$\frac{p_o}{p_1} = \left(\frac{2}{\gamma + 1}\right) \frac{\gamma M_s^2 - \left(\frac{\gamma - 1}{2}\right) \left[3\gamma - 1\right] M_s^2 - (\gamma - 1)}{1 + \frac{\gamma - 1}{2} M_s^2}$$  \hspace{1cm} (3.20)

and

$$\frac{T_o}{T_1} = \left(\frac{2}{\gamma + 1}\right)^2 \frac{1}{M_s^2} \left[(\gamma - 1)M_s^2 + \frac{3 - \gamma}{2}\right] \left[3\gamma - 1\right] M_s^2 - (\gamma - 1)$$  \hspace{1cm} (3.21)

where $p_1$ and $T_1$ are the initial driven-tube pressure and temperature respectively.

The Reynolds number was calculated based on stagnation conditions as

$$Re/m = \frac{p_o M_{\infty}}{\mu_o} \sqrt{\frac{3.5}{C_v T_o}} \left(\frac{T_o}{T}\right)^{1.5} \left(\frac{T/T_o + 110/T_o}{1 + 110/T_o}\right)$$  \hspace{1cm} (3.22)

where $C_v$ is the specific heat at constant volume and the viscosity was computed by Sutherland’s law which is given by

$$\frac{\mu}{\mu_{ref}} = \left[\frac{T}{T_{ref}}\right]^{\frac{3}{2}} \left[\frac{T_{ref} + s}{T + s}\right]$$  \hspace{1cm} (3.23)
Figure 3.12: Shock speed estimate
where \( s = 110 \text{ K}, \ T_{\text{ref}} = 273 \text{ K}, \) and \( \mu_{\text{ref}} = 1.71 \times 10^5 \text{ N s/m}^2. \)

The initial driver- and driven-tube conditions produced a shock Mach number of 2.15 with a run-to-run variation of less than \( \pm \) 5 percent and the low shock Mach number ensured that real gas effects were negligibly small. The test conditions for the present study were a nominal freestream Mach number of 8, stagnation pressure and temperature of \( p_0 = 5.38 \text{ MPa (780 psia)} \) and \( T_0 = 800 \text{ K (1440 °R)} \) respectively and a Reynolds number of \( Re = 10.2 \times 10^6 \text{ m}^{-1} \) (3.1 million/ft). The static pressure and temperature under these conditions were 0.55 kPa (0.08 psia) and 58 K (105 °R) respectively. The incoming freestream flow was 1.24 km s\(^{-1} \) (4,080 ft/s). The flat plate was at room temperature \( (T_w \approx 290 \text{ K, 522 °R}) \) and thus the experiments were performed under cold-wall conditions \( (T_w/T_o \approx 0.35). \)

The low accuracy in estimating the shock Mach number was thought to be misleading because the accuracy was limited by the accuracy of resolving the shock fronts and the burst-pattern of the diaphragms. It was not due to poor control of the initial tunnel conditions. The five percent repeatability in the estimated shock Mach number between runs resulted in an estimated maximum scatter of thirteen percent, seven percent and twenty percent on the stagnation pressure and temperature, and unit Reynolds number respectively for the ensemble of runs. Therefore, a pressure transducer was installed on the upstream of interaction zone to provide a reference pressure. Through normalizing the surface pressure, variations in the surface pressure measurements due to the poor estimate of \( M_s \) were minimized.

### 3.3.2 Condensation

The onset of condensation in the test section of hypersonic wind tunnel was of some concern. If condensation of water vapor or test gases occurred, the flow would
no longer be isentropic and significant but unknown effects would be felt in the test section [20, 74]. In this study, air was processed through a drying system and thus the condensation of water vapor was negligible.

Concerning condensation of dry test gas itself, the problem was avoided by testing at a sufficiently high stagnation temperature. The minimum stagnation temperature required to prevent static saturation of the test gas is given as [62]:

\[
\frac{370}{T_o} \left[ 1 + \frac{M^2_{\infty}}{5} \right] = 4.7 + 3.5 \log_{10} \left[ 1 + \frac{M^2_{\infty}}{5} \right] - \log_{10} p_o
\]  

where \( T_o \) is in K and \( p_o \) is in bar. For \( M = 8.0 \) operation with \( p_o = 54 \) bar, the minimum stagnation temperature required was 730 K (1300 °R). Also, Daum and Gyarmathy [20] observed that air condensed through the same physical mechanism as that of pure nitrogen at low pressure conditions. For nitrogen and air, the saturation temperature may be approximated by [62]

\[
T_{sat} = 47.7 + 6.8 \log_{10} p_{\infty}
\]

where \( T_{sat} \) is in K and freestream static pressure is in the range of 0.5 mm Hg \( \leq p_{\infty} \leq \) 10 mm Hg. In the present experiments, the freestream static pressure was about 4.2 mm Hg, which yielded a saturation temperature of 52 K. Thus, with a stagnation temperature and freestream static temperature were 800 K and 58 K respectively, a condensation-free flow was achieved in the test section.

3.3.3 Test Time

The test time available for a reflected shock tunnel is limited by the time required to start the nozzle flow and the establishment of steady flow over the test configuration. The test period is terminated by either the arrival of high-pressure driver gas or the leading edge of the expansion waves reflected from the end of the
driver tube [27]. Smith [64] further indicated that the starting process of a hypersonic nozzle flow is dominated by unsteady expansion waves while the shock wave system is of minor importance. This can be illustrated by a sample surface pressure trace obtained with a fast-response Kulite transducer mounted on a flat plate in the test section as shown in Fig. 3.13a. The figure shows that a nozzle starting shock and a system of unsteady expansion waves propagated into test section after bursting of the Mylar diaphragm. The duration of the nozzle starting process was mostly determined by the passage of the unsteady expansion waves. An empirical relation suggested by Davies and Bernstein [21] for the time required to approach a steady state is given as

\[ t = \frac{X_{le}}{\omega U_\infty} \]  \hspace{1cm} (3.26)

where \( t \) is the time measured from the instant of starting shock arrival at plate leading edge, \( X_{le} \) is the streamwise distance from the leading edge and \( \omega = 0.3, 0.5 \) for laminar and turbulent boundary layers respectively. Some estimates of the starting time based on shock-speed measurements in this study is shown in Fig. 3.13b. It can be seen that the starting time was slightly higher than that predicted by Davies and Bernstein. This discrepancy may be due to the uncertainty of shock-speed measurements.

Also, the initial pressure of the test section is important for the determination of tunnel run time. A conclusion of Smith [64] is that sufficiently low initial pressure is necessary to assure a minimum starting time. In this study, the initial pressure was 130–350 Pa (0.02–0.05 psia) and the test time was estimated to be of the order of 400 \( \mu \text{sec} \), Fig. 3.13a.

3.3.4 Freestream Uniformity

Freestream uniformity is crucial since the interaction data would be useless if the experiments are not conducted in well-controlled conditions. It is known that
Figure 3.13: Starting process
a conical nozzle produces a relatively thick wall boundary layer in which the useful spanwise extent is limited by tunnel sidewall boundary layer interference. Pitot pressure measurements were conducted to measure the inviscid core in the wind-tunnel test section.

Fig. 3.14 shows the spanwise \( x_{le} = 33 \text{ cm (13 in.)} \) and \( y = 10 \text{ cm (4 in.)} \) and vertical \( x_{le} = 33 \text{ cm (13 in.)} \) and \( z = 0 \) distributions of Pitot pressure obtained with the flat plate in place. Although it would be desirable to obtain data closer the tunnel wall, the experimental setup proved to be difficult. In Fig. 3.14, it can be seen that \( p_{pit} \) showed a slightly decrease at \( z = -8.3 \text{ cm (3.63 in.)} \) or \( y = 12 \text{ cm (4.75 in.)} \), which indicated the pressure of the wall boundary layer. Also, the Pitot pressure surveyed did not vary by more than two percent within the inviscid core. With an 8.3 cm half-span inviscid core, sidewall interference for the present experiments was expected to be insignificant.

3.3.5 Undisturbed Boundary Layer

An estimate of the end of transition was given by Deem and Murphy as [23]:

\[
x_{tr} \text{ (inches)} = \frac{6.47 \times 10^3 + 2.33 \times 10^3 (M_\infty - 3)^{1.5}}{(Re/\text{inch})^{0.6}}
\]  

(3.27)

For a Mach 8 flow at a unit Reynolds number of \( 10.2 \times 10^6/\text{m} \), the boundary layer on the present flat plate became turbulent at \( x \approx 0.46 \text{ m (18 in.)} \). A turbulent boundary layer was obtained by Johnson [43] under conditions similar to those of the present experiment. Therefore, in the measurement region \((> 0.7 \text{ m})\), the boundary layer was expected to be turbulent.

Four Pitot pressure profiles measured from \( x = 0.74-0.82 \text{ m (29.1-32.1 in.)} \) are shown in Fig. 3.15. Although it would be extremely desirable to obtain data closer
Figure 3.14: Pitot pressure distributions in test section
Figure 3.15: Pitot pressure profiles, flat plate
to the wall, the probe design proved to be extremely difficult. The profiles collapsed very well because of the close proximity of the measuring stations. Data scatter was primarily associated with locating the probes. The Pitot pressure profiles showed a rapid change at the boundary layer edge $\delta$ which provided a useful criterion its location. The normalized Mach number and velocity profiles computed from the data reduction outlined in Section 3.2.4 above are shown in Figs. 3.16 and 3.17. These profiles were regarded as self-similar within experimental accuracy. The velocity profiles also appeared to be full which indicated a turbulent flow at the measurement locations.

From the profile data, boundary layer lengthscales could be obtained and some of these are plotted in Fig. 3.18. These lengthscales were compared with those obtained with EDDYBL, a popular boundary-layer code developed by Wilcox [76]. It can be seen that the code underpredicted $\delta$ slightly but overpredicted $\delta^*$ and $\theta$ substantially. A reason for the latter discrepancy may be because of the lack of data at the bottom of the layer, causing inaccuracies in integrating the experimental data, or may be due to the inadequacies of the code itself.

To further understand whether an equilibrium, turbulent boundary layer existed, the Van Driest II transformation was applied to the velocity profiles and the data were fitted to a combined wall-wake law, as discussed in Section 3.2.4. The transformed data are plotted in wall coordinates in Fig. 3.19 together with the wall-wake law, where $y^+ = yU_+ / \nu$. The present data showed an extremely small wake component which was most likely due to the low Reynolds number of the flow [30], these being in the range of $Re_\theta = 1800-2300$. A small wake component is also characteristic of some other hypersonic profiles [6, 28, 48]. Bradshaw [9] gave

$$\Pi = 0.58[1 - 1.42 \exp(-4.35Re_\theta / Re_{\theta_{\text{min}}})], \quad Re_\theta < 5000 \quad (3.28)$$
Figure 3.16: Mach number profiles, flat plate
Figure 3.17: Velocity profiles, flat plate
Figure 3.18: Boundary layer length scales, flat plate
Figure 3.19: Wall-wake profiles, flat plate
where \( Re_{\theta,\text{min}} = 5000(1 + 0.1M_\delta^2)[1 - 0.3(1 - T_w/T_{aw})] \) for \( M_\delta < 10 \) and \( T_w/T_{aw} > 0.1 \). For the present study, with \( M = 8 \) and \( T_w/T_{aw} = 0.35 \), the wake component \( \Pi \) was found to be 0.08 or \( \Delta U^*/U_\tau = 2\Pi/K \approx 0.39 \). Thus, although the boundary layer satisfied transition criteria at the measurement stations, it is perhaps more appropriate to label the boundary layer a post-transitional one. However, conclusions on the state of the turbulent boundary layer tend to be based on adiabatic flows at low Mach numbers, and knowledge of non-adiabatic hypersonic flows is presently inadequate to allow a better appreciation of their nature.
CHAPTER 4

RESULTS AND DISCUSSIONS

The mean and fluctuating surface pressure and boundary layer surveys of interacting flows are discussed in this chapter. The upstream and downstream influence in the presence of expansion corner and shock impingment are examined by comparison with inviscid solutions. Emphasis is also given to the mutual interference of shock and expansion flows. The unsteadiness of shock wave/boundary layer interactions through the analysis of wall pressure fluctuations is then discussed. The results of both surface pressure and Pitot surveys are first examined separately with the emphasis of different experimental configurations and then compared with each other. The conclusions from this chapter are summarized in section 5.1.

4.1 Flat Plate Flow

The flat plate surface pressure distribution at 0.12–0.91 m (4.9–35.8 in.) from the leading edge was obtained and compared with the inviscid solution and laminar weak interaction theory. In the turbulent portion of the flow, the surface pressures were analyzed for their statistical properties.

4.1.1 Viscous Interactions

The surface pressure distribution along the centerline of the flat plate is shown in Fig. 4.1. The surface pressure is plotted against distance from the leading edge
Figure 4.1: Surface pressure distribution, flat plate
of the flat plate and the surface pressure is normalized by the stagnation pressure obtained from shock-speed calculations. Fig. 4.1 also shows the inviscid pressure and the pressure distribution using laminar weak interaction theory [5], both assuming Mach 8 flow. It can be seen that there a slight, favorable longitudinal pressure gradient existed along the flat plate. Also, the measured and predicted pressures are replotted against the hypersonic viscous similarity parameter $\bar{X}$ as shown in Fig. 4.2. The weak interaction theory is not expected to be valid further to the rear of the flat plate but is included to provide an indication of the contribution of the leading-edge viscous-inviscid interaction to the above-mentioned favorable pressure gradient. It can be seen that conical flow did induce a more favorable pressure gradient than that predicted from weak interaction theory near the leading edge. The subsequent measured surface pressure distribution showed only a small favorable pressure gradient, close to the trend of the laminar theory, and became practically negligible toward the rear of the flat plate. The surface pressure for $x > 0.7$ m (27.5 in.), where the measurements were concentrated, can therefore be assumed constant, with a scatter of three percent.

4.1.2 Characteristics of Pressure Signal

The surface pressure fluctuations were found to be normally distributed. Examples of the normalized PDF are plotted in Fig. 4.3. Also, the standard deviation of the surface pressure was found to be comparable to the semi-empirical predictions of Laganelli and Martellucci [49]. The present measurements gave $\sigma_p/p_w \approx 0.084$, with a two percent scatter, which was slightly lower than Laganelli and Martellucci's predicted value of $\sigma_p/p_w = 0.088$. The lower value of the measurements may be due to the limited frequency bandwidth of the data but was nonetheless within the range of validity of the predictions. Further, the surface pressure fluctuations in terms of the
Figure 4.2: Weak interaction, flat plate
freestream dynamic pressure $q_p/q_{\infty}$ was about 0.002. This 0.2 percent rms value of the surface pressure fluctuations at Mach 8 implied that surface pressure fluctuations were significant, as is also evident in Raman’s data from Mach 5 through 10 [57]. In fact, for cold walls, Laganelli and Martellucci’s prediction results in $\sigma_p/p_w \propto M_e^2$, indicating that the surface pressure fluctuations increase quadratically with freestream Mach number. Thus, the large surface pressure fluctuations would require a deeper consideration of the assumption of negligible pressure fluctuations at high Mach number as commonly made in turbulence modeling.

In addition, space-time correlations of the surface pressure fluctuations were also examined. The transducers were spaced 0.5 $\delta_o$ apart with the first transducer located at $x = 0.737$ m (29 in.). The cross-correlations $R_{pp}(\xi, \tau)$ are plotted in Fig. 4.4. These cross correlations showed a decrease in the peak value with increase in transducer spacings as expected in turbulent flows. The peak values $R_{pp, \text{max}}$ are replotted against transducer spacings $\xi/\delta_o^*$ and $\xi/\delta_o$ in Fig. 4.5a and 4.5b. Although Dolling and Dussauge [24] thought that it is appropriate to use $\delta_o^*$ to scale the transducer spacing, Tan et al. [71] felt that this may be inappropriate because $\delta_o^*$ may not be related to the large-eddy size. The lines in the figure indicate the trends of some previous subsonic and supersonic data. The present data appeared to follow the trend of Speaker and Ailman’s [67] data and were typical of high-speed results. Dolling and Dussauge [24] stated that the supersonic results tend to be high because of the limited frequency range and the present results showed the same feature.

Further, the convection velocity $U_c$ against transducer spacing $\xi/\delta_o^*$ is plotted in Fig. 4.6. The convection velocity increased with transducer spacing, this being observed by previous investigators. The increase in convection velocity is commonly interpreted as follows. Small scale (high frequency) pressure fluctuations are thought
Figure 4.4: Space-time correlation, flat plate
Figure 4.5: Maximum of space-time correlation, flat plate
Figure 4.6: Convection velocity, flat plate
to be convected slowly at velocities typical of the lower portion of the boundary layer. These small fluctuations which have a short time constant decay rapidly. But large-scale pressure fluctuations that are associated with large eddies within the boundary layer are convected downstream at higher velocities and decay slowly, thereby accounting for the increase in convection velocity with increasing transducer spacing.

4.2 Mild Expansion Flow

The results of mild expansion corner flows are given in this section. The hypersonic similarity parameter \( K \) was used as a scaling parameter for turbulent flow past expansion corners. Statistical analyses along with Pitot surveys were performed to understand the nature of the flow under the influence of the expansion.

4.2.1 Influence of Expansion Corner

The mean surface pressure distributions for the flow around the expansion corners are plotted in Fig. 4.7. The pressures are plotted in nondimensional form, using an upstream surface pressure as reference, against \( \bar{x} = x/\delta_0 \), where \( x \) is the streamwise surface coordinate centered at the corner. Also shown in the figure are the inviscid pressure distributions for an incoming Mach 8 flow. The upstream influence was negligibly small for both test cases, which was also observed by previous studies [16, 33]. Further, it can be seen that the measured pressures approached the downstream inviscid values at some distance \( \bar{x}_D = x_D/\delta_0 \) which was estimated to an accuracy of ±0.2, where the distance \( x_D \) is the downstream influence as discussed in section 2.2. The present data showed that the larger the corner angle, the larger the downstream influence. This detailed observation is contrary to the approximate estimate of Narasimha and Viswanath [55] who suggested a constant downstream in-
Figure 4.7: Surface pressure distributions, expansion corners
fluence of about one boundary layer thickness, a result that may be reasonably valid for weaker supersonic interactions.

It was thought that the downstream influence depended primarily on an inviscid parameter $K$ (Sec. 3.1.2) which is considered to be an indication of the strength of the disturbance generated by the expansion corners. The data are replotted together with data extracted from previous investigations in Fig. 4.8. The collapse of data from Mach 1.76 through Mach 8 supports the validity of $K$ as a scaling parameter, at least for $x_D$. Dussauge and Gaviglio’s [26] downstream influence is approximately equal to the weaker case of the present study for approximately the same value of $K$.

It can also be seen from Fig. 4.8 that the surface pressure of weak expansions reaches the downstream inviscid value quickly. These weak conditions can be achieved as low supersonic Mach numbers even for considerably large corner angles as in the case of Dussauge and Gaviglio [26]. On the other hand, in the hypersonic regime, the downstream influence based on the crude estimates from Bloy’s [8] figure appears to tend to a limiting value at large $K'_{\ast}$ although there is not sufficient evidence to support this conclusion.

### 4.2.2 Damping of Surface Pressure Fluctuations

Other than mean pressure data, pressure fluctuations were also analyzed. Surface pressure fluctuations downstream of the corner are normally distributed. Examples of PDFs are plotted in Fig. 4.9. The PDF of the 4.25° expansion showed somewhat more distortion than that of the 2.5° expansion and this may be due to an insufficiently long data record.

The standard deviation of the surface pressure $\sigma_p$ was normalized by the local mean surface pressure $p_w$, undisturbed freestream dynamic $q_{\infty}$, and the incoming
Figure 4.8: Downstream influence, expansion corners
Figure 4.9: PDF, expansion corners
standard deviation $\sigma_{p,\infty}$. The results are illustrated in Fig. 4.10a–c. The rms error to $\sigma_p$ was estimated to be about ± 6 percent. The measured upstream surface pressure fluctuation was slightly below that predicted by Laganelli and Martellucci [49], as was also observed for the flat plate flow discussed in section 4.1.2. In addition, the measured pressure fluctuations decreased downstream of the expansion corner, with $\sigma_p/p_w$ decreasing from 0.08 upstream to about 0.046 and 0.033 for 2.5° and 4.25° deflection angle respectively at $\tilde{x} = 4$. Also, $\sigma_p/q_{\infty}$ decreased from about 0.002 on the flat plate to about 0.001 and 0.0007 for the two respective corners at $\tilde{x} = 4$. Moreover, $\sigma_p$ did not show any asymptotic behavior which indicated that the flow may not be fully relaminarized at the most downstream measurement location. Also, $\sigma_p$ did not increase within the limited range of measurements, that is, the flows were not retransitioning. Thus, the measured data indicated that the expansion flow was in nonequilibrium even at the most downstream measurement station.

The above observation is clearer when viewed in terms of the predicted pressure fluctuations level for a fully developed, turbulent flat-plate flow under downstream conditions obtained from the Prandtl-Meyer expansion. The predicted $\sigma_p$ by Laganelli and Martellucci [49] are shown as the short horizontal lines in Fig 4.10. It can be seen that $\sigma_p/p_w$ was about 45 and 60 percent below the turbulence predictions for the respective weaker and stronger expansions. Also, $\sigma_p/q_{\infty}$ was about 40 and 53 percent below the prediction for 2.5° and 4.25° corners. It is obvious that the pressure fluctuations for a stronger expansion flow would take a longer distance to recover to the presumed ultimate turbulence level. Further, the decrease of surface pressure fluctuation may indicate a nonequilibrium boundary layer. Also, in Fig. 4.10c, the pressure fluctuations are normalized by the incoming standard deviation $\sigma_{p,\infty}$. It can be seen that the surface pressure fluctuations damped to about 37 and 21 percent of the incoming value for 2.5° and 4.25° corners respectively at $\tilde{x} = 4$. The data again
Figure 4.10: Surface pressure fluctuation distribution, expansion corners
indicated an incomplete relaminarization process.

In addition, the space-time correlation of the surface pressure fluctuations was also examined, Fig 4.11. The correlations were obtained with the first transducer located at \( x = 1.25 \delta_o \) and transducer spacings of \( \xi = 0.5 \delta_o \) and \( 1.0 \delta_o \). The cross-correlations downstream of the expansion appeared broadly similar in shape as that of the flat plate as shown in Fig. 4.4. Also, the convection velocity was determined to be \( U_c/U_\infty = 0.71, 0.79 \) for the 2.5° corner and \( 0.55, 0.60 \) for the 4.25° corner with \( \xi = 0.5 \delta_o, 1.0 \delta_o \) as shown in Fig 4.12. There appeared to be a definite increase of \( U_c \) with transducer spacing \( \xi \). However, due to limited measurements, the effect of expansions on the convection of the pressure fluctuations is not entirely clear.

The individual transducer data were further correlated by themselves to give the autocorrelation \( R_{pp}(\tau) \). The present data are plotted against a normalized convection distance \( \tilde{D}_c = \tau U_c/\delta_o \). The use of \( \tilde{D}_c \) invoked the Taylor hypothesis, where \( \tilde{D}_c \) represents the distance in terms of \( \delta_o \) that a disturbance is convected at a velocity \( U_c \). The autocorrelations of the surface pressure fluctuations \( R_{pp}(\tau) \) are plotted in Fig. 4.13 for both corners, where \( U_c (= 0.65 U_\infty) \) and \( \delta_o \) were assumed constant even though they were expected to increase through the expansions. In this figure, the solid line represents the autocorrelation upstream of the corner, and the symbols connected by the dotted lines are the downstream autocorrelations. The autocorrelations showed shapes typical of those encountered in the surface pressure measurements at low Mach numbers, namely a rapid decrease followed by a shallow negative lobe and a subsequent shallow positive lobe. The autocorrelations for both corners did not show any distinct trends. It was thought that this may be due to the weak expansion encountered. Also, the autocorrelations for larger \( \tilde{D}_c \) may not be well represented due to the short-duration of the test period and the bandwidth limitation.
Figure 4.11: Space-time correlation, expansion corners
Figure 4.12: Convection velocity, expansion corners
An integral scale was estimated from the autocorrelations. The integral scale was estimated to be about $4.5\delta_o$ and $6.7\delta_o$ for 2.5° and 4.25° corners respectively, with a scatter of ±20 and ±10 percent. Physically, the integral scale is $75\pm15\mu s$ and $110\pm10\mu s$ respectively. Despite the difficulties associated with data bandwidth, the trend of the data indicated that the large-scale surface pressure fluctuations maintained their identities longer for the stronger expansion.

4.2.3 Boundary Layer Surveys

Pitot profiles measured from $x = 0.75-2.75\delta_o$ (19-70 mm or 0.375-1.375 in.) are shown in Fig. 4.14. Although it would be extremely desirable to obtain profiles further downstream of the corner, the most downstream measurement station was limited by the test model. Self-similarity of Pitot pressure profiles was not observed for both expansion corners which implied that the boundary layer was not in equilibrium within the measurement range. This supports the observation deduced from mean and surface fluctuation pressure distributions.

Further, the normalized Mach number profiles and velocity profiles were obtained and displayed in Figs. 4.15 and 4.16. The velocity profiles for both expansion corners showed that there was a decrease in fullness downstream of the corner, which indicated that a new sub-boundary layer may be present. Further, the velocity profiles for the 4.25° expansion corner showed a slower tendency for the relaxation of boundary layer within the measurement range. It is thought that a slower response at the initial stage of boundary layer development for a stronger expansion flow may be due to a longer relaxation distance required for a highly expansive flow to reach a "new equilibrium" flow. In other words, the boundary layer for a stronger expansion flow exhibited a gradual change compared with that of the weaker expansion. This
(a) 2.5° expansion angle.

(b) 4.25° expansion angle.

Figure 4.14: Pitot pressure profiles, expansion corners
(a) 2.5° expansion angle.

(b) 4.25° expansion angle.

Figure 4.15: Mach number profiles, expansion corners
Figure 4.16: Velocity profiles, expansion corners
supports the observation from the mean surface pressure measurements which showed a longer downstream influence for a stronger expansion flow.

From the profile data, boundary layer length scales were obtained and plotted in Fig. 4.17a–c. The boundary layer thickness \( \delta \) was generally higher downstream of the corner. The increase in \( \delta \) was mainly due to a reduction of density through the expansion [16]. Further, the distributions of momentum and displacement thickness followed the shape of the distributions of \( \delta \). The changes of \( \delta^* \) and \( \theta \) were more significant than those of \( \delta \). It is considered that the increase of \( \delta^* \) and \( \theta \) was due to an increase of \( \delta \) and the presence of an accelerated boundary layer. Also for \( \alpha = 2.5^\circ \), the increase of \( \delta^* \) and \( \theta \) was larger than that of \( \alpha = 4.25^\circ \), which indicated a decrease in fullness of velocity profiles downstream of the corner for a weaker expansion flow.

4.3 Oblique Shock/Boundary Layer Interaction

The experimental results of shock impingement with the flat plate flow are discussed in this section. The shock impingement position, based on inviscid shock calculation, was located at 0.77 m (30.25 in.) from the flat-plate leading edge. The influence of shock strength was examined through the mean and fluctuating pressure distributions. Pitot pressure profiles were also obtained.

4.3.1 Upstream Influence

The surface pressure distributions for two test cases are shown in Fig. 4.18. The surface pressure normalized by the upstream reference pressure \( p_\infty \) is plotted against \( x/\delta_o \) where \( x \) is the streamwise location with respect to the inviscid-shock impingement position. The inviscid pressure distributions are also shown for comparison. In
Figure 4.17: Boundary layer length scales, expansion corners
Figure 4.18: Surface pressure distributions, $\beta = 2^\circ, 4^\circ$
Fig. 4.18, no distinctive kink or pressure plateau, which are associated with incipient separation or fully-separated flow [22], are seen. Therefore, attached flows existed for both 2° and 4° shock generators. In previous studies [28, 39, 56], the incipient separation is observed as being a function of $M_\infty$, with the resistance to separation increasing with Mach number. In the present study, the pressure ratios $p_F/p_\infty$ of 2.1 and 4.0 for the 2° and 4° shock generators respectively were below the incipient separation pressure ratio $p_{sep}/p_\infty \approx 6.0$ [56].

Further, as discussed in Section 2.3, the interaction length, which is defined as the distance between the upstream influence and the downstream influence, for a given Mach number increases with shock strength. In Fig 4.18, the interaction length for 4° shock generator ($\approx 5.5\delta_v$) is comparatively larger than that of 2° shock generator ($\approx 2\delta_v$). It is noted that slight irregularity of the downstream surface pressure for the 2° shock generator may be due to the uncertainty in setting the shock generator between runs. Also, the upstream influence scale obtained from the surface pressure distribution is shown in Fig. 4.19a, in which the upstream influence scale $x_u$ normalized by the incoming boundary-layer momentum thickness ($\delta_v^* = 0.44$ mm) is plotted against the inviscid pressure ratio $p_F/p_1$. The results of Reda and Murphy [58] are shown for comparison. For an attached flow, the normalized upstream influence scale for the 2° shock generator was approximately the same as Reda and Murphy's. However, the upstream influence scale increased drastically when the flow is separated. Therefore, for the 4° shock generator, $x_u/\delta_v^*$ is comparatively smaller than that of Reda and Murphy, in which the flow is fully separated for a given pressure ratio of 4.0.
Figure 4.19: Upstream influence, $\beta = 2^o, 4^o$
4.3.2 Unsteadiness of Interactions

The distributions of surface pressure fluctuations for $\beta = 2^\circ$ and $4^\circ$ are shown in Fig. 4.20a–c. The pressure fluctuations $\sigma_p$ was normalized by the local mean pressure $p_w$, upstream rms pressure $\sigma_{p,r}$ and upstream dynamic pressure $q_{\infty}$ respectively. The normalization emphasizes the relative magnitude of the pressure fluctuations against the normalizing factor. The general characteristics of the rms distribution were typical of some previous studies [25, 35, 53]. Fig. 4.20a showed that the intensity of pressure fluctuations $\sigma_p / p_w$ increased rapidly downstream of the upstream influence and reached a maximum. Then $\sigma_p / p_w$ decreased gradually and tended to the normalized rms pressure fluctuation of the upstream, undisturbed boundary layer. A new equilibrium rms level was obtained about 2 to 3 $\delta_\alpha$ downstream of the inviscid shock impingement position. The data also showed the dependence of the pressure fluctuation upon the shock strength. The maximum of the rms pressure fluctuation for $\beta = 4^\circ$ was comparatively larger than that of $\beta = 2^\circ$. Moreover, the rms pressure fluctuation for the $2^\circ$ shock generator tended to approach an equilibrium condition more quickly, which indicated a smaller interaction region for a weaker shock strength. Also in Fig. 4.20b–c, the rms pressure fluctuations increased at the start of the interaction and were followed by a plateau region. It can be seen that the surface pressure fluctuations $\sigma_p / q_{\infty}$ in the plateau region were about 0.3 and 0.7 percent for $\beta = 2^\circ$ and $4^\circ$ respectively, where $\sigma_p$ of the incoming flow was about 0.2 percent of the freestream dynamic pressure $q_{\infty}$. It indicated that the amplification of pressure fluctuations in a hypersonic flow was significant even for a small deflection angle. The increase in the rms level as observed has been thought to be a manifestation of the amplification of turbulence through the interaction region [72].

The "peak" of the rms pressure fluctuation in Fig. 4.20a is a common character-
Figure 4.20: Pressure fluctuation distributions, flat plate
istic of shock wave/boundary layer interactions in both two- and three-dimensional flows. Further, as highlighted in Section 2.1, the rms pressure fluctuation $\sigma_p/p_w$ is a function of $M_{\infty}^2$ for an equilibrium, cold-wall turbulent boundary layer. This implies that $\sigma_p/p_w$ for an undisturbed, hypersonic turbulent boundary layer is larger than that for a supersonic flow. In Fig. 4.21, the normalized maximum rms pressure fluctuation $\tilde{\sigma}_{p,\text{max}}$ is plotted against shock strength $p_F/p_1$, where $\tilde{\sigma}_p$ is $(\sigma_p - \sigma_{p,o})/p_{w,o}$. The collapse of the present data and Dolling & Or's [25] results was fairly good. This may indicate that the “peak” of the rms pressure fluctuation is mainly a function of the shock strength $p_F/p_1$.

Further, the “peak” of the rms pressure fluctuation is thought to be an indication of strong intermittent behavior at the start of the interaction [72]. This intermittent nature can be further examined through the probability density function PDF. The results are summarized in Figs. 4.22 and 4.23. The PDF for $\beta = 2^\circ$ was skewed with the most probable value at half a rms count below the mean level. The surface pressure signal was distributed more randomly further downstream. For $\beta = 4^\circ$, the PDF exhibited a similar feature. However, the distribution was more skewed near the upstream influence line ($x = -2.5\delta_o$) with the most probable value at about a quarter rms count below the mean level. Further downstream ($x = -1.5\delta_o$), the PDF was bimodal. The surface pressure signal alternated between two maxima of the probability curve ($-0.75$ and $0.25 \sigma_p$ respectively). The PDF near the inviscid shock impingement location was nearly Gaussian with the most probable value slightly above the mean level, which agreed the trend of Dolling and Or [25].
Figure 4.2: Maximum pressure fluctuation, flat plate
Figure 4.22: PDF, $\beta = 2^\circ$
Figure 4.23: PDF, $\beta = 4^\circ$
4.3.3 Pitot Pressure Surveys

The Pitot pressure profiles measured from 1.27-3.81 cm (0.5-1.5 in.) downstream of the shock impingement position are shown in Fig. 4.24. Distance and pressure are normalized with respect to the upstream boundary layer thickness and Pitot pressure at the boundary layer edge. The normalized Pitot pressure profiles showed a similar feature as that of previous studies [37, 79], such as a pressure jump across the reflected shock. Further, in the present study, the reflected shock waves were imbedded inside the boundary layer. From the Pitot pressure profiles, a “kink” associated with the pressure jump across the shock wave can be seen near the test surface as shown by the dotted line. The reflected shock angles were estimated to be about 6°, 9° ± 0.5° for β = 2°, 4°, whereas the inviscid reflected shock angles were 9° and 11° respectively. The discrepancy may be due to the spatial resolution of measured boundary layer profiles.

Further, the boundary layer thicknesses through the interactions are plotted in Fig. 4.25. It can be seen that δ showed a decrease downstream of the interactions, which was mainly due to the pressure rise associated with the weak interactions [68].

4.4 Mutual Interactions of Shock/Expansion Flows

The mutual influence of impinging shock wave/expansion corner in a turbulent boundary layer is discussed in this section. The expansion corners, α = 2.5° and 4.25°, were located at 0.77 m (30.25 in.) from the leading edge of the flat plate. Interactions were generated by external shock generator in the form of a 2° or 4° sharp wedge. Also, the shock impingement positions were adjusted at the corner or one boundary layer thickness upstream and downstream of the corner. Twelve distinct test cases with surface pressure measurements and limited Pitot surveys were examined.
a. Flat plate, $\beta = 2^\circ$

b. Flat plate, $\beta = 4^\circ$

Figure 4.21: Pitot pressure profiles, $\beta = 2^\circ, 4^\circ$
Figure 4.25: Boundary layer thickness, $\beta = 2^\circ, 4^\circ$
4.4.1 Surface Pressure Distributions

The surface pressure distributions were normalized by the freestream static pressure and are plotted in Figs. 4.26-4.28. The pressure distributions for the $\alpha = 2.5^\circ$ expansion corner are shown in the top half of each figure while those on the bottom half are for the $\alpha = 4.25^\circ$ case. In each figure, pressure distributions for two deflection angles ($\beta = 2^\circ$ and $4^\circ$) are presented. The incident shock was assumed to impinge at the exact location based on inviscid calculations. The inviscid pressure distributions are shown as lines for comparison.

The surface pressure distributions with the incident shock upstream of the expansion corner are shown Fig. 4.26. For $\alpha = 2.5^\circ$ and $\beta = 2^\circ$, the agreement with inviscid pressure downstream of the corner was poor which may be due to the uncertainty with setting up the shock generator. Otherwise, it can be seen that the surface pressure gradually increased at first. The surface pressure downstream of the expansion corner reached the inviscid value in a relatively shorter distance ($x_D = 1-2 \delta_s$) compared with an expansion corner flow without shock impingement ($x_D = 5-6 \delta_s$, Fig. 4.8). The downstream influence appeared to decrease for a stronger overall interaction characterized by $p_F/p_1$. Also the peak pressure rise for a given deflection angle ($\beta = 2^\circ$ or $4^\circ$) decreases for a stronger expansion, e.g. for $\beta = 4^\circ$, the peak pressure rise $p_{w, pk}/p_\infty \approx 3.4$ and $2.6$ for $\alpha = 2.5^\circ$ and $4.25^\circ$ expansion corners respectively. This implied that there were mutual influences between the incident shock and the expansion. These influences depended on the overall inviscid pressure ratio $p_F/p_1$.

For a boundary layer with shock impingement right at the corner, the pressure distributions are shown in Fig. 4.27. The agreement with the inviscid pressure distribution was fair. Under certain conditions, the reflected shock wave at the corner was
(a) 2.5° expansion corner.

(b) 4.25° expansion corner.

Figure 4.26: Surface pressure distributions, $x_{s_h} = -1$
Figure 4.27: Surface pressure distributions, $\bar{x}_{sh} = 0$
Figure 4.28: Surface pressure distributions, $\bar{x}_{sh} = 1$
nearly neutralized by the Prandtl-Meyer expansion, which agreed with the observation of Chew [16]. It is also noted that the surface pressure downstream of the 2.5° expansion corner showed a better agreement with the inviscid pressure distribution than that of the 4.25° expansion corner. For $\alpha = 4.25^\circ$ and $\beta = 2^\circ$ ($p_F/p_\infty = 0.95$ for this case), the surface pressure distribution showed a longer downstream influence. It indicated that a stronger favorable pressure gradient affected the surface pressure to a longer extent downstream of the corner.

When the shock impinged downstream of the corner, the surface pressure distributions, Fig. 4.28, showed a smaller upstream influence measured from the inviscid shock impingement location. Also, the surface pressure with 4.25° expansion for both deflection angles showed an overshoot compared with inviscid pressure levels, which agreed with the observation of Chew [16]. It was thought that the overshoot pressure is due to the mutual interactions of incident shock and expansion fan. For an expansion-shock flow, the shock strength decreased as the incident shock crossed the expansion fan. The interactions upstream of the shock impingement location had a strong effect on the surface pressure distribution. It was also interesting to note that the surface pressures for all four test cases did not clearly indicate a pressure drop as shown in the inviscid pressure distributions. The surface pressure increased gradually due to the influences of the expansion process and the upstream propagation of disturbances.

The upstream influence scale $\bar{x}_u$ is summarized in Fig. 4.29, in which $\bar{x}_u$ is plotted against the shock impingement position $\bar{x}_{sh}$, where $\bar{x}_{sh} \equiv x_{sh}/\delta_o$. It can be seen that the upstream influence increased for stronger shock strength ($\beta = 4^\circ$) as expected. As the shock moved downstream, the upstream influence decreased, particularly for $\beta = 4^\circ$. This indicated that the pressure rise induced by the incident
Figure 4.29: Upstream influence, expansion corner with shock impingement
shock was effectively neutralized by the expansion. Further, the upstream influence for a given deflection angle decreased with a stronger expansion (α = 4.25°). In other words, a stronger expansion further weakened the pressure rise associated with the incident shock. In the following section, the mutual influence of expansion and shock wave is discussed in more detail through an analysis of pressure fluctuation data.

### 4.4.2 Surface Pressure Fluctuations

Figs. 4.30–4.35 show the rms distributions of the pressure fluctuations. The pressure fluctuation $\sigma_p$ was normalized by the local surface pressure $p_\infty$, the upstream rms pressure $\sigma_{p,0}$ and the freestream dynamic pressure $q_\infty$.

The pressure fluctuation distributions, Figs. 4.30a and 4.31a, with impinging shock upstream of the corner showed some similar features to one another, e.g., the "peak" rms pressure fluctuation associated with the strong intermittent behavior of the interaction, a damping downstream of expansion corner and an increase at $x = 3.3\delta_o$ and $4.2\delta_o$ for $\alpha = 2.5^\circ$ and $4.25^\circ$ respectively. The increase of the pressure fluctuations may imply that the boundary layer may be recovering to a new equilibrium state. Also, as discussed in Section 4.2.2, the $\sigma_p$ data for an expansion flow without shock impingement did not show any asymptotic behavior or an increase within the measurement range. With impinging shock, the downstream influence of the expansion process decreased. Further, it is interesting to compare the fluctuations with that of the flat plate flow with shock impingement. In Fig. 4.31c, $\sigma_p/q_\infty$ shows a decrease downstream of the expansion corner. At $x = 2.5\delta_o$, $\sigma_p/q_\infty$ was 0.0015 and 0.0038 for $\beta = 2^\circ$ and $4^\circ$ respectively. The pressure fluctuations are only about 50 percent of the flat plate value (Section 4.3.2). The attenuation induced by the favorable pressure gradient is clear.
Figure 4.30: Pressure fluctuation distributions, $\alpha = 2.5^\circ$, $\bar{x}_{sh} = -1$
Figure 4.31: Pressure fluctuation distributions, $\alpha = 4.25^\circ$, $\bar{x}_{sh} = -1$
Figure 4.32: Pressure fluctuation distributions, $\alpha = 2.5^\circ$, $\bar{x}_{sh} = 0$
Figure 4.33: Pressure fluctuation distributions, $\alpha = 4.25^\circ$, $\bar{z}_{sh} = 0$
Figure 4.34: Pressure fluctuation distributions, $\alpha = 2.5^\circ$, $\bar{x}_{sh} = 1$
Figure 4.35: Pressure fluctuation distributions, $\alpha = 4.25^\circ$, $\bar{x}_{sh} = 1$
Furthermore, as the incident shock impinged right at the expansion corner, the distributions of pressure fluctuations are shown in Figs. 4.32a and 4.33a. The characteristic shapes were basically the same as that of shock impingement location upstream of the expansion corner. However, the "neutralization" between the reflected shock and expansion tended to reduce the rms "peak". For $\alpha = 4.25^\circ$ and $\beta = 2^\circ$, $(\sigma_p/p_\infty)_{\text{max}}$ was about 20 percent below the peak value for the impinging shock upstream of the corner. Also, in terms of the dynamic pressure $q_\infty$ and the upstream pressure fluctuation $\sigma_{p,0}$, the distribution of pressure fluctuations for $\alpha = 2^\circ$ showed a distinct behavior for both deflection angles in which there were a plateau region downstream of the corner and a damping at $x = 3.3\delta_0$. The presence of the plateau region of pressure fluctuation may imply that the interaction was mainly dominated by the shock strength over this particular weaker expansion.

When the impinging shock moved downstream of the corner ($\tilde{z}_{sh} = 1$), the pressure fluctuation distributions, Figs. 4.34a and 4.35a, showed similar shapes again. However, in Figs. 4.34b-c, the pressure fluctuation for $\alpha = 2.5^\circ$ and $\beta = 2^\circ$ showed a plateau region downstream of the impinging shock and the influence of the expansion was not completely clear. Also, in Figs. 4.34b and 4.35b, the downstream influence between $(\sigma_p/\sigma_{p,0})_{\text{max}}$ and the recovery distance of the boundary layer tended to be shorter compared with that of impinging shock upstream of the corner. For $\beta = 4^\circ$, the downstream influence was about $2.25\delta_0$ and $3.5\delta_0$ for $\alpha = 2.5^\circ$ and $4.25^\circ$ respectively while the interaction region was about $3.5\delta_0$ and $4.0\delta_0$ for impinging shock upstream of the corner. The reduction of interaction length implied that the influence of the incident shock was weakened by the expansion fan as the shock impinged downstream of the corner. Further, it is interesting to note that Chew's study [16] indicates an independence between the expansion and the incident shock flows at $x_{sh} \approx 1.5-2.5 \delta_0$, in which the pressure drop due to the expansion fan is
completed before the impinging shock. In the present hypersonic flow, the surface pressure decreased in a more uniform fashion due to the highly-swept expansion fan. The independence in a hypersonic flow may be achieved only when the impinging shock is further downstream of the corner.

The "peak" of pressure fluctuations \((\sigma_p/p_w)_{\text{max}}\) is summarized in Fig. 4.36a. It is plotted against the normalized shock impingement position \(\tilde{x}_{sh}\). It is clear that the peak of the pressure fluctuation decreases as the impinging shock moves downstream. Also, the rms distributions of \(\alpha = 4.25^\circ\) for both deflection angles appeared to be more uniformly attenuated compared with that of \(\alpha = 2.5^\circ\), which indicated the influence of expansion corner on the interaction. A further examination of the amplification of pressure fluctuations in term of dynamic pressure is plotted in Fig. 4.36b. For shock impingement upstream and at the corner, \((\sigma_p/q_\infty)_{\text{max}}\) was roughly the same. The amplification of pressure fluctuations was mainly dominated by the shock motion.

Further, examples of normalized probability density functions PDFs are plotted in Figs. 4.37 and 4.38. The PDFs showed a highly skewed upstream signal and a bimodal distribution of pressure fluctuation further downstream, which indicated the intermittent nature of the signals. Also, the PDFs of \(\beta = 4^\circ\) showed a further upstream propagation of disturbances than that of \(\beta = 2^\circ\). This indicated a larger upstream influence for a stronger shock strength.

### 4.4.3 Pitot Pressure Measurements

Pitot pressure measurements were performed for only two test conditions where the oblique shock (2° and 4° deflection angle) impinged right at the 2.5° expansion corner. Pressure and distance were normalized by the Pitot pressure \(p_{\text{pit,e}}\) at the boundary layer edge and the upstream boundary layer thickness \(\delta_o\) respectively. The
Figure 4.36: Maximum pressure fluctuation, expansion corners
Figure 4.37: PDF, $\alpha = 2.5^\circ$, $\beta = 2^\circ$, $\bar{x}_{sh} = 1$
Figure 4.38: PDF, $\alpha = 2.5^\circ$, $\beta = 4^\circ$, $\bar{x}_{sh} = 1$
normalized Pitot pressure profiles measured from $x = 0.75-2.75 \delta_0$ (4.8-35 mm or 0.375-1.375 in.) downstream of the corner are shown in Fig. 4.39. Compared with the profiles of a flat-plate flow with impinging shock (Section 4.3.3), the profiles for both deflection angles showed a fairly similar shapes. The "kink" associated with the reflected shock waves was seen near the test surface but with less strength compared with the flat-plate flow with shock impingement, which may be due to the "neutralization" of reflected shock waves by the corner. It may also be noted that the assumption of constant static pressure across the boundary layer may not hold for this complicated flow [5]. Therefore, no velocity profiles are presented here.

Further, the distributions of boundary layer thickness are shown in Fig. 4.40. The boundary layer thickness appeared to decrease downstream of the corner. For flows with the presence of the expansion corner, the boundary layer thickness over the interaction region was larger than that of the flat-plate flow with impinging shock. The increase of $\delta$ was due to the effect of the expansion corner, in which $\delta$ increased owing to the reduction of density through the expansion fan (section 4.2.3). Moreover, the boundary layer thickness for $\beta = 4^\circ$ was generally lower than that for $\beta = 2^\circ$. This resembled the trend as that of a simple impinging-reflecting shock system for a weak interaction.
Figure 4.39: Pitot pressure profiles, $\bar{x}_{sh} = 0$
Figure 4.40: Boundary layer thickness, $\bar{x}_{th} = 0$
CHAPTER 5

CONCLUSIONS AND RECOMMENDATIONS

5.1 Conclusions

Experiments were carried out to study the mutual influence of shock waves and expansion fans in a hypersonic flow. The interactions were generated by oblique shocks impinging near expansion corners with the aim of understanding the effect of shock strength, expansion and shock impingement location on the interactions. All tests were conducted at a nominal Mach number of 8 with a Reynolds number of $10.2 \times 10^6$ per meter. The stagnation pressure and temperature were 5.38 MPa and 800 K respectively and the experiments were performed under cold-wall conditions, $T_w/T_o \approx 0.35$.

The interactions were characterized by measuring mean and fluctuating surface pressure. Pitot pressure surveys were also performed. The conclusions from the present investigations, which are discussed in Chapter 4, are summarized as follows:

1. The downstream influence length of expansion corner flows was found to scale with a hypersonic similarity parameter $K$ for a wide range of supersonic and hypersonic Mach numbers, indicating that such flows can be treated as primarily rotational and inviscid.

2. The surface pressure fluctuations of hypersonic turbulent flow downstream of small expansion corners were found to be normally distributed through the expansion process but were severely attenuated.
3. The boundary layer showed a decrease in fullness of the velocity profiles downstream of the expansion corners.

4. The interactions of the flat plate flow with shock impingement were unsteady. The unsteadiness was characterized by an intermittent region at the beginning of the interactions. The amplification of turbulence was significant even for a small angle of attack in a hypersonic flow. The collapse of normalized rms pressure fluctuation peak for supersonic and hypersonic Mach numbers indicated that the rms peak of the interactions was mainly controlled by the shock strength.

5. The upstream influence scale decreased as the shock moved downstream of the corners. This suggested that the pressure rise induced by the impinging shock was effectively neutralized by the expansion.

6. A similar shape of the rms pressure fluctuations was obtained at all different test configurations. It was characterized as a rms peak associated with the strong intermittent behavior of the interaction, a damping downstream of expansion corners and an increase at $x = 3 - 5\delta_\infty$. The increase of pressure fluctuations may imply the start of a recovery from the expansion.

7. The rms peak of the pressure fluctuations decreased as the impinging shock moved downstream. This indicated a strong influence of the expansion on the unsteadiness of the interactions.

5.2 Recommendations

The present investigation has gained several interesting conclusions and some subsequent studies are needed for verification:

1. The possibility of a limiting downstream influence scale for an expansion flow
may imply a Mach number independence of strong expansion. An analogy of free-interaction in strongly separated flows may exist, in which the expansion process is independent of pressure ratio or other pressure gradient parameters if the expansion is strong enough. Further investigation is needed to study this asymptotic downstream influence behavior.

2. The effect of shock strength, expansion fan and impinging shock location has been examined, but only at fixed Reynolds number and wall temperature condition. Further study is necessary to check their effect on the overall interaction. Also, it would be of interest to carry out measurements as the shock impinges further upstream or downstream of the expansion corner to study the mutual influence of impinging shock wave/expansion fan.

3. The present investigation examined the weak interactions only. Further study with much stronger impinging shocks, that induces separation, and matching expansion corners can be carried out to investigate the influence of interaction scales by the expansion corner.
BIBLIOGRAPHY


