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Experimental Investigation of Crossing Shock Wave-Turbulent Boundary Layer-Bleed Interaction

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EXPERIMENTAL INVESTIGATION OF CROSSING SHOCK WAVE - TURBULENT BOUNDARY LAYER - BLEED INTERACTION

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Abstract

Results of an experimental investigation of a symmetric crossing shock wave/turbulent boundary layer/bleed interaction are presented for a freestream unit Reynolds number of 1.68×10^7 /m., a Mach number of 2.81, and deflection angles of 8 degrees. The data obtained in this study are bleed mass flow rate using a trace gas technique, qualitative information in the form of oil flow visualization, flow field Pitot pressures, and surface static pressure measurements using pressure sensitive paint. The main objective of this test is two-fold. First, this study is conducted to explore boundary layer control through mass flow removal near a large region of separated flow caused by the interaction of a double fin-induced shock wave and an incoming turbulent boundary layer. Also, a comprehensive data set is needed for computational fluid dynamics code validation.

Nomenclature

ΣA	=	total bleed area
C_f	=	skin friction coefficient
H	=	shape factor
M	=	Mach number
P	=	Static pressure
P_t	=	Total pressure
P_{t2}	=	Pitot pressure
Re	=	Reynolds number
Q	=	sonic flow coefficient
T	=	Static temperature
T_t	=	Total temperature
U	=	axial velocity
$\dot{m}$	=	mass flow rate
x,y,z	=	cartesian coordinate system
δ	=	boundary layer thickness
δ^*	=	displacement thickness
θ	=	momentum thickness

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Subscripts

w	=	wall condition
0	=	upstream reference condition
∞	=	freestream reference condition

Introduction

Three-dimensional viscous flow phenomenon resulting from two crossing and glancing shocks interacting with a turbulent boundary layer has been recognized as one of the critical problems in many of the important propulsion components such as supersonic inlets, nozzles, and combustors. The interaction between the strong pressure gradients generated by two symmetric sharp fins and three-dimensional flows caused by flow separations may lead to flow distortion in those components and degrade the overall performance of an aircraft. In particular, the distortions introduced by boundary layer separation inside an inlet reduce the efficiency of the whole propulsion system, and, in the limit, this reduction may lead to catastrophic failure of the engine.

One of the previous computational investigations of the equal shock strengths and turbulent boundary layer interaction without bleed was done by Reddy[1], who used a time marching 3-D full Navier-Stokes code, PARC3D[2], for the Mach numbers of 3.5 and 4.0 and shock generator angles of 6 and 10 degrees. The comparison with the experimental result conducted by Hingst and Williams[3,4] showed that the prediction by CFD method in general agreed quite well with the surface pressure data for both unseparated and separated cases. But, for the separated case, the reverse flow region was predicted to be slightly upstream and extended larger than that observed experimentally. The discrepancy between the computational and experimental data in the reversed flow region could be attributed to the turbulence model which was the Baldwin and Lomax[5] algebraic model in the CFD calculation and the presence of flow trace oil in the experiment. To increase the accuracy of the computational method, it may require higher order turbulence models such as a Chien[6] k-ε two-equation model in the future CFD code validation effort. Also, Davis[7] performed an experiment for a Mach number of 3.44 and deflection angles of 2, 6, 8, and 9 degrees using various shock generator plate lengths. The results showed that the distance from the shock crossing location to the trailing edge expansion off the flat plate must be greater than the incoming boundary layer thickness to suppress the upstream influence of the expansion affecting the interaction region.

To test the effectiveness of boundary layer suction or bleed, Barnhart et. al[8]., measured stagnation pressure and flow angularities for a turbulent boundary layer crossing a single glancing sidewall shock wave subject to boundary layer suction near the interaction region for Mach 2.5 and 3.0 and an inviscid flow deflection of 8 degrees. Without the bleed, the Mach 2.5 flow case was nearly separated while the Mach 3.0 case was fully separated. With 30% boundary layer bleed for Mach 2.5 and 23% for Mach 3.0, the data showed no lines of streamline coalescence emanating from the leading edge of the fin which indicates that the separation for either case did not occur.

In this experiment, as shown in Fig. 1, the boundary layer control bleed is applied in the region of crossing shock wave/turbulent boundary layer interaction to provide favorable pressure gradients and to control flow separation. In addition to understanding the flow physics involved in these interactions, this case may provide extensive data for computational fluid dynamic (CFD) code validation. With a uniform incoming boundary layer, the boundary conditions around the computational domain are relatively simple and well defined except near the bleed holes where extra care needs to be taken to accurately simulate bleed mass flow rate. Also, for equal shock strengths the flow has a plane of symmetry reducing the required calculation volume by half.

Experimental Approach

The present investigation was conducted in the NASA Lewis Research Center 1x1 ft. Supersonic Wind Tunnel. This wind tunnel is a continuous flow facility with Mach number variation provided by interchangeable nozzle blocks. The crossing shock/boundary layer experiment is configured by using two movable shock generator plates of the same lengths of 19.93 cm at 8° of deflections and two fixed extension plates of 5.08 cm each at zero angle to the undisturbed flow. These plates span the tunnel test section and produce oblique shocks when at angle of attack to the free stream. The interaction of these shock waves with the naturally occurring boundary layer on the tunnel walls defines the experiment. The boundary layers used are those that are not subjected to the cross flow pressure gradients in the tunnel nozzles. Actuation of the shock generators is accomplished by rotating the generators so that their trailing edges touch the fixed extension plates. This allows the tunnel blockage to be kept at a minimum during start-up. For reference, the position of the leading edge relative to the upstream reference plane (x=0) and to the wind tunnel centerline (z=0) is 6.35 cm and 6.97 cm, respectively. A schematic of the test configuration with reference coordinates is shown in Fig. 1 (a).

Three rows of slanted bleed holes with diameters of 6.35 mm are mounted in the side wall. These holes are slanted at 20 degrees to the flow direction and the holes are staggered with respect to the immediate upstream row. The bleed flow exits into a plenum and from there to a exhaust line where the flow rate is measured using a trace gas technique discussed in Davis and Reichert[9]. A valve on the bleed exhaust line can be actuated to back pressure the bleed plenum to obtain varying bleed flow rates. A schematic of the bleed holes is shown in Fig. 1 (b). The uncertainty of the bleed flow rates in the present measurements is estimated to be between ±2.50% of the caculated mass-flow..

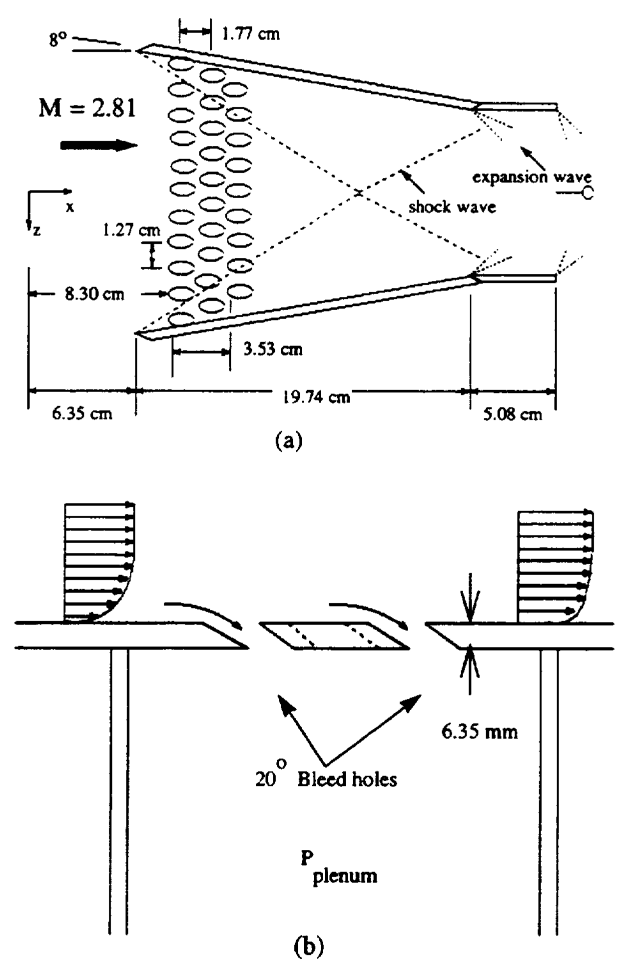


Figure 1. Schematic of crossing shock wave/ boundary layer/bleed experiment

For upstream reference conditions, the flowfield at x=0 and z=0 is measured with a traversing Pitot probe and its boundary layer parameters are summarized in Table 1. The downstream flow field measurements includes surface oil flow, mean surface static pressure, and flow field Pitot pressure surveys. The near-wall limiting stream line behavior is investigated using oil flow visualization. A powdered fluorescent dye is mixed with oil and painted on the surface between the shock generator plates. The wind tunnel is then rapidly shut down to preserve the pattern. Surface static pressure distribution is obtained using a pressure sensitive paint technique. This technique, which is described by Bencic[11], provides a complete map of the static pressure with high spacial resolution of ±0.05 psi. Pitot pressure surveys are made at the exit of the interaction model. They are obtained using single Pitot probe that was actuated through the boundary layer. The probe is physically moved in the cross plane to obtain successive profiles resulting in a complete cross plane survey.

M_∞	2.81
U_∞ m/s	596.0
$P_{w,\infty}$ kPa	7.0
$P_{t,\infty}$ kPa	192.9
$T_{t,\infty}$ K	287.6
Re /m	1.681×10^7
δ_0 mm	24.41
δ^*_0 mm	7.27
θ_0 mm	1.63
$C_{f,0} \times 10^3$	1.35

Table 1: Upstream flow conditions

Results and Discussion

Sonic Flow Coefficient

To increase average kinetic energy in the boundary layer, mass removal of low energy flow in the boundary layer through multiple bleed holes is applied so that the flow is less susceptible to separation in the presence of an adverse pressure gradient such as across shock waves. The efficiency of a bleed configuration for removal of low energy flow is often quantified by sonic flow coefficient defined as follows:

$$Q = \frac{\dot{m}}{\dot{m}_{ideal-choked}} \qquad (1)$$

where $\dot{m}_{ideal-choked}$ is computed using the following equation

$$\dot{m}_{ideal-choked} = \frac{0.5318 P_t \Sigma A}{\sqrt{T_t}} \qquad (2)$$

The experimental study conducted by Willis, et al.[10] shows that 20° bleed configuration captures a significantly larger fraction of the total pressure associated with the boundary layer than 90° configuration. In this experiment, this 20° bleed configuration is chosen, fabricated, and tested under the flow condition described above. Figure 2 shows the sonic flow coefficient, Q, as a function of the bleed plenum static pressure, P_{plenum}, normalized by freestream total pressure, $P_{t,0}$. The presence of a kink at about Q = .09 or $P_{plenum}/P_{t,0}$ = 0.04 indicates the location of near-choked flow.

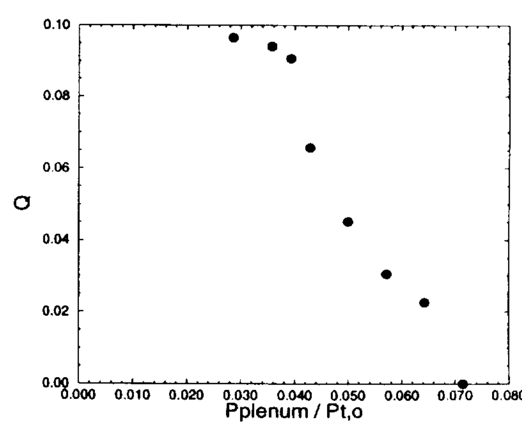
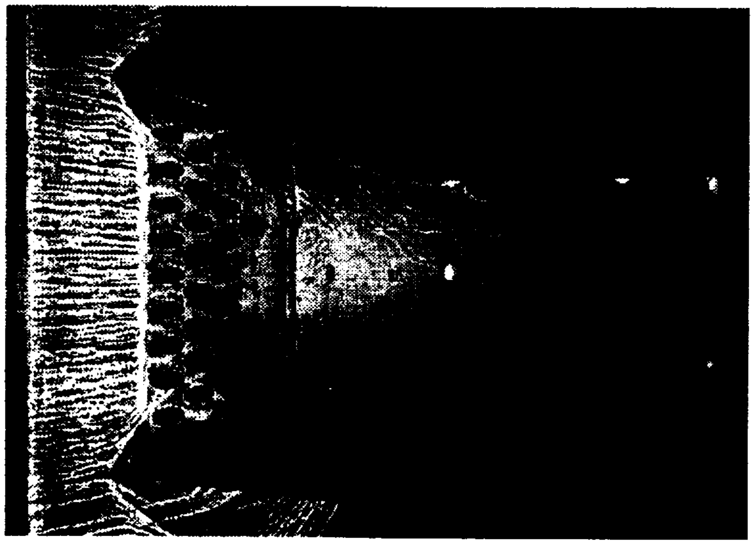


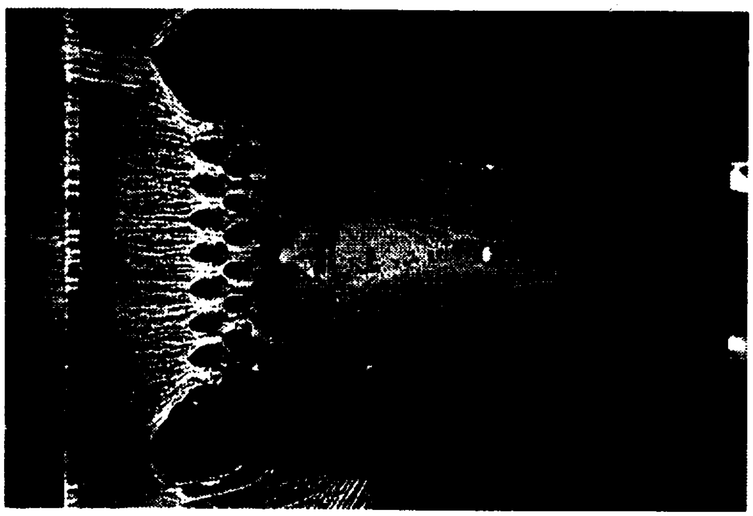
Figure 2. Sonic flow coefficient distributions

Flow Visualization

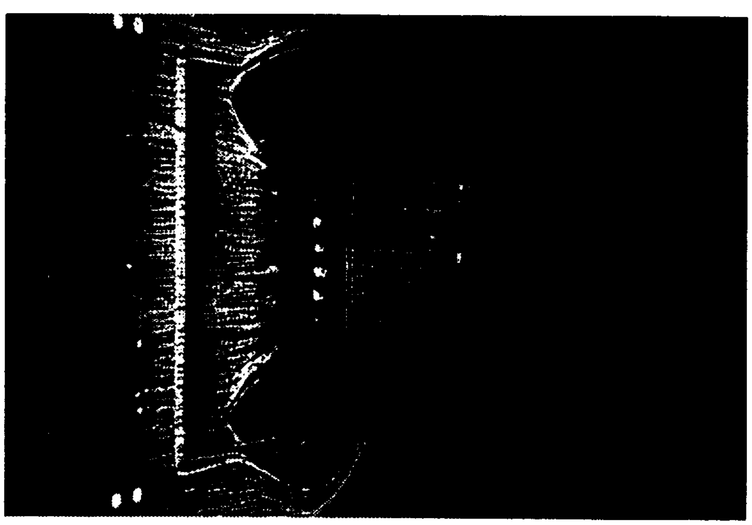
Qualitative study of surface streamline characteristics is achieved using 140 wt. oil with a powdered fluorescent dye. The oil is painted on the side wall surface between the two shock generators. In order to capture



(a) Q = 0
$P_{plenum}/P_{t,0} = 0.0714$



(b) Q = 0.0453
$P_{plenum}/P_{t,0} = 0.050$



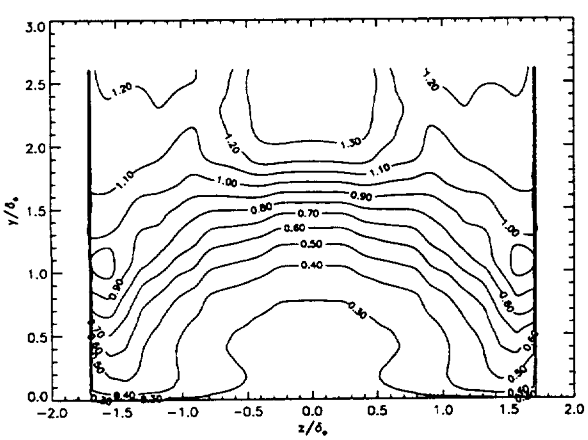
(c) Q = 0.0966
$P_{plenum}/P_{t,0} = 0.0286$

Figure 3. Visualization of surface streamline through oil flow

steady oil flow, the wind tunnel is rapidly shutdown to minimize a normal shock influence that is introduced during the normal shutdown of the wind tunnel. Fig. 3 shows the effectiveness of bleed rates on the surface streamline. With zero bleed rate, Q = 0, Fig. 3(a) indicates that there is a severe flow reversal region due to shock boundary layer interaction. In addition to this, there seems to be a reversed flow in front of first row of bleed holes caused by flow recirculation within the plenum and the holes. Fig. 3(b) with Q = 0.0453 shows weak flow reversal downstream of the bleed holes. In Fig. 3(c), with choked mass bleed rate of Q = 0.0966, there is no significant flow separation or reversal and it shows well defined flow pattern. Comparing these three oil flows, the effectiveness of bleed rate on crossing shock boundary layer interaction can be established without other measurements.

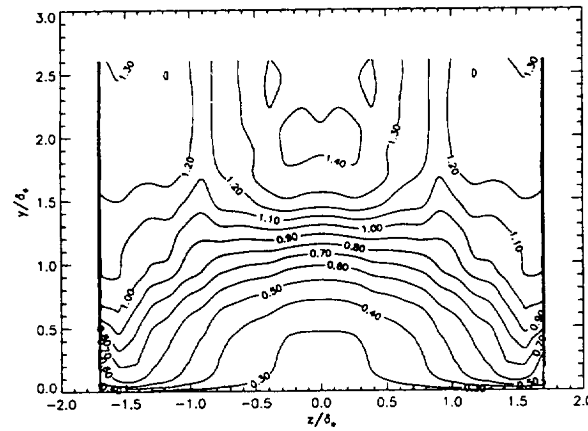
Pitot Pressure Contours

Pitot pressure surveys are conducted in the exit plane perpendicular to the side wall extension plate. The measurements are obtained in a similar way done by Davis and Hingst.[12] Fig. 4 shows results of the Pitot pressures measurements normalized by the undisturbed upstream value of $P_{t2,0}$ of 74.5 kPa for Q = 0, 0.0453, and 0.0966 that correspond to zero bleed, intermediate bleed, and choked bleed mass flow rate. For flow symmetric reason, only top half of the exit plane shown in Fig. 3 is measured. Then, during the data reduction, the figure is mirror imaged and rotated 90° counter clockwise for presentation purposes. Without any bleed, Q = 0, Fig. 4(a) shows clear evidence of flow separation near the center line, indicated by the lifting of

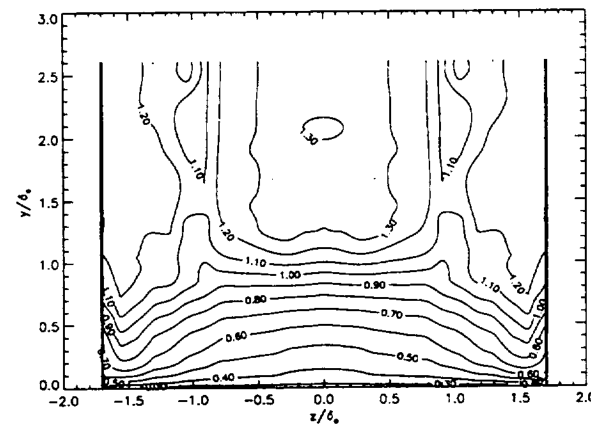


(a) Q = 0

Figure4. Pitot pressure distributions ($P_{t2}/P_{t2,0}$) (Continued . . .)



(b) Q = 0.0453



(c) Q = 0.0966

Figure 4. Pitot pressure distributions ($P_{t2}/P_{t2,0}$)

the boundary layer. As the bleed rate is increased, the boundary layer thickness is decreased and the extent of separation region is also diminished as in Fig. 4(b). With choked bleed rate of Q= 0.0966, Fig. 4(c) illustrates no significant flow separation hinted by disappearance of lifted boundary layer.

Pitot Pressure Profiles on a Center Line at Exit Plane

Boundary layer surveys at nine different plenum pressures ranging from $P_{plenum}/P_{t,0}$ = 0.02857 to 0.08036 are obtained at the cross section of the extension plate exit plane (x=26.07 cm) and symmetric plane (z=0 cm) to provide a data set for computational fluid dynamics (CFD) code validation. The measurements are done in the same way as Pitot pressure contour surveys. In Fig. 5, the Pitot pressure profiles are plotted against non-dimensional height normalized by the up-

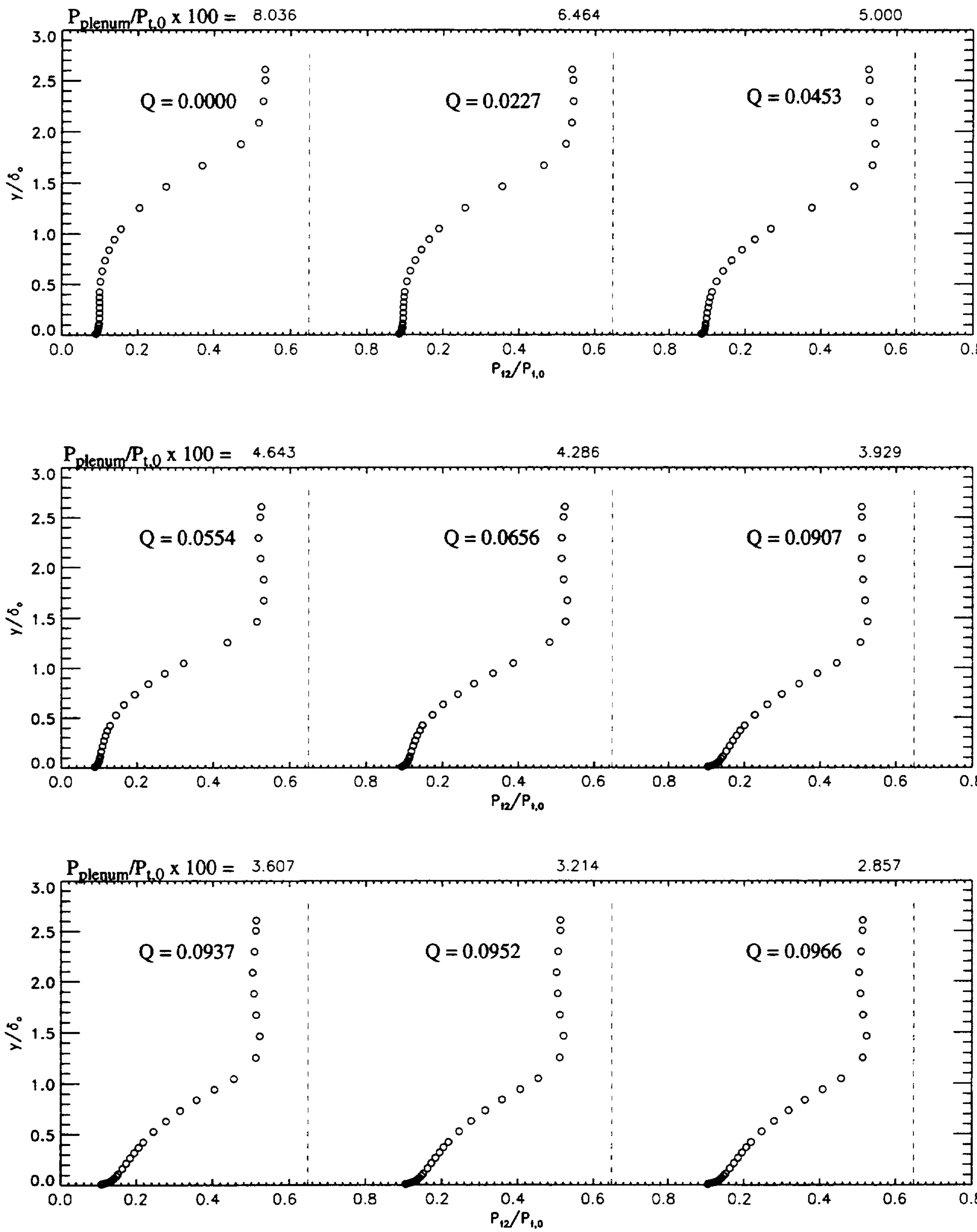


Figure 5. Pitot pressure profiles using various bleed mass flow rate at the intersection of exit plane (x=26.07 cm) and axial center line (z=0), ---- inviscid Pitot pressure ($P_{t2}/P_{t,0}$ = 0.6477)

stream boundary layer thickness of $\delta_0 = 24.41$ mm. The ratios of bleed plenum pressure and upstream total pressure are shown on the top of each profile and, for each bleed mass flow rate, the sonic flow coefficients, Q, are shown at the corresponding bleed plenum pressure ratios. For no bleed mass flow rate (Q=0), the profile shows zero pressure gradient near the wall and an inflection point, indicating the flow separation has occurred at the extension plate exit plane. These data correlate well with the results observed in the oil flow and the pitot pressure contour shown in Fig. 3(a) and Fig. 4(a). As the bleed plenum pressure P_{plenum} decreases, i.e. sonic flow coefficient Q increases, the boundary layer thickness is decreased and the inflection point is moved toward the wall. At Q=0.0453, which corresponds to the flow case of Fig. 3(b) and Fig. 4(b), the flow separation is still visible but with less severity than lower Q values. As the bleed plenum pressure decreases further, the zero pressure gradient near the wall is gradually eliminated, indicating the replacement of low energy flow with higher energy flow in the boundary layer. For choked bleed mass flow rate greater than Q = 0.0907 ($P_{plenum}/P_{t,0} < 0.03929$), there is no indication of flow separation and the profile shows a 'healthy' boundary layer shape. At Q = 0.0966 or $P_{plenum}/P_{t,0} = 0.02857$, corresponding to the fully choked mass flow rate case of Fig. 3(c) and Fig. 4(c), the boundary layer profile is very much same as the one for Q = 0.0907. From this figure, similar conclusion as in the flow visualization and the pitot pressure contour can be made about the effectiveness of bleed mass flow rate to the crossing shock boundary layer interaction. However, it is noted that there are quantitative discrepancies between analytical ($P_{t2}/P_{t,0} = 0.6477$) and measured ($P_{t2}/P_{t,0} \sim 0.52$) data in the inviscid zone which can not be explained by the author.

Surface Pressure

Surface static pressure data are obtained by using pressure sensitive paint (PSP) for various bleed plenum pressures. Fig. 6 shows six surface pressure contours normalized by upstream wall static pressure of 7.0 kPa for bleed flow conditions from no bleed to fully choked bleed flow, i.e. Q = 0 to 0.0966. For zero bleed mass flow rate of Q=0, due to severe flow separation caused by crossing shock boundary layer interaction, there is no discernible shock structures found on the contour. Also, it should be noted that there is a sharp pressure gradient in front of the first row of bleed holes which may indicate a flow recirculation within the bleed plenum and bleed holes. These phenomena seems to be occurring for all Q values less than 0.0453, but as the bleed mass flow rate is increased, the contours show decreased pressure near the bleed region and less pressure gradient in front of the bleed holes. At Q = 0.0453, or $P_{plenum}/P_{t,0}$ = 0.05, the crossing shock structure appears just downstream of the bleed holes accompanied by barrier shocks at the ends of bleed holes. As the bleed plenum pressure is further decreased, i.e. Q is increased, a more distinctive shock structure can be seen. At a choked bleed mass flow rate of Q = 0.0966 or $P_{plenum}/P_{t,0} = 0.0286$, a definite crossing shock structure is seen just downstream of the last row of bleed holes with oblique shocks emanating from the leading edges of the each sharp fin. For each static pressure contour, expansion fans are clearly visible at the end of 8^o shock generators where the flow returns to parallel to the tunnel and at the end of extension plates where the flow expands to the tunnel due to low pressure flow coming from the other side of the shock generator plates.

Conclusion

An experimental investigation of a symmetric crossing shock wave/turbulent boundary layer/bleed interaction was conducted for Mach 2.81 and shock generator angles of 8 degrees. The objectives of the experiments were to understand the flow physics that are involved in the interaction and to provide a comprehensive data set for computational fluid dynamics code validation. The investigation was done at various bleed plenum pressure to find the effectiveness of the bleed mass flow rate on the interaction by utilizing both qualitative and quantitative techniques. The qualitative results from oil flow visualization show that the flow recirculation caused by high pressure gradient across the crossing shock waves can be minimized or eliminated through the use of boundary layer bleed. For quantitative study, Pitot pressure and surface statice pressure are measured using a traversing Pitot probe and pressure sensitive paint. Even though the Pitot pressure measurements in the inviscid zone do not agree with the analytical solution, the pressure sensitive paint data show that with increasing bleed mass flow rate, the flow becomes less separated and the shock structure is more distinctive. With fully choked bleed rate, the separation is successfully eliminated and two-dimensional flow is recovered. For CFD validation purpose, the experiment provides useful data to test various boundary conditions or schemes such as bleed or turbulence modellings.

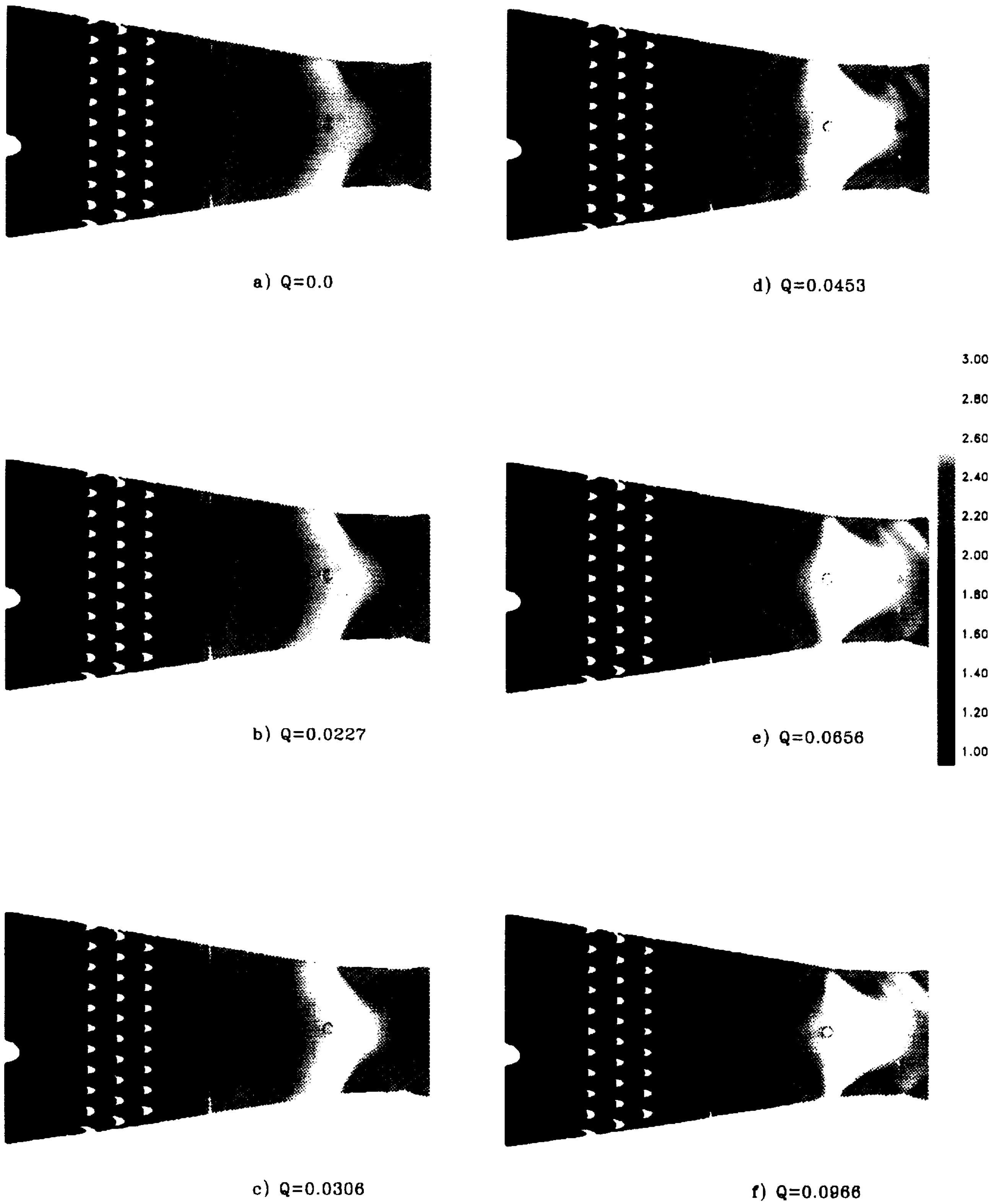


Fig. 6 Normalized surface pressure from pressure sensitive paint ($P_w/P_{w,0}$).

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