

Summary of Shock Wave Turbulent Boundary Layer Interaction Experiments In a Circular Test Section

Jonathan Sasson *

Case Western Reserve University, Cleveland, OH, 44106

Heath H. Reising[†]

HX5 L.L.C., Brook Park, OH, 44142

David O. Davis[‡]

NASA Glenn Research Center, Cleveland, OH, 44135

Paul J. Barnhart[§]

Case Western Reserve University, Cleveland, OH, 44106

A series of experiments were performed at Mach 2.5 in a 17 cm diameter circular test section to characterize an impinging/reflected shock wave turbulent boundary layer interaction generated by a cone-cylinder centerbody. The cone-cylinder centerbody generates a conical shock wave that interacts with the naturally occurring boundary layer developing on the test section wall. Three different cone angles were used in the experiment to study unseparated, incipiently separated, and separated interactions. When the cone-cylinder centerbody is positioned on the centerline, a flowfield which is two-dimensional in the mean is generated. Three dimensional interactions were also created by offsetting the cone-cylinder centerbody from the test section centerline. The results are intended to provide benchmark quality datasets for computational fluid dynamics (CFD) validation without the pitfalls inherent in rectangular configurations where corner effects prohibit a truly two-dimensional flow in the mean. The experimental measurements included surface flow visualization, wall static pressure, flowfield Pitot tube pressure, constant-voltage anemometry (CVA) normal hot-wire, and particle image velocimetry (PIV) measurements. The hot-wire measurements were used to calculate mean mass flux and total temperature profiles, mass flux and total temperature turbulence intensities, and the mass flux-total temperature correlation. The PIV measurements provide three-dimensional mean velocity measurements. Agreement between the pressure, hot-wire, and PIV measurements is established in the undisturbed upstream flowfield.

Nomenclature

a_w	Overheat parameter
C_f	Skin friction coefficient
D	Diameter
e	CVA output fluctuating voltage
E	CVA output instantaneous voltage
H	Shape factor
H_i	Incompressible shape factor
ℓ	Hot-wire length
ℓ_{OF}	Offset distance

*Ph.D. Graduate, Mechanical and Aerospace Engineering Department, AIAA Senior Member.

[†] Aerospace Engineer, Inlets & Nozzles Branch, AIAA Senior Member

[‡] Aerospace Engineer, Inlets & Nozzles Branch, AIAA Senior Member.

[§] Professor, Mechanical and Aerospace Engineering, AIAA Senior Member.

L	Length
L_0	Upstream influence length
L_C	Compression length
$\dot{m}$	Mass flow rate
M	Mach number
P	Pressure
R	Radius; specific gas constant; electrical resistance
Re	Reynolds number
t	Time
T	Temperature
T_c	CVA time constant
u, v, w	Cartesian velocity components
V	Velocity
V_w	Hot-wire voltage
X	Non-dimensional x -coordinate
x, y, z	Cartesian coordinates
$\bar{y}$	Distance normal from the wall
r, θ, x	Cylindrical coordinates
r, φ, θ	Spherical coordinates

Greek Symbols

α	Shock generator cone half-angle
β	Shock wave angle
γ	Ratio of specific heats; intermittency
δ	Velocity boundary layer thickness
δ^*	Displacement thickness
η	Temperature recovery factor
θ	Test section angle; momentum thickness
μ	Viscosity; Mach angle
Π	Boundary layer wake strength parameter

Subscripts

0	Freestream or undisturbed upstream station ($x=35.56$ cm)
1	Upstream or first value
2	Downstream or second value
aw	Adiabatic wall
bm	Bellmouth nozzle
D	Diameter
e	Boundary layer edge
eff	Effective
L	Length; probe cable
OF	Offset

P	Pitot
prb	Probe
r	Recovery value
ref	Reference value
rms	Root-mean-square
S	Static; inviscid shock impingement
SG	Shock generator
T	Wind tunnel total (stagnation) condition
TS	Test section
turb	Turbulent
u	Uncompensated; upstream of shock
w	Wall; wire
xx	Streamwise
τ	Friction value

I. Introduction

SHOCK wave boundary layer interactions (SWBLI) are commonplace in nearly all applications involving supersonic or hypersonic Mach numbers. The flowfield structure changes dramatically with the formation of complex shock patterns that replace a relatively simple, predominantly inviscid, unseparated flow structure. Great strides in both experimental and numerical techniques have been made in an effort to understand these interactions. Due to the complexity of the interaction, experimental investigations are limited to simple geometries, and often lack the detail necessary to be useful as CFD validation experiments. In 1994, Settles and Dodson [1], recognizing the key role CFD plays in turbulence modeling, reviewed 105 experiments to create a database of experiments to be used as CFD validation. The geometries evaluated are of the same general configuration as those with SWBLI occurring in engine inlets, control surfaces, and wing-body junctures. During this study, Settles and Dodson established a rigorous acceptance criteria to determine which experiments truly contribute anything useful to the understanding of SWBLI. Based on their requirements, only 19 experiments were deemed suitable as validation experiments of benchmark quality. Their recommendations for further experimentation include more complex types of "building block" interactions with emphasis on three-dimensional flowfields and turbulence data such as Reynolds stresses and velocity fluctuations.

Davis [2] previously characterized a SWBLI at Mach 2.5 in a circular test section using Pitot probes. The advantage of the circular test section is that it avoids the effects of test section geometry. Results obtained from planar configurations with side walls do not necessarily lead to two-dimensional flows. Geometries studied in rectangular wind tunnels introduce additional challenges as the sidewall boundary layer and corners affect the flow, resulting in complex three-dimensional flowfields. These effects reduce the quality of the investigation toward understanding the flow physics or as a benchmark quality validation experiment. In the same study performed by Davis [2] using oil flow with an oblique SWBLI in a rectangular wind tunnel, it was shown that centerline measurements alone are not representative of a two-dimensional flowfield for a strong, separated interaction in a rectangular test section.

These effects have also been shown in CFD studies. Sidewall interactions have been found to be more substantial than the interaction that is the focus of the experiments. These sidewall effects also frequently influence the SWBLI in the focus region. Through a series of CFD computations of oblique shock wave reflections in a rectangular wind tunnel, Benek [3] showed how corner effects affect the local separation size. The numerical results show three dimensional effects even for larger test section sizes. The turbulent model used (anisotropic $k - \omega$ model) however did not account rotating vortices in the corner flow. Even without the vortices, the magnitude of the separation was found to be related to the width of the test section similar to the experimental results from Bruce [4]. The conclusion of the numerical results was that the sidewall need to be modeled in CFD studies for a better comparison to "two-dimensional" data. The final conclusion of all these investigations is that axisymmetric configurations are the only practical way to eliminate adverse effects created by the test section.

Of the 105 experiments reviewed by Settles, only 27 involved experiments with axisymmetric test articles, and only five involved an incident shock wave, created by a cone, impinging on the test section walls. In recent years, only two

additional experiments with this configuration have been performed [5, 6]. All of these experiments summarized in Table 1. The experiments in Table 1 provide mean wall pressures, and in some cases Pitot pressures as well. The investigations in [7], [8], and [5] provide Reynold stresses, but the experiments are not considered of benchmark quality, and the flow separation studied is a weak separation.

The goal of this investigation is to provide a set of measurements of benchmark quality for CFD validation. None of the studies in the previous section meet the criteria established by Settles for CFD validation experiments [1]. In particular, they lack the necessary detail to validate Direct Numerical Simulations or Large Eddy Simulations. These types of simulations are the only ones capable of computing the fluctuating flowfield. Mean flow quantities obtained in this investigation can also be used to validate RANS calculations. These are important as they serve as initial conditions for LES. RANS calculations also play a role in the preliminary design of supersonic inlets as they allow for rapid design iterations. They are also used to quickly evaluate the effects of different configurations during wind tunnel testing.

Table 1 Previous work using conical shock generators.

Date	Authors	Test Section	Interaction	Measurements
1968	Panov [9]	Rectangular	Unseparated and separated (3-D)	1. Schlieren images 2. Oilflow visualization 3. Static wall pressures
1970	Seebaugh and Childs [10]	Circular	Unseparated and separated (axisymmetric)	1. Static wall pressure 2. Pitot pressures
1973	Rose [7]	Circular	Unseparated (axisymmetric)	1. Static wall pressure 2. Pitot pressures 3. Total temperature 4. Reynold stresses (hot-wire)
1980	Chou and Childs [11]	Circular	Unseparated (asymmetric→ cone offset from $\mathcal{Q}$)	1. Static wall pressure 2. Pitot pressures 3. Flow angles
1980	Kussoy and Viegas [8]	Circular	Separated (asymmetric→ cone at angle of attack)	1. Static pressure 2. Pitot pressures 3. Flow angles 4. Reynold stresses (hot-wire)
2000	Gai and Teh [6]	Rectangular	Unseparated and separated (3-D)	1. Schlieren images 2. Oil-flow visualization 3. Static wall pressures
2014	Hale [5]	Rectangular	Separated (3-D)	1. Schlieren images 2. Oil-flow visualization 3. Static wall pressures (Pressure sensitive paint) 4. Velocity fluctuations (PIV)

In previous experiments performed by Davis, the conical shock wave was generated by a sharp tipped 3.134 cm diameter cone-cylinder with two different half-cone angles (10° and 13.5°) aligned on the test section centerline. The experiment showed no separation with the 10° cone and incipient separation at 13.5° . To produce stronger interactions, for the present investigation, a larger centerbody (5.000 cm) and a third cone with a 16° half-angle were used. In addition to pressure instrumentation, the interaction is further studied in this investigation using a normal hot-wire to capture turbulent intensities and fluctuating mass flux and PIV to measure the mean velocity components. The results show a

significant increase in turbulent intensity through the interaction with attenuation to near incoming levels downstream of the interaction. Preliminary results from the PIV measurements are also presented herein in the incoming boundary layer and through an interaction of particular interest.

II. Experimental Setup and Instrumentation

The experimental configurations for this investigation were both a two-dimensional (axisymmetric) and three-dimensional (asymmetric) shock wave turbulent boundary layer interaction in a circular test section. The nomenclature for the coordinates and velocity components can be found in Figure 1. Figure 2 provides a basic overview of the experimental configuration. The shock generator was a cone-cylinder with half-angle, α , and cylinder radius, R_{SG} , aligned in the freestream direction. The cone half-angle was varied to create both unseparated and separated interactions. Symmetric interactions were created by aligning the shock generator with the centerline of the test section. Asymmetric interactions of constant shock strength were created by displacing the shock-generator away from the test section centerline by a distance ℓ_{OF} .

A. Facility

The tests were conducted in the 225 sq-cm Supersonic Wind Tunnel (SWT) located in test cell W6-B at NASA GRC. This facility can be configured with square (15 x15 cm), circular (17 cm diameter) or octagonal test sections. For the present investigation, 17 cm circular test sections were used as shown in Figure 3. As illustrated, the facility consists of five major components: an electric heater (not shown), a flow conditioning plenum, a bellmouth subsonic contraction, a convergent-divergent (C-D) supersonic nozzle, and circular test section. The heater incorporated a closed loop control that maintained the incoming flow at a constant total temperature of 37.8°C.

Two test sections were fabricated for the investigation. The first "conventional" test section was used for all the intrusive probe measurements and the second "PIV" test section was designed with optical access for the PIV measurements. Both test sections have an inner diameter of 17.0 cm and an overall length of 71.12 cm. The conventional test section, shown in Figures 3 and 4, were fabricated with two opposing ports (windows) to allow access for probe setup and alignment of the shock generator.

B. Shock Generator Hardware

A cone-cylinder was used to create a conical shock wave turbulent boundary layer interaction in the test section. The cylinder acted as a support arm for the cone tip, and had a diameter of 5.0 cm, the same as the base of the cone. The cone tip and support arm combined are referred to as the shock generator. The cone tip is replaceable to test multiple cone configurations (see Figure 5 inset). Three different cone half angles were used in this experiment to evaluate the shock wave turbulent boundary layer interaction at various strengths: 10.0° (unseparated), 13.5° (incipient separation), and 16.0° (separated). The length of the shock generator was also adjustable. For each cone half-angle, a spacer length was used such that the conical shock generated by the cone tip intersected the center of the test section window (within +/- 0.635 cm) when the shock generator was aligned with the centerline of the test section.

The shock generator was cantilevered from a knife-edge support located at the aft end of the test section as shown in Figure 5. The shock generator can be bolted on the knife-edge support at three distinct locations. The centerline location was used to create an axisymmetric interaction. The two other locations on either side of the centerline were used to displace the shock generator away from the test section centerline. These offsets corresponds to a distance of $R_{TS}/4$ and $R_{TS}/3$, where R_{TS} is the test section radius. The knife-edge support spans the test section and is attached to a rotatable ring. Inasmuch as it was easier to setup the probes to traverse in a single plane (BB in Figure 4), the shock generator was rotated in the test section to survey at various circumferential positions. The spacer length for a particular cone was maintained constant as the shock generator was offset and rotated around the test section. In this manner, the data gathered with this configuration can be used for validation with a single CFD study per cone half angle and offset combination.

C. Flow Visualization

Surface flow visualization techniques were used in this experimental program. A blank (non-instrumented) aluminum window was used for the flow visualization experiments. Surface flow tracing was performed using a mixture of kerosene and low viscosity oils with fluorescent dye in suspension, similar to the kerosene/lampblack method used in blow-down wind tunnels. The oil/kerosene and fluorescent dye mixtures were varied until the desired image resolution

was achieved in a series of trial runs. The wind tunnel was run for a brief time, approximately 2 to 5 minutes, then rapidly shutdown to preserve the pattern. After shut down, a very thin layer of oil and dye was left on the window. The blank window was removed from the test section and the resulting flow pattern was photographed under black light exposure, causing the florescent dye to glow. The resulting images showed detailed surface streamline traces in the asymmetric shock wave turbulent boundary layer interaction region. These streamlines traces were then used to study the resulting flow separation, and how the flow structure changed with increasing shock strength and circumferential location. Separation and reattachment traces were also able to be determined from the resulting streamline field.

D. Pressure Instrumentation

Static pressure taps were used to obtain wall pressure distributions upstream, within, and downstream of the SWBLI. From the wall pressure distribution, the upstream influence of the interaction can be determined. Static pressure was also used to discern flow symmetry around the test section, important when studying axisymmetric cases. Details of the test section static pressure taps are presented in Ref. [2]. A new addition for this investigation were the static pressure taps on one of the test section aluminum window blanks, shown in Figure 4. The window with taps, placed along the CS azimuthal location, has the highest resolution with 85 equally spaced taps (0.3175 cm) along its length (27.94 cm). The shock generator was positioned such that the taps on the window measured the static pressure distribution in the SWBLI for the desired circumferential plane. Static pressure surveys were taken every 22.5° for all three cone angles and offset positions. Static pressure surveys with the shock generator on the centerline were obtained without varying the circumferential angle. Pressures along AA, BB, and the window were constantly monitored to ensure flow symmetry.

The total pressures were measured with a Pitot tube attached to the probe sting of the traversing mechanism. The probe sting consisted of an ABS slender cone with a diameter of 1.27 cm. This slender probe sting design was necessary for the wind tunnel to start with the 13.5° and 16° shock generators. The probe sting was electrically isolated from the support rod using the ABS cone. This allowed probe wall touch to be established by electrical continuity. The probes used are shown in Figure 6. The longer, offset probe was used to reach around the shock generator sting for circumferential locations 0° and 180° as shown in Figure 7. The tip of the probes were made from 0.0508 cm stainless steel tube with 0.015 cm walls. The remainder of the probe was constructed using telescoping stainless steel tubing as shown in Figure 6.

E. Normal Hot-Wire

The probes used in this investigation were the normal wire streamlined probe TSI Model 1214 shown in Figure 8. This model was specifically designed for supersonic flow, and uses a single wire perpendicular to the flow to measure the instantaneous mass flux in the streamwise direction, $\rho(t)u(t)$, henceforth defined as $\rho u(t) \equiv \rho(t)u(t)$. The normal wire probes had an internal electrical resistance of 0.2 Ω . All probes used a platinum-coated tungsten wire with a diameter of 5 μm and an approximated an aspect ratio (ℓ/d) of 200. The length of each wire, ℓ , was then approximately 1.1 mm. These probes were mounted on a probe-sting which was connected to the traverse mechanism. The probes were then connected to a constant voltage anemometer in the control room via a 40 m long cable. The electrical resistance of the cable was 1.914 Ω . The combined resistance of the probe and the cable, R_L , was 2.114 Ω . Details of the hot-wire calibration are presented in Ref. [12].

F. Test Matrix

The test matrix used for oilflow, pressure, and hot-wire measurements is shown on Table 2. It consists of two offset positions, three distinct cone-half angles, and nine circumferential locations for the asymmetric cases. The symmetric case is independent of offset position, as the shock generator is aligned with the test section centerline. For this table the Reynolds number based on test section diameter was maintained at 4.0×10^6 .

For the symmetric cases, only the 10°-cone and the 16°-cone were studied with the normal hot-wire. No separation was expected from the 10°-cone, while the 16°-cone has been shown [10] to result in strong flow separation. Hot-wire measurements of these two locations provide insight into the turbulence structure of an unseparated and separated interaction without the effects of three dimensional secondary flows resulting from test section corners. These two cases provide extremes on either side of the interaction strength that can be used for axisymmetric code validations. The 13.5°-cone was studied only with surface and Pitot pressures to show that while it does produce a separated interaction, it is not as strong as 16°-cone, and the flow structures are similar between both cases. Surface flow visualization for these cases was previously done by Davis [2].

Table 2 Test matrix.

θ	Asymmetric ($\ell_{OF1} = R/4$)			Asymmetric ($\ell_{OF2} = R/3$)		
	$\alpha = 10$ (unsep.)	$\alpha = 13.5$ (sep.)	$\alpha = 16$ (sep.)	$\alpha = 10$ (unsep.)	$\alpha = 13.5$ (sep.)	$\alpha = 16$ (sep.)
$\mathcal{Q}$	s, p, h	s, p	s, p, h	$\leftarrow$ symmetric		
0°	o, s, p	o, s, p	o, s, p, h	s	s, p	s
22.5°	s	s	s	s	s	s
45°	s	s	s	s	s	s
67.5°	s	s	s	s	s	s
90°	o, s	o, s	o, s	s	s	s
112.5°	s	s	s	s	s	s
135°	s	s	s	s	s	s
157.5°	s	s	s	s	s	s
180°	o, s, p	o, s, p	o, s, p, h	s	s, p	s

o= oil-flow visualization

s= static surface pressure measurement

p= Pitot tube measurement

h= hot-wire measurement

The shock generator was offset from the test section symmetry axis to create asymmetry and establish circumferential pressure gradients. Surface pressures were measured every 22.5 degrees along the test section to study the effects the pressure gradients have on the interaction and resulting flow separation. All three shock generators and both offsets were used in this study. The results of the surface pressure study were used to down select asymmetric cases of interest to study with surface visualization, Pitot tubes, and normal hot-wires. For the asymmetric cases, Pitot and hot-wire measurements were only made at the 0° and 180° circumferential locations, as there was no significant turning of the flow at these locations. The flow angle was not known *a priori* for the other circumferential locations. This would result in large uncertainties in Pitot and normal hot-wire measurements.

No significant difference was found between the symmetric interaction and the asymmetric interaction at ℓ_{OF1} for the 10° shock generator. Therefore this shock generator was not further examined at ℓ_{OF2} . The 13.5° -cone was used to study the flow with Pitot measurements at ℓ_{OF2} . This location was not studied any further with a stronger shock generator or with normal hot-wires since no major changes were found in the separated interaction at this location. The benchmark data sets from the interactions at ℓ_{OF1} can be used for code validation.

G. Particle Image Velocimetry

Particle image velocimetry (PIV) measurements were obtained using the same seeding system, laser, and cameras described in Ref. 13. In that work, the seed particles were demonstrated to have a response time of around $3\mu s$ and not appreciably alter test section flow uniformity. The local seeding method allows for oil droplet seed to be placed in the region of interest without occluding the viewing windows.

The main difficulty in the current PIV application was driven by the challenges in obtaining the required optical access without disturbing the SWBLI flowfield. The solution reached for this test utilized a pair of cameras viewing from opposite sides of the laser sheet which are rotated upward and aft of the region of interest. This is depicted in Figure 9 which also shows the translating table which allowed for synchronous translation of the PIV cameras. This was done remotely during test to allow for data to be collected at multiple streamwise locations without interrupting facility operation. Further detail on the PIV viewing windows and the camera arrangement will be presented in an upcoming report which details the full PIV dataset.

III. Incoming Boundary Layer

The state of the incoming boundary layer was carefully documented in order to define the boundary conditions of the SWBLI. At minimum, the mean properties of the incoming turbulent boundary layer must be documented for

an experiment of benchmark quality. The characteristics of the incoming boundary layer play an important role in determining the behavior of the shock wave turbulent boundary layer interaction. The incoming boundary layer was characterized with Pitot and hot-wire data. Pitot probe data were used to define the mean velocity profile and the integral properties of the boundary layer. Hot-wire measurements were also used to describe mean and fluctuating quantities such as velocity, temperature, turbulence intensities, and normal stresses. The mass flux signal obtained with the hot-wire was further reduced into time independent statistical quantities and normalized frequency content to analyze the fluctuations in the boundary layer. The incoming boundary layer was defined at data station 0 at $x = 35.56$ cm for the probe data. For the PIV measurements, optical access and background effects limited the upstream station at which detailed profiles could be extracted to $x > 41.19$ cm. These locations were sufficiently upstream so as not to be influenced by the downstream interaction for all shock generators and circumferential positions. All measurements for the incoming boundary layer were taken without the shock generator installed.

All measurements were taken on the bottom of the test section (BB in Figure 4). Symmetry has been previously verified by Davis [2]. The location of the Pitot pressure and hot-wire surveys were identified by a distance from a zero reference taken as the inviscid shock impingement location for each corresponding shock generator and circumferential position. The x -coordinate was then scaled by the incoming boundary layer displacement thickness ahead of the interaction, δ_0^* , resulting in the non-dimensional parameter $X = (x - x_S)/\delta_0^*$ for axial location, where x_S is the inviscid incident shock impingement location. Before discussing the results through the SWBLI, a detailed analysis of the incoming undisturbed boundary layer is presented.

A. Total Temperature Profile

The measured total temperature profile by the hot-wire is shown in Figure 10. The profile shows a region within the boundary layer where the total temperature is greater than the freestream (i.e. plenum) total temperature. This total temperature "overshoot" is approximately 0.5%. The magnitude of the total temperature overshoot depends on the Prandtl number of the fluid. In a flow with adiabatic walls and a fluid with Prandtl number equal to 1, the energy fluxes by conduction and viscous stresses in the boundary layer are equal in magnitude and opposite in direction. As a consequence, the total temperature in the boundary layer is constant and equal to the freestream total temperature. For fluids where $Pr < 1$, the thermal diffusivity is larger than the momentum diffusivity, resulting in the observed temperature overshoot.

Figure 10 also shows the total temperature profile calculated by reducing the Pitot data in two distinct ways. The data plotted with open circles shows the resulting total temperature profile by reducing the data with the common Crocco-Busemann relation between static temperature and velocity. The profile does not capture the total temperature overshoot. Using the Whitfield-High approximation [14] between static temperature and velocity shows the total temperature overshoot. The temperature profile required to reduce the Pitot measurements to a mean velocity profile was calculated by assuming the wall is adiabatic. The adiabatic wall assumption was validated by measuring the actual wall temperature. The adiabatic wall temperature calculated using the freestream Mach number was 292.47 ± 0.93 K. The measured wall temperature was 290.7 ± 0.5 K. While the calculated and measured values are not within their margin of error, a small difference of 0.6% between the nominal values was considered insignificant. Therefore, assuming an adiabatic wall to reduce the Pitot data is justified.

B. Velocity Profile

The incoming turbulent boundary layer mean velocity profile reduced from the Pitot measurements using the Whitfield-High approximation is shown in Figure 11a. The edge of the boundary layer was defined at the $\bar{y}$ coordinate at which the velocity was equal to 0.995 times the freestream velocity ($\bar{u} = 0.995\bar{u}_e$). Linear interpolation was used to find this location. The maximum uncertainty in the mean velocity profile is 0.5%. The resulting mean velocity profile was also accurately described using the wall-wake function developed by Sun and Childs shown in Equation 1. The resulting boundary layer velocity profile was used in a least-squares curve fit using Equation 1 to determine the boundary layer thickness, δ , the skin friction coefficient, C_f , and the coefficient of the wake function, Π . The details of the least-squares curve fitting techniques used are found in Ref [15].

$$\frac{\bar{u}}{u_e} = \frac{1}{\sqrt{\sigma}} \sin \left\{ \sin^{-1} \sqrt{\sigma} \left[1 + \frac{1}{\kappa} \frac{u_\tau}{u_e^*} \left(\ln \frac{\bar{y}}{\delta} + 2 \sqrt{1 - \frac{\bar{y}}{\delta}} - 2 \ln \left(1 + \sqrt{1 - \frac{\bar{y}}{\delta}} \right) \right) - \frac{\Pi}{\kappa} \frac{u_\tau}{u_e^*} \left(1 + \cos \pi \frac{\bar{y}}{\delta} \right) \right] \right\} \quad (1)$$

where

$$\frac{u_\tau}{u_e^*} = \sqrt{\frac{C_f}{2} \frac{\sigma}{1 - \sigma}} \frac{1}{\sin^{-1} \sqrt{\sigma}} \quad \sigma = \frac{\frac{\gamma - 1}{2} M_e^2}{1 + \frac{\gamma - 1}{2} M_e^2}$$

The fit obtained using the wall-wake function is also shown in Figure 11a. The values for δ , C_f , and Π were determined from the least-squares fit. The mean velocity profiles (experimental and fit) in terms of van Driest scaled variables are shown in Figure 11b. The non-dimensional velocity shown in Figure 11b, u^+ , is the Van Driest velocity. The data shows excellent agreement with the law-of-the-wake behavior.

The boundary layer displacement thickness, δ^* , and the boundary layer momentum thickness, θ , were calculated from the following definitions.

$$\frac{\delta^*}{R} = 1 - \sqrt{1 - 2 \int_0^{\delta/R} \left(1 - \frac{\bar{\rho}}{\rho_e} \frac{\bar{u}}{u_e} \right) \left(1 - \frac{\bar{y}}{R} \right) d \left(\frac{\bar{y}}{R} \right)} \quad (2)$$

$$\frac{\theta}{R} = 1 - \sqrt{1 - 2 \int_0^{\delta/R} \frac{\bar{\rho}}{\rho_e} \frac{\bar{u}}{u_e} \left(1 - \frac{\bar{u}}{u_e} \right) \left(1 - \frac{\bar{y}}{R} \right) d \left(\frac{\bar{y}}{R} \right)} \quad (3)$$

Using the ideal gas law, a density profile was determined from the temperature profile obtained with the Whitfield-High approximation, $\bar{T}/T_e = (\bar{\rho}/\rho_e)^{-1}$. The wall-wake fit and the density profile was used to evaluate the above integrals.

The incoming boundary layer mean velocity profile was also computed from the normal hot-wire measurements by assuming a constant static pressure through the boundary layer, using the wall static pressure. In this case, the mean mass flux and total temperature were measured directly and there was no need to assume a specific temperature profile.

Mean static pressure and temperature were used to obtain a mean density profile from the ideal gas law. The mean density profile was then used to calculate the mean velocity profile from the measured values of mean mass flux. The wall-wake model was once again used as a fit to the data and obtain values of δ , C_f , and Π . The resulting mean velocity profile, and the corresponding experimental measurements, are shown in Figure 12a. The mean velocity profiles in terms of van Driest scaled variables are shown in Figure 12b. The mean velocity profile determined from the mass flux measurements is also in excellent agreement with the law-of-the-wake behavior. The maximum uncertainty in the mean velocity profile based on hot-wire measurements is approximately 2%. Details of the uncertainty analysis are presented in Ref [12].

The mean velocity profile wall-wake least-squares fits obtained with Pitot and hot-wire measurements are shown in Figure 13. The Pitot profile used the Whitfield-High approximation to calculate the temperature profile. The hot-wire profile agreed with the velocity profile based on Pitot measurements within 0.15%. The calculated boundary layer properties were well within their uncertainty bounds. These results further validate the choice of calibrating the hot-wire at a single total temperature.

The PIV technique directly measured the flow velocity and thus only required a simple ensemble average to produce a mean profile. As seen in Figure 13, while the mean velocity profiles for the two probe techniques are in close agreement, the PIV measurement shows a distinct reduction across the entire boundary layer. This is unlikely to be solely an artifact of the difference in sampling station and will be a subject of future investigation in the detailed PIV analysis.

C. Normal Stresses

For the CVA system, the output voltage of the system is related to the fluctuating parameters of the incident flow, such as velocity, temperature, or density. For small fluctuations, sensitivity coefficients specify the change in wire output per unit change in each flow parameter. As part of the wire calibration, these coefficients were obtained by taking the derivative of the heat transfer function with respect to each of the flow parameters, and evaluating the resulting function at the mean operating condition. The output of the wire was then specified as part of these sensitivities and the fluctuating flow parameters. The mean square of this equation is shown below.

$$\frac{\overline{e'^2}}{\overline{E}^2} = S_{\rho u}^2 \frac{\overline{(\rho u)'^2}}{(\overline{\rho u})^2} + S_{T_T}^2 \frac{\overline{T_T'^2}}{\overline{T_T}^2} + 2S_{\rho u}S_{T_T} \frac{\overline{(\rho u)' T_T'}}{\overline{\rho u} \overline{T_T}} \quad (4)$$

Equation 4 represents the modal analysis of the hot-wire output voltage. The left-hand-side of Equation 4 is known in terms of the CVA outputs of mean and fluctuating voltages, and the overheat of the wire. The three unknowns on the right-hand-side of Equation 4 are the normalized fluctuations of mass flux $\left(\overline{(\rho u)'^2}/(\overline{\rho u})^2\right)$ and total temperature $\left(\overline{T_T'^2}/\overline{T_T}^2\right)$, as well as the normalized correlation between these quantities $\left(\overline{(\rho u)' T_T'}/\overline{\rho u} \overline{T_T}\right)$. The sensitivities, $S_{\rho u}$ and S_{T_T} were determined from the hot-wire calibration, and were dependent on overheat parameter. These three quantities are the turbulent intensities. In principle, the three unknown turbulent intensities can be solved for by collecting data at just three different overheat parameters. The accuracy of the results is significantly improved by collecting data at a larger number of overheat parameters and using a least-square method to solve for these turbulence intensities [16].

Figure 14 shows the turbulent intensities measured in the incoming boundary layer. The intensities are normalized by their corresponding local value. The turbulent intensities are larger near the wall, and are reduced to a small positive number near zero in the freestream. Also shown is the difference in results using only three overheat parameters versus using 11. Using fewer overheat parameters tends to overestimate the turbulent intensities, and small uncertainties in measurements results in large uncertainties in flow variables. This is due to the difficulties in inverting the 3x3 coefficient matrix. Data was gathered at 11 different overheat parameters during this investigation ranging from 0.05 to 1.0. The approximate uncertainties in the turbulent intensity are 6.5% and 8.3% for the mass flux and total temperature turbulent uncertainty, respectively. The approximate uncertainty in the correlation between these two is approximately 10.3%.

The turbulent intensities were used to calculate the turbulent mass flux term, $\overline{\rho' u'}$, and correct the velocity profile calculated from the mean mass flux measurements

$$\overline{\rho' u'} = -\frac{\overline{\rho}}{\overline{T}} \overline{u' T'} \quad (5)$$

where

$$\overline{u' T'} = \frac{\overline{u} \overline{T}}{(\alpha + \beta)^2} \left[\frac{\overline{T_T'^2}}{\overline{T_T}^2} - \alpha \beta \frac{\overline{(\rho u)'^2}}{(\overline{\rho u})^2} + (\alpha - \beta) \frac{\overline{\rho' u' T_T'}}{\overline{\rho u} \overline{T_T}} \right] \quad (6)$$

and

$$\alpha = \frac{1}{1 + \frac{\gamma - 1}{2} \overline{M}^2}, \quad \beta = \alpha(\gamma - 1) \overline{M}^2.$$

The corrected mean velocity profile was used to recalculate the $\overline{\rho' u'}$ term, and the process was iterated until $\overline{\rho' u'}$ converges within 10^{-5} . The turbulent strength, or the root-mean-square velocity, u_{rms} , in the incoming boundary layer was calculated using the mean velocity, density, Mach number, and temperature previously calculated from the hot-wire data as well as the turbulent intensities.

$$u_{\text{rms}} = \sqrt{\overline{u'^2}} \quad (7)$$

Where

$$\overline{u'^2} = \frac{\overline{u}^2}{(\alpha + \beta)^2} \left[\frac{\overline{T_T'^2}}{\overline{T_T}^2} + \alpha^2 \frac{\overline{(\rho u)'^2}}{(\overline{\rho u})^2} + 2\alpha \frac{\overline{\rho' u' T_T'}}{\overline{\rho u} \overline{T_T}} \right] \quad (8)$$

The turbulent intensity (rms velocity normalized by the local mean velocity) across the boundary layer is shown in Figure 15. As expected, the streamwise velocity rms has a maximum value near the wall and decreases close to zero in the freestream. Klebanoff [17] showed that in the viscous sublayer this relative intensity reaches a constant value, and the velocities vary linearly with distance from the wall. When extrapolated to the wall, the curve for $\sqrt{u'^2}/\bar{u}$ seems to suggest a limiting value smaller than 0.4, consistent with the value obtained by Klebanoff and Morkovin [17]. The uncertainty in the root-mean-square velocity is approximately 7.1%.

For this investigation, only data in the streamwise direction was collected. The only normal stress that can be calculated with a single normal hot-wire is the streamwise normal stress, defined by the following.

$$\tau_{xx} = \bar{\rho} \overline{u'^2} + 2\bar{u} \overline{\rho' u'} \quad (9)$$

The resulting profile normalized by the wall shear is shown in Figure 16. The wall shear stress is calculated by the following equation

$$\tau_w = 1/2 \rho_\infty u_\infty^2 C_f \quad (10)$$

where C_f is the skin friction coefficient calculated from the wall-wake fit to the hot-wire data. The values of the normal stress go to zero in the freestream, and have a maximum value near $2\delta_0/3$. This is consistent with results found in references [7, 17]. The uncertainty in the streamwise normal stress is approximately 11%.

IV. Shock Wave Boundary Layer Interaction Results

A. Oil Visualization

An initial series of experiments were conducted in an effort to provide qualitative flow visualization information about the conical shock wave turbulent boundary layer interaction. Surface flow tracing visualization was applied in order to resolve the surface flow patterns in the SWBLI region about the various cones. Three different half-cone angles were used at ℓ_{OF1} : 10° , 13.5° , and 16° . Each shock generator was rotated into the 0° , 90° and 180° circumferential position (covering approximately 105 degrees of the test section) to visualize the flow. As mentioned previously, a blank window was used to perform the flow visualization experiments. The blank window covers approximately 35 degrees of the test section.

The flow visualization technique used in this experiment is similar to the kerosene/ lampblack technique typically applied in blow-down supersonic wind tunnels where rapid starting is possible. In this method, a test surface is coated with a fine, dark powder, typically lampblack, in a suspension of kerosene. Once the tunnel is started, the kerosene/ lampblack mixture flows into a pattern approximating the wall shear flowfield. The kerosene is quickly evaporated, freezing the resulting pattern of dried lampblack, which is then lifted off the test surface.

Since the tunnel used in this investigation was not a blow-down tunnel, slight modifications to the previously described technique were necessary. The blank window was first coated with a thin layer of penetrating oil (WD-40), and then a mixture of oil and kerosene with fluorescent powder dye in suspension was applied to the blank window. Once the kerosene was evaporated from the mixture, a very thin layer of oils and fluorescent dye traced the wall shear flowfield. The blank window was then photographed under a blacklight, causing the fluorescent dye to glow. Due to the penetrating oil, it was not possible to transfer the pattern into a sheet of paper as is typical of this technique. The penetrating oil was found to be a critical step in this process to achieve the necessary surface tension properties on the blank window. This is typically not necessary as most wind tunnel surfaces are made out of stainless steel and not aluminum as in this SWT.

The resulting images of the shear flowfield on the test section surface are shown in Figures 17 through 19. In each image, the inviscid shock trace is shown as a solid black line. When the shock generator is aligned on the centerline, the shock trace is given by

$$x_S = x_{tip} + \frac{R}{\tan \beta} \quad (11)$$

where β is the shock angle, and x_{tip} is the x -coordinate of the cone tip. When the shock generator is offset from the centerline, the shock trace is a function of the offset and the circumferential location along the test section walls. The resulting shock trace equation is

$$x_S = x_{\text{tip}} + \frac{\sqrt{(\ell - R \cos \theta)^2 + (R \sin \theta)^2}}{\tan \beta} \quad (12)$$

which reduces to Equation 11 when the offset, ℓ , equals to zero (i.e. the shock generator is aligned with the test section centerline).

Figure 17 shows the wall shear flowfield for the 10° cone. For $\theta = 0^\circ$ and $\theta = 180^\circ$ circumferential locations, a dashed line indicates the upstream influence of the shock impingement, but no flow separation is present. This dashed line follows the same trace as the inviscid shock impingement as shown by the solid line. The upstream influence length is $4.7\delta_0^*$ at $\theta = 0^\circ$ and $5.17\delta_0^*$ at $\theta = 180^\circ$. There is a faint line downstream of the initial interaction for $\theta = 0^\circ$ corresponding to the downstream interaction due to the reflected shock wave off of the centerbody of the shock generator. At $\theta = 90^\circ$, the flow indicates a small degree of separation leeward of the centerline indicated by a small pooling of oil. The resulting oil streamlines show how the flow turns near the separation, making it difficult to survey the flow with invasive instrumentation at this circumferential location. The flow continues to turn downstream of the separation into a second separation region at the downstream interaction.

The wall shear flow for the 13.5° cone is shown in Figure 18. Clear separation is indicated at all three circumferential positions. At all three circumferential locations, the separation and reattachment lines follow the same trace as the inviscid shock trace. The separation at $\theta = 90^\circ$ is significantly stronger than with the 10° cone. The separation trace still follows the inviscid shock trace. The streamlines indicate sharper flow turning at this location. At $\theta = 180^\circ$ the interaction and separation appear stronger than at $\theta = 0^\circ$. Although difficult to discern due to the quality of the oil flow at this location, the separation line still follows the incident shock trace. The flow recirculates in the region between the separation and reattachment line, and downstream the reattachment point the flow appears to converge towards the centerline and continues in this fashion even downstream of the inviscid shock impingement location.

Figure 19 shows the wall shear flowfield for the 16° cone. Once again, all three circumferential locations show high degree of separation. At $\theta = 0^\circ$, the flow diverges from the centerline as it moves downstream. Away from the centerline immediately downstream the separation line, the flow appears to immediately turn almost 90 degrees. This behavior is also seen on the 13.5° cone, but the turning angle is significantly sharper on the 16° cone. Except for the area near the centerline, the separation line follows the inviscid shock trace. At $\theta = 90^\circ$, the flow has a sharper turn into the separation line compared to the previous shock generators. At $\theta = 180^\circ$, the reattachment point appears as a node on the centerline. The flow appears to converge into this point downstream the separation line. There are no converging streamlines moving downstream unlike what was observed for the 13.5° cone at this circumferential location. The circumferential locations of $\theta = 0^\circ$ and $\theta = 180^\circ$ were surveyed with a hot-wire probe for the 16° shock generator on ℓ_{OF1} . The exact locations are shown with red dots in Figure 19.

No instance of surface oil visualization for an interaction of constant strength in a asymmetric configuration have been found in the literature. The patterns shown in Figures 17-19 are symmetric about the window centerline for the $\theta = 0^\circ$ and $\theta = 180^\circ$ cases for all three shock generators. Significant flow turning is seen for the $\theta = 90^\circ$ case. It appears the flow starts to sharply turn between $\theta = 15^\circ$ and $\theta = 75^\circ$ and continues to turn between $\theta = 105^\circ$ and $\theta = 165^\circ$, with the largest turn angle at $\theta = 90^\circ$. This is consistent with the results of Chou [11], where the largest flow angle was found near the wall at $\theta = 90^\circ$.

B. Pressure Surveys

The mean wall pressure measurements of the shock generators aligned on the test section centerline are shown in Figure 20 for cone half-angles of 10° , 13.5° , and 16° . The profiles for the 10° , 13.5° , and 16° cone represent an unseparated, an incipiently separated, and a separated SWBLI, respectively.

For all profiles, the pressure begins to increase upstream of where the incident shock would have impacted the wall in the absence of a boundary layer. The pressure rise is a result of the incident shock wave that is transmitted upstream through the subsonic part of the boundary layer. This leads to a spreading of the wall pressure distribution over a distance on the order of approximately five displacement thicknesses. This distance is referred to as the upstream influence length, L_0 . There is a trend in the location of the initial pressure rise with cone angle. From Figure 20, it can be seen that the initial pressure rise is further upstream with increasing cone angle (i.e. shock strength). The upstream influence length for the $\alpha = 16^\circ$ shock generator is almost double that of the $\alpha = 10^\circ$ shock generator.

The wall pressure for all profiles continues to rise downstream of the incident and reflected shock waves due to the adverse pressure gradient imposed by the conical flowfield. The pressure continues to rise until the flow is exposed to the favorable pressure gradient created by the expansion fan emanating from the cone shoulder. The pressure begins to

decrease upstream of where the expansion fan impinges the wall in the absence of the boundary layer. The distance between inviscid shock wave and expansion fan impingement is referred to as the compression length in this investigation, and it is shown schematically in Figure 2, denoted as L_C . For $\alpha = 10^\circ$, L_C is large enough that the favorable pressure gradient has no influence in the immediate vicinity of the SWBLI. For $\alpha = 13.5^\circ$ and $\alpha = 16^\circ$, the expansion fan interacts with the flow near the interaction region. The gradual pressure increase through L_C is more pronounced for $\alpha = 10^\circ$. The expansion fan emanating from the cone shoulder for $\alpha = 10^\circ$ impacts the boundary layer further downstream compared to the other shock generators. The normalized levels of static pressure show a clear trend of higher pressures corresponding to larger cone angles. The maximum normalized pressure levels of 1.96, 2.26, and 2.30, are found for cone half-angles 10° , 13.5° , and 16° , respectively.

The profiles for the 13.5° and the 16° cone show an inflection point near $X = 6$ indicative of flow separation. This inflection point is difficult to discern due to the adverse pressure gradient generated by the conical flowfield and the effects of the expansion fan. Separation is difficult to predict solely with static pressure profiles in a conical flowfield for a SWBLI for low to moderate shock strengths.

The circumferential pressure field around the test section wall was also measured for the shock generators at distinct offsets from the test section centerline. The resultant distributions for ℓ_{OF_2} are presented in Figures 21 through 23. The wall static tap locations, as well as the incident inviscid shock trace are included for reference. In each figure, the contour levels shown are the same to allow for direct comparison of the extent of the pressure levels in the interaction over shock strength and offset distance.

Similar to the centerline results, the magnitude of the pressure distributions increases with increasing cone angle. The intensity of the pressure rise can be seen to also vary with circumferential angle and offset position. Also compared to the centerline results, the onset of the interaction is moved upstream at $\theta = 0^\circ$ (shock generator is closest to the wall), and downstream at $\theta = 180^\circ$ as the shock generator moves farther away from the test section wall. The dashed vertical line represents the measured location of the initial pressure rise for the corresponding shock generator on the centerline. In all profiles, the incident shock impingement based on an inviscid solution is also shown. The shock location on the wall moves further downstream with increasing circumferential angle. It is important to note that the peak pressure does not occur at the shock impingement point, but rather at the expansion fan impingement point due to the compression region created by the conical flowfield. While this conical flowfield makes it difficult to discern flow separation, an approximate separation line is plotted for the 13.5° and 16° cases based on the perceived inflection points in the pressure distributions. This approximate separation line follows the a similar curve as the shock trace.

In all cases, the maximum pressure occurs at the farthest upstream impingement location, $\theta = 0^\circ$. At this location the shock generator is closest to the wall. The maximum pressure is larger compared to the distributions of the shock generators on the centerline. Normalized pressure levels of 1.96, 2.38, 2.53 for cone half angles 10° , 13.5° , and 16° at ℓ_{OF_1} and 1.99, 2.46, 2.65 respectively, at ℓ_{OF_2} . At this location, the distance between the shock impingement and expansion fan impingement, the compression length L_C , is the largest. The greater compression length allows the flow to reach a higher pressure before expanding again.

Sample Pitot pressure profiles measured in the conical SWBLI regions are presented in Figures 24 and 25. Three distinct samples are highlighted in these two figures. Sample A shows a typical profile upstream of the interaction through the incident shock wave. Sample A crosses the incident shock wave but not the reflected shock. Sample B is a typical profile just downstream of the impingement point of an unseparated interaction. Sample C in Figure 25 highlights a typical profile downstream of the inviscid impingement point for a separated interaction. Samples B and C both cross a reflected shock wave and the resulting expansion fan off the cone shoulder. For all three profiles, a sample measured Pitot profile is shown as well as an inviscid Pitot profile calculated by the method of characteristics. A Mach number profile is also shown for each sample. This Mach number is solely based on the inviscid solution. Mach number profiles cannot be determined based on the measured Pitot data alone. Static pressure is also required to determine Mach number and the only data point available is the static wall pressure.

C. Normal Hot-Wire Results

1. Mean Mass Flux

The resulting mean mass fluxes are presented in Figures 26 and 27. The mean mass fluxes are non-dimensionalized by the incoming freestream mass flux, $\bar{\rho}u/(\bar{\rho}u)_0$. Also shown in the figures as a solid blue line at each station is the incoming mean mass flux profile. The profile shown is from the wall-wake velocity fit using the hot-wire data gathered at the incoming boundary layer location shown in Figure 12. The incident and reflected shock waves plotted

are determined from the Pitot profiles.

Figure 26 shows the mean mass flux distributions for $\alpha = 10^\circ$ and $\alpha = 16^\circ$ for the shock generator aligned on the test section centerline. For $\alpha = 10^\circ$, there is no significant difference between the profiles moving downstream through the interaction. There is a slight increase in mass flux in the profiles at the first two stations downstream of the incident shock. The remaining profiles do not show any discernible change in mass flux downstream of the reflected shock wave. The profiles follow the same general trend as the incoming mass flux profile. The mass flux increases from incoming levels as the probe moves downstream, and the greatest increase from the incoming profile is seen at the last station, with an overall mass flux approximately 20% greater than in the incoming boundary layer. The incident shock wave and resulting interaction are not strong enough to cause a significant change in the mass flux profile. As expected, there is a large increase in mass flux as the probe moves downstream of the incident shock wave.

For the 16° shock generator, the first station shown, station 12, is right at the edge of the upstream influence region. The mass flux distribution measured on station 12 is in good agreement with the incoming profile until the probe is downstream of the incident shock wave. There is an increase in mass flux downstream of the incident shock of approximately 30%. The next measuring station, station 13, is at the onset of the detected separation. There is an inflection point in the profile near $\bar{y}/\delta_0 = 0.4$. The inflection in the mass flux profile represents unsteadiness in the flow. There is a slight decrease in mass flux as the probe continues to move in the positive $\bar{y}$, however the mass flux still agrees with the measured incoming mass flux levels within the uncertainty bounds.

For the next two stations, the mass flux increases significantly through the boundary layer downstream of both the incident and reflected shock waves. At station 15, there is a peak mass flux level of approximately 1.8. The mass flux decreases once again when the probe is upstream of the reflected shock wave. Station 15 shows evidence of shock unsteadiness near the edge. The shock wave at this location is constantly moving upstream and downstream of the probe affecting the mean reading.

2. Turbulent Intensities

The mass flux turbulence intensities for the shock generator aligned on the centerline are shown in Figure 28. The weak interaction of the 10° shock generator results in very little change in the turbulent intensity profiles compared to the incoming levels. The mass flux intensities remain constant through the interaction with the uncertainty bounds of the incoming levels. The downstream profiles shown in the last two stations show an increase in turbulent intensity below $\bar{y}/\delta_0 = 0.6$ of approximately 18%. There is also a 10% decrease in turbulent intensity near the boundary layer edge compared to the incoming levels. The recovering boundary is less turbulent than the incoming boundary layer. The weak unseparated interaction does not significantly amplify the mass flux fluctuations in the boundary layer. Through this interaction, the maximum turbulent intensity is found near the $\bar{y}/\delta_0 = 0.6$, just as in the incoming boundary layer.

The separated interaction created by the 16° cone has a greater impact on the measured turbulent intensities. The intensities on station 12 are nearly identical to the incoming boundary layer levels apart for a small increase as the probe moves downstream the incident shock wave. Station 13, near the onset of separation, shows an increase in turbulent intensity near the wall, and a decrease back to incoming turbulent intensities as the probe moves away from the wall. There is another slight increase at the point of incident shock crossing. The point at which the interaction and incoming turbulent intensities first equal one another corresponds to a location of an inflection point in the mass flux.

The greatest increase in turbulent intensity is seen in station 14, with an increase of approximately 60% compared to incoming turbulence intensity. This increase in intensity is partially due to the unsteady reflected shock wave. Near the edge of the boundary layer the turbulence strength decreases back to incoming values. Moving downstream the increase in strength near the center of the boundary layer remains, although it is only 25% higher than incoming levels. The lower turbulent levels near the wall indicate that high-momentum, low-turbulence fluid has been brought into the region near the wall.

Figure 29 shows the mass flux turbulent intensities for the 16° cone offset from the test section centerline. The profiles shown are for the $\theta = 0^\circ$ and the $\theta = 180^\circ$ circumferential locations. These separated interactions show similar characteristics as the mass flux turbulent intensity profiles for the cone on the centerline upstream, and immediately downstream of $X = 0$. Near the wall, the profiles upstream of $X = 0$ show a greater increase in the levels of turbulence compared to the centerline case. The increase in turbulent intensity for the offset cases from the incoming levels is closer to 80% compared to the 60% measured in the centerline case. There is also a greater degree of relaxation near the edge of the boundary layer for these profiles upstream of $X = 0$. There is also a sudden increase shown near the shock crossing. This is due to the increase in mass flux fluctuations as the shock wave moves upstream and downstream of the probe.

The downstream profiles for $\theta = 0^\circ$ and $\theta = 180^\circ$ are also very similar to the downstream profiles for the centerline case. The intensity near the wall decreases back to incoming levels as the probe moves downstream. The maximum remains near the center of the boundary layer, and the maximum turbulence intensity also decreases moving downstream. The absolute turbulence intensities for $\theta = 180^\circ$ are greater downstream compared to the $\theta = 0^\circ$ and centerline case.

The turbulence intensities can also be used to calculate normal stresses as well as rms velocities. The rms velocity and turbulent mass flux are required to calculate the streamwise normal stress component. The calculations for these terms require knowledge of the mean velocity. For the incoming boundary layer the mean velocity through the boundary layer was calculated from both the Pitot and hot-wire measurements by assuming the static pressure is constant through the boundary layer. This assumption does not hold true for the SWBLI measurements. The static pressure is changing in the boundary layer due to the conical flowfield and the separated interaction. An attempt was made to combine the Pitot pressure measurements with the resulting mean mass flux and total temperatures obtained with the hot-wire to calculate the mean Mach number throughout the interaction. The following equation was used to attempt to solve for the mean Mach number.

$$\overline{\rho u} = \frac{\bar{P}_{Tp}}{R(M) \sqrt{\bar{T}_T}} \sqrt{\frac{\gamma}{R} \bar{M}} \sqrt{1 + \frac{\gamma - 1}{2} \bar{M}^2} \quad (13)$$

Unfortunately the equation could not be solved to calculate mean Mach number. The hot-wire is sensitive to ρu while the Pitot tube is sensitive to ρu^2 . Any small variation between the measurements coupled with additional uncertainties due to probe location results in large changes in the Mach number. These variations in Mach number are shown in Figure ?? by comparing the resulting Mach number profile in the incoming boundary layer from reducing the hot-wire measurements with the mean wall pressure and with the Pitot pressure. The error in the Pitot pressure reduction is as large as 20% with Mach numbers greater than the freestream Mach number. Through a separated SWBLI this error only increased, resulting in unrealistic Mach numbers. Therefore the streamwise normal stresses and rms velocities could not be calculated for the SWBLI from Pitot probe measurements.

D. PIV Results

The PIV data were obtained at several discrete camera positions which provided overlapping fields of view which spanned the region of interest in the streamwise direction. Each data set covers a 45×25 mm region of the flow field. A complete survey of the measurable test section region was made first with no model installed to characterize the baseline test section boundary layer. For these measurements, 1000 unique velocity fields were obtained for each camera position. For SWBLI datasets, ensembles of 1000 image pairs were again collected within and downstream of the shock interaction region, while 500 pairs were taken in regions upstream where the velocity should be identical to the baseline boundary layer measurements. While a forthcoming report will delve into detailed analysis of the PIV data, Table 3 provides an overall summary of the breadth of SWBLI cases studied. All PIV datasets were obtained with $\ell_{OF} = R/4$.

Table 3 PIV dataset overview.

	$\alpha = 10^\circ$			$\alpha = 13.5^\circ$			$\alpha = 16^\circ$		
$Re \rightarrow$ $\theta \downarrow$	2×10^6	3×10^6	4×10^6	2×10^6	3×10^6	4×10^6	2×10^6	3×10^6	4×10^6
$\mathcal{C}$	x		x	x		x	x	x	x
0°	x		x	x		x	x	x	x
90°	x		x	x		x	x	x	x
180°	x		x	x		x	x	x	x

All PIV data on offset shock generator is at $\ell_{OF} = R/4$.

In addition to providing an independent redundant velocity measurement, the PIV data was of particular interest for the $\theta = 90^\circ$ position due to the difficulty in making a probe measurement in that interaction [12]. In this highly swept interaction, the flow direction turns rapidly near the wall requiring specialized probe geometry tailored specifically for this application. However, because a stereoscopic PIV measurement was made, all three velocity components of the flow can be quantified in this region. Figure 31 shows the mean field of the 3 velocity components in the impingement

region for the $\alpha = 16^\circ$ case at $\theta = 90^\circ$. The structure of the streamwise and wall-normal velocity component, U and V , are typical of any impinging SWBLI. Figure 31c shows the aforementioned spanwise flow which occurs within the interaction in this case. Here the peak W value is greater than 25% of the freestream U value. The region of $|W| > 100$ m/s extends for approximately $3\delta_0$ in the streamwise direction and up to $0.3\delta_0$ from the wall.

The full PIV dataset will be analyzed in detail and reported in a future publication. The results presented therein will contain the 3 mean velocity components as well as statistics for the velocity fluctuations and all 6 turbulent stress components. The intent will also be to make detailed comparison with the probe measurements previously obtained in Ref. 12.

V. Conclusion

Complete definition of the upstream conditions are vital for CFD computations. The flow structure upstream of the SWBLI has a significant impact on the solution, especially in cases with flow separation. Therefore any benchmark quality validation datasets must include careful characterization of the incoming flow. The 17 cm circular test section in the NASA GRC facility was characterized in detail using the same experimental techniques previously listed. The incoming total temperature profile was measured with the normal hot-wire probe. The profile was found to have a region of total temperature greater than the freestream plenum total temperature. The overshoot in mean total temperature, $\overline{T}_T$, is expected in adiabatic compressible flows. The region total temperature greater than the freestream ranges from $\overline{y}/\delta_0 = 0.38$ to $\overline{y}/\delta_0 = 1.1$, with a maximum total temperature overshoot of 0.55% greater than the freestream value at $\overline{y}/\delta_0 = 0.63$.

The temperature profile is required to reduce Pitot measurements. No significant effect was found by reducing the Pitot measurements using a profile that does not account for the total temperature overshoot such as the Crocco-Busemann approximation. The maximum difference between velocity profiles reduced with and without the temperature overshoot was 0.57%. This results in a difference of 6% in the momentum thickness and 8.2% in the shape factor.

The incoming velocity profiles determined by Pitot and hot-wire measurements had a maximum deviation of 0.5%, further validating the various assumptions made with the hot-wire calibration. Particle image velocimetry measurements show a significant increase in wall friction, which may be due to difference in surface conditions between the two test sections used. These differences will be analyzed in detail in a future detailed analysis of the PIV data.

Static wall pressure measurements confirm the observations on the extent of the SWBLI found from the surface visualization experiments. For the centerline cases, the initial increase in the static pressure is seen to occur further upstream as the interaction strength increases with increasing cone angle. The upstream influence is $4.4\delta_0^*$, $6.1\delta_0^*$, and $8.1\delta_0^*$ for the 10° , 13.5° , and 16° cones, respectively. The static pressure continues to rise downstream of the incident shock due to the conical flowfield created by the shock generators. The maximum pressure, normalized by the incoming static pressure, increases with increasing cone angle. The peak pressure location however decreases with increasing cone angle as the impinging location of the expansion fan emanating from the cone shoulder moves further upstream with higher cone angles.

The asymmetric cases showed a peak pressure varying circumferentially around the test section. The expansion fan impingement varies with circumferential angle. At $\theta = 0^\circ$, the shock generator is closest to the wall, and the expansion fan interacts with the boundary layer further downstream than at $\theta = 180^\circ$. For all 6 asymmetric cases tested, the maximum peak pressure was found at $\theta = 0^\circ$. The inflection point in the wall pressure distribution agreed with the separation location found with surface oil flow visualization. Due to the conical flowfield, a secondary inflection point representing flow reattachment could not be calculated. Static surface pressures alone are not sufficient to predict flow separation and reattachment in conical flowfields.

Pitot pressure measurements were also used to characterize the shock wave turbulent boundary layer interaction. Pitot measurements were also taken in the freestream, outside of the boundary layer. Flow separation was found in the Pitot measurements, shown as constant pressure near the wall. Caution should be used in interpreting these measurements since the bow shock created by the Pitot tube interacts with the wall when the probe is close to it, and affects the flow separation. The profiles for the asymmetric interaction showed a stronger interaction, with a greater change in Pitot pressure across the reflected shock wave, but the flow structure remained the same comparing the centerline cases to the asymmetric cases. The unseparated profiles showed little change due to the SWBLI, and the flow immediately outside the boundary layer was accurately predicted by inviscid methods.

The separated interactions show a significant change in the flow structure through the SWBLI. The readings above the boundary layer showed evidence of a reflected shock system, with areas of compression and expansion. There was also evidence of shock unsteadiness found in the Pitot pressure measurements. In certain locations the discontinuity in

Pitot pressure due to the reflected shock wave showed a reading across it, indicative of an oscillating shock wave. The flow was found to relax back to a Pitot pressure profile similar to the incoming profile downstream of the interaction and the expansion fan.

The turbulence intensities for the unseparated case remained virtually unchanged through the SWBLI. The separated cases show a large increase in turbulence intensity moving through the interaction. The large increase by almost a factor of 2 occurred near the shock crossing. The turbulence intensity reaches a maximum when the reflected shock wave crossed the boundary layer near $\bar{y}/\delta_0 = 0.8$. The intensity remained elevated, but the levels started to attenuate moving downstream from the location of the peak intensity. Downstream of the interaction the turbulence levels remained elevated, with the maximum at $\theta = 180^\circ$. For the centerline case, the turbulence intensity levels downstream of the interaction were nearly identical to the incoming levels, approximately 16% higher near the center of the boundary layer. Apart from slightly higher intensity levels, the turbulence intensity profiles for the asymmetric case had the same trends and similar structure as the centerline case profiles.

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A. Figures

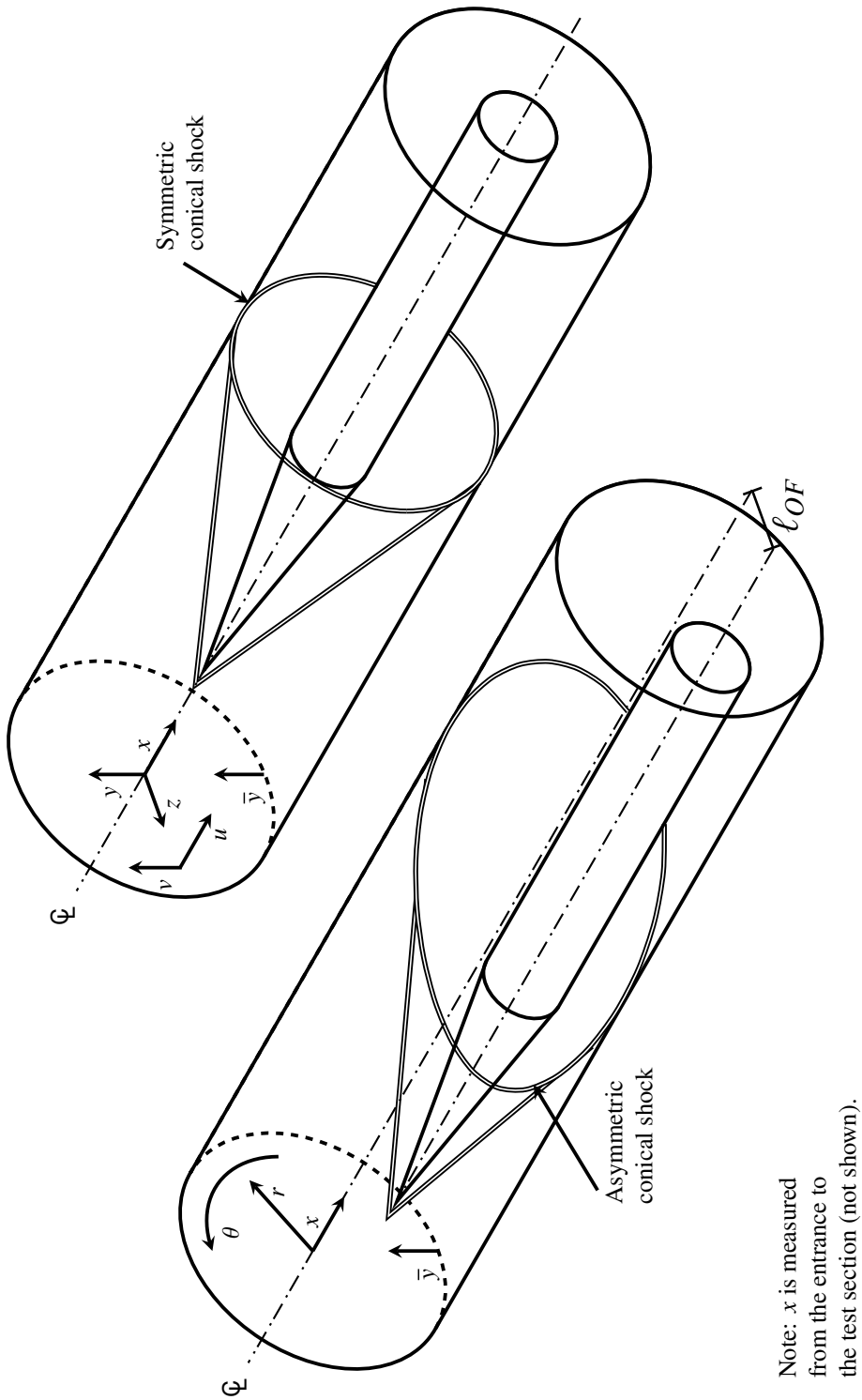


Figure 1 Basic geometry and coordinate system.

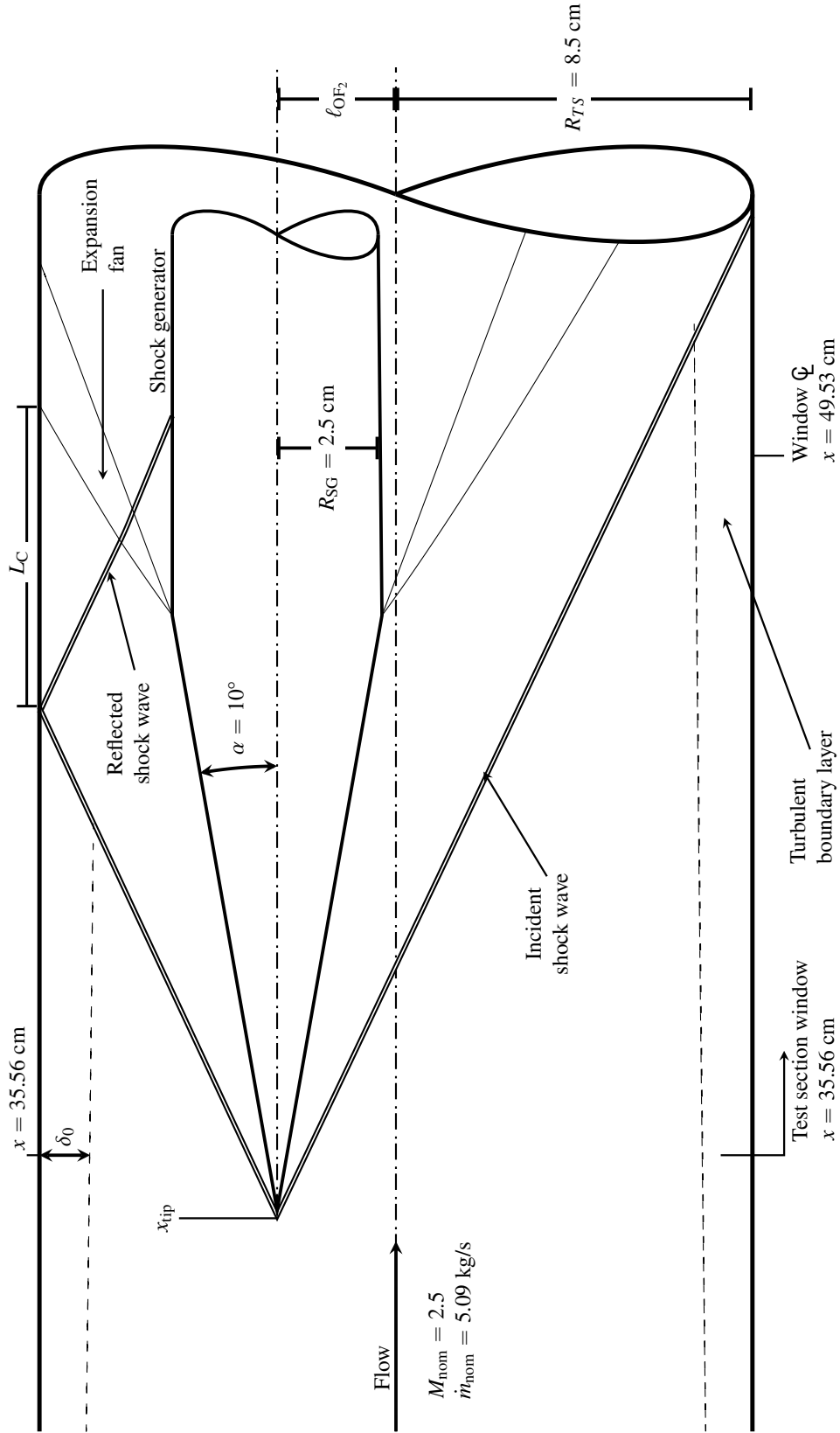


Figure 2 Experimental setup.

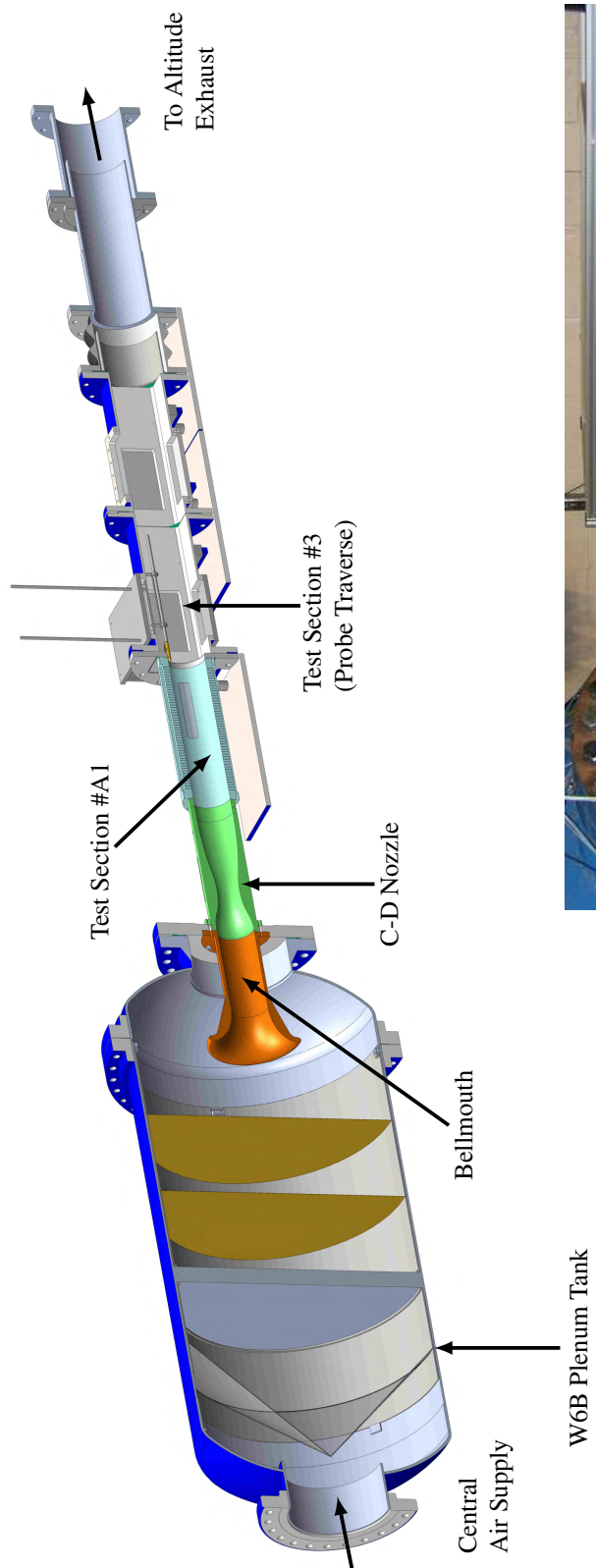


Figure 3 W6B Facility: 17 cm SWT.

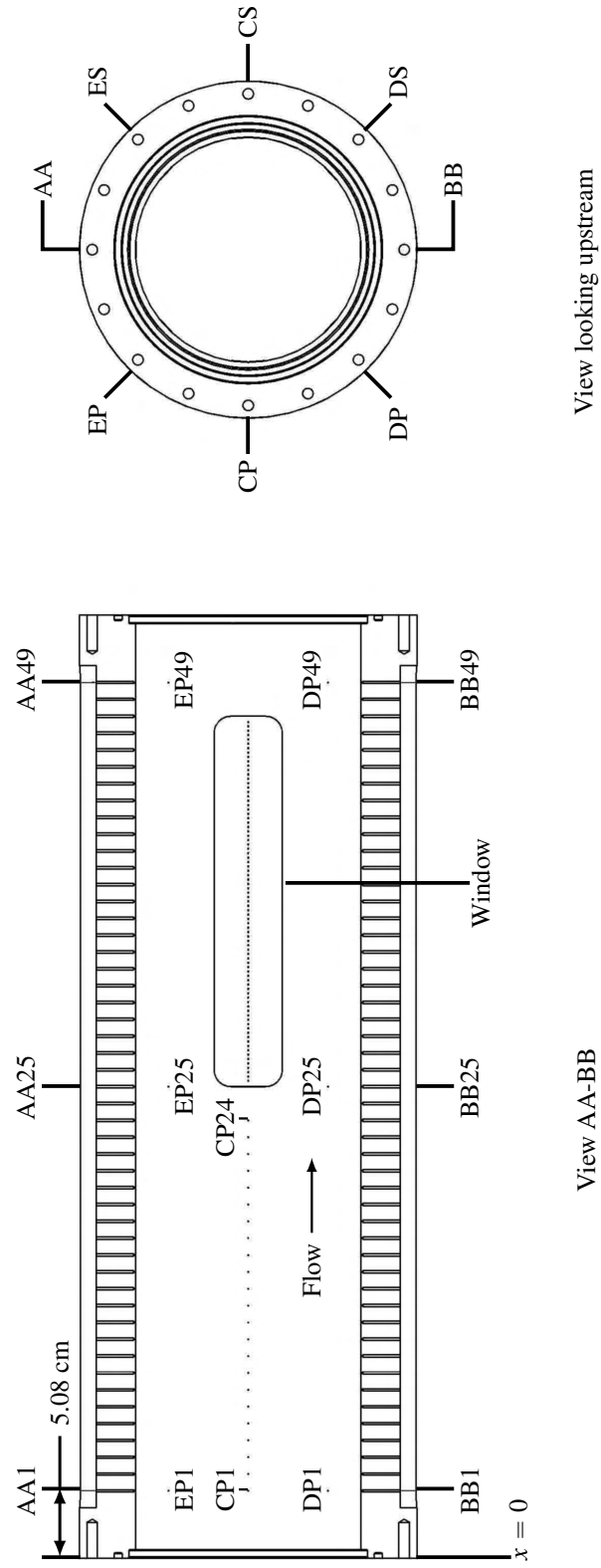


Figure 4 Conventional test section static pressure tap stations.

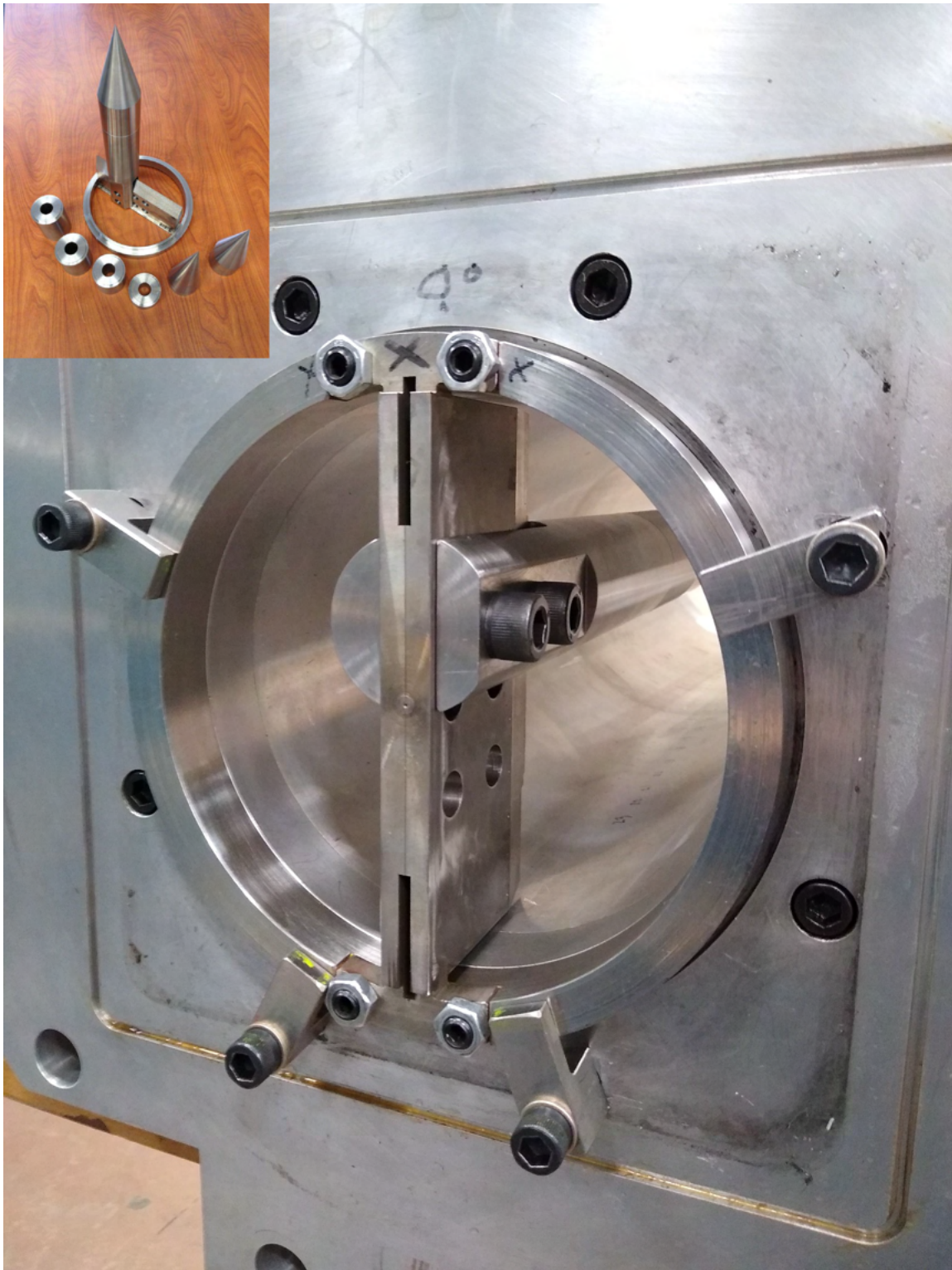


Figure 5 Shock generator support assembly.

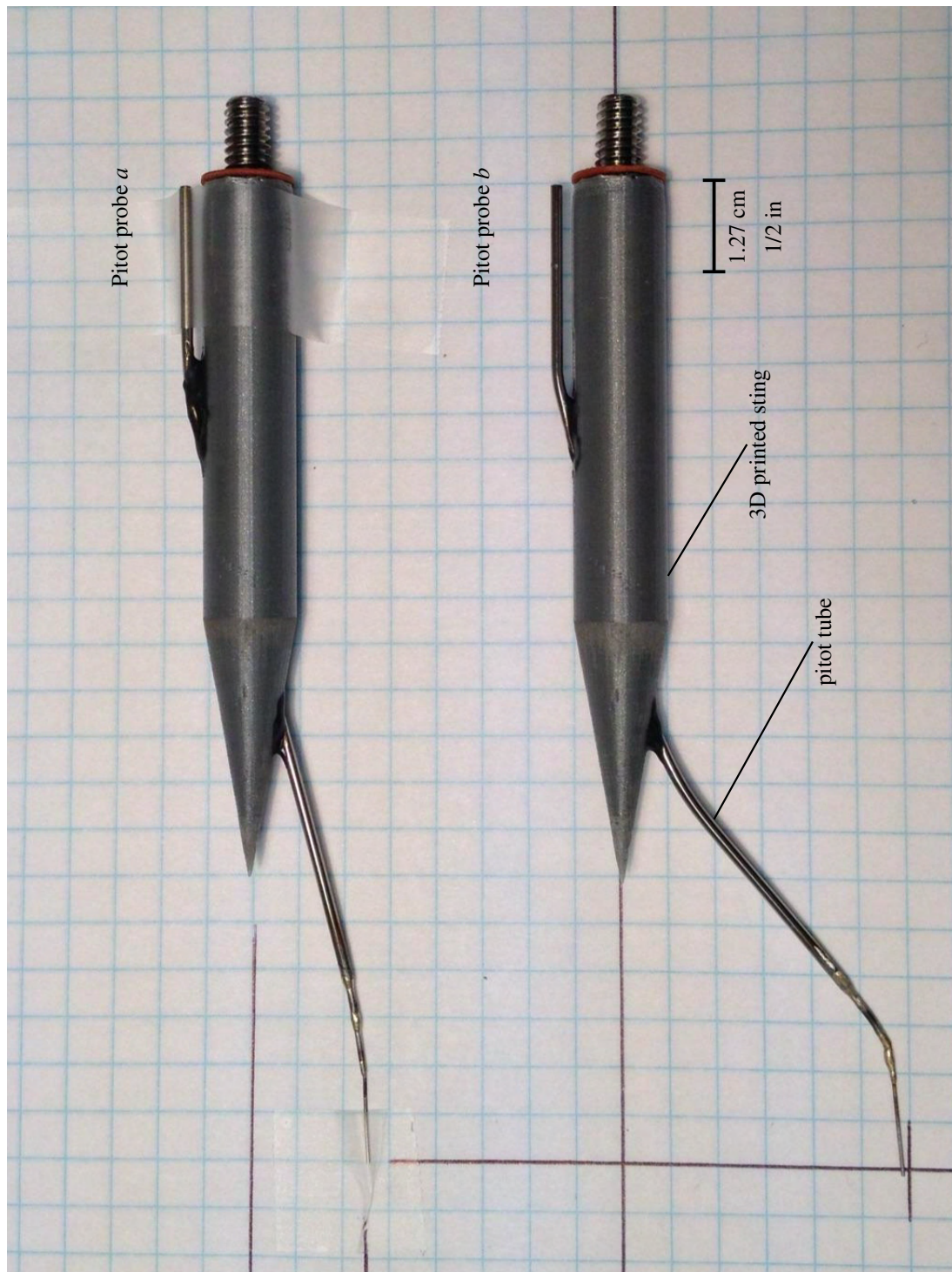


Figure 6 Pitot tube probes.

Note: $\theta = 0^\circ$ is defined as the location where the shock generator is closest to the test section wall.

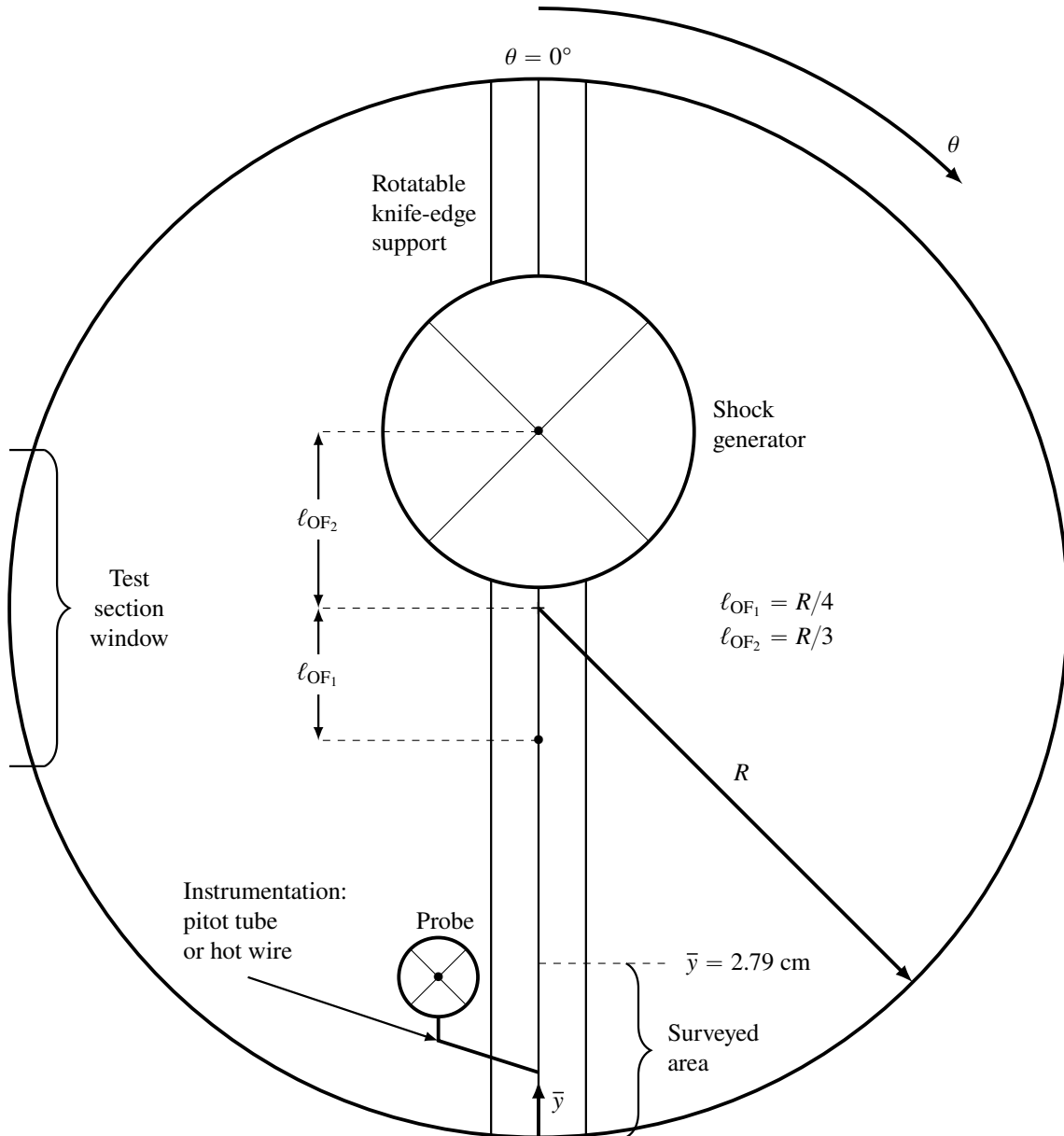


Figure 7 Downstream view of test section.

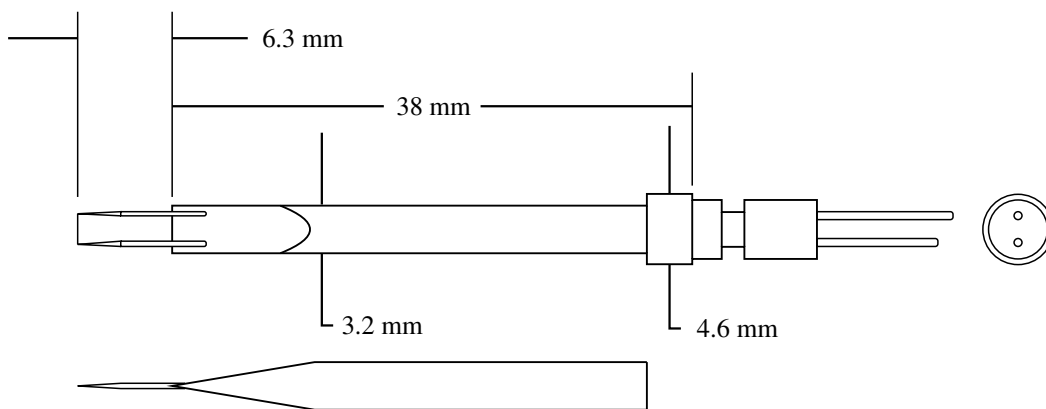


Figure 8 Normal streamlined hot-wire probe

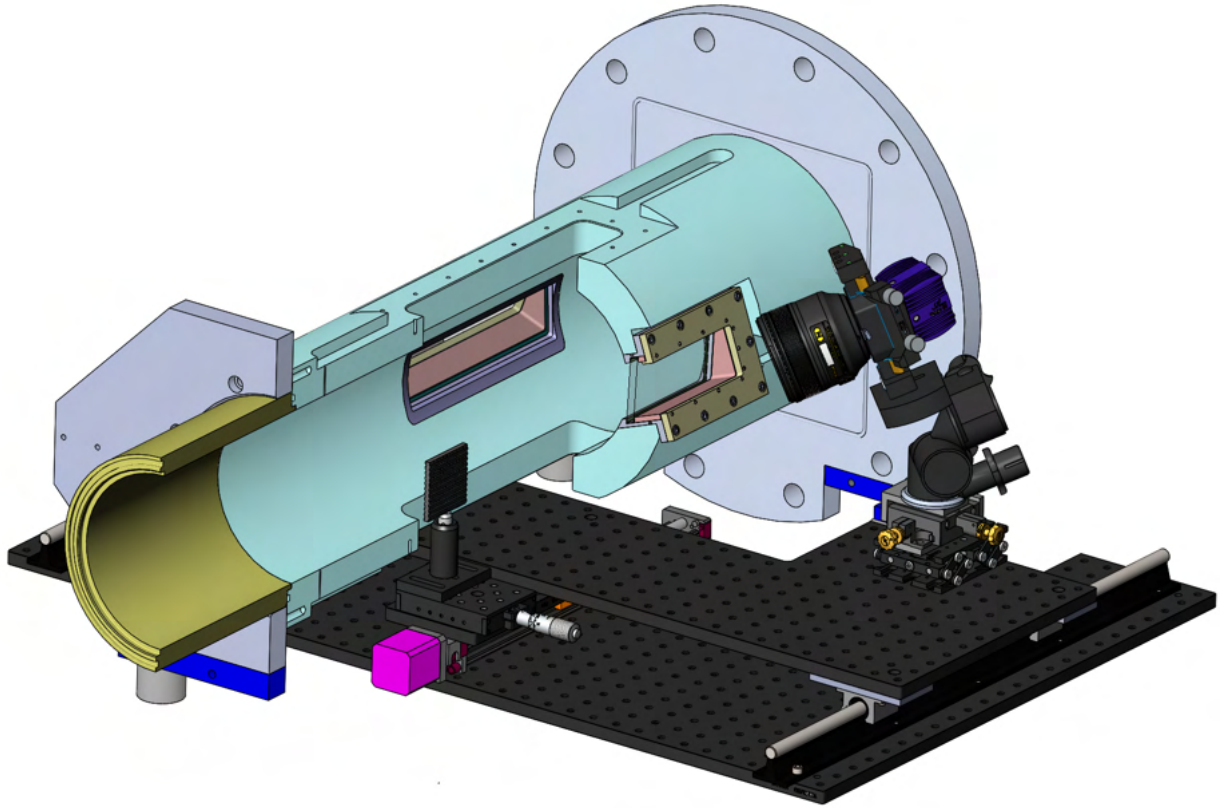


Figure 9 Diagram of the PIV test section assembly. Cutaway shows dot target as inserted into test section for camera calibration and registration.

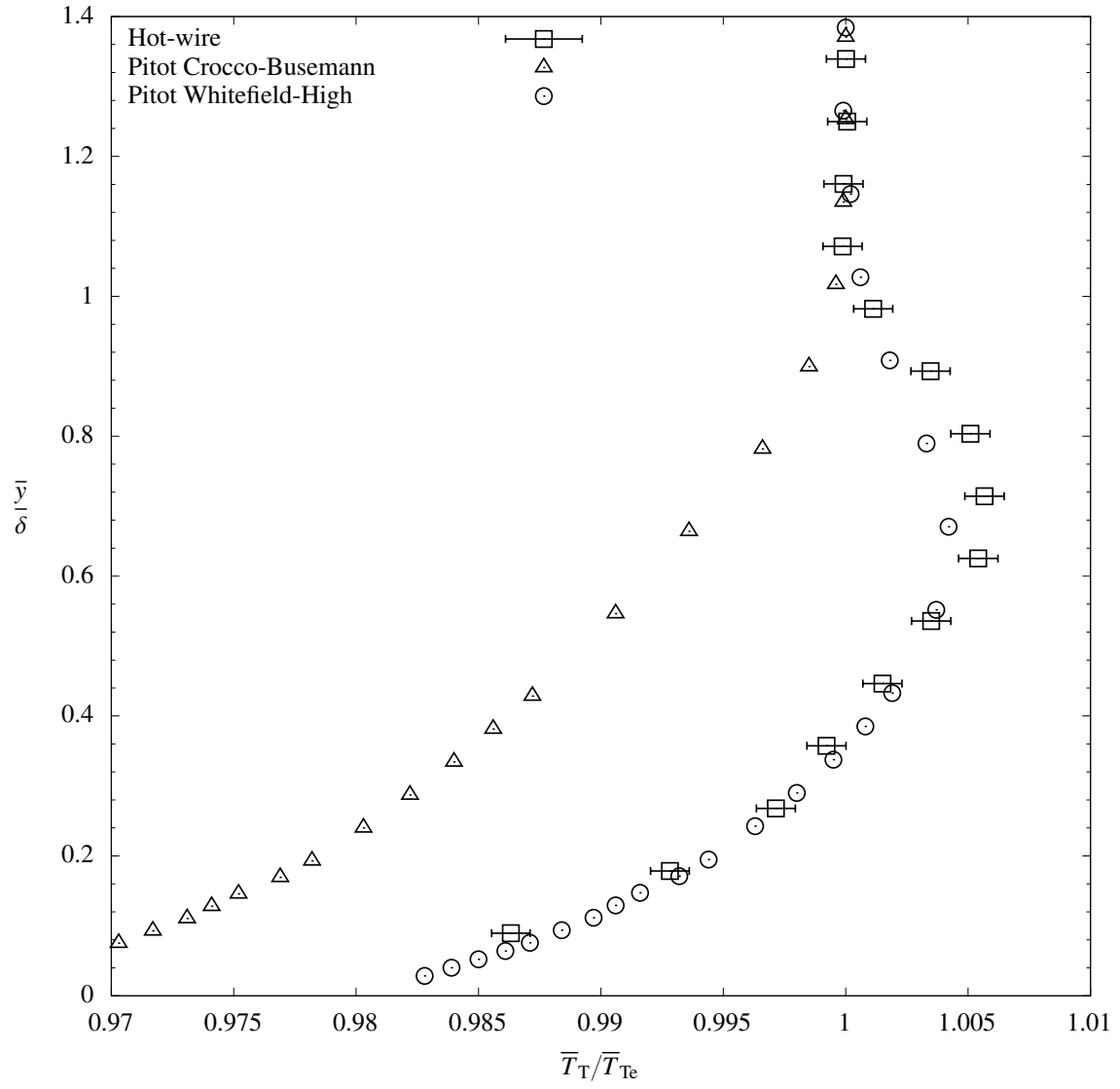


Figure 10 Incoming boundary layer mean total temperature profile.

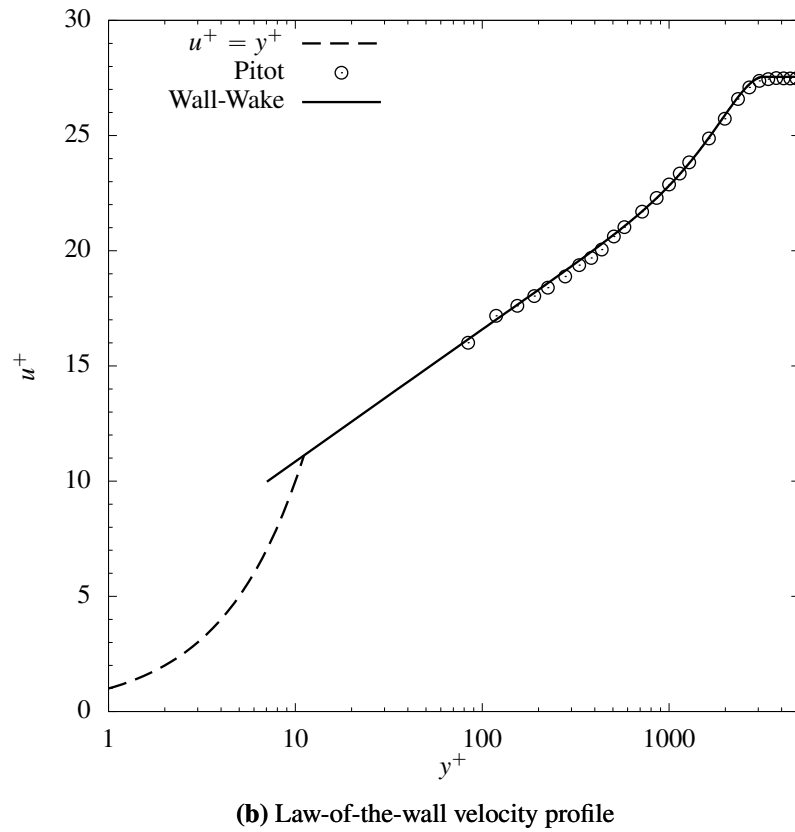
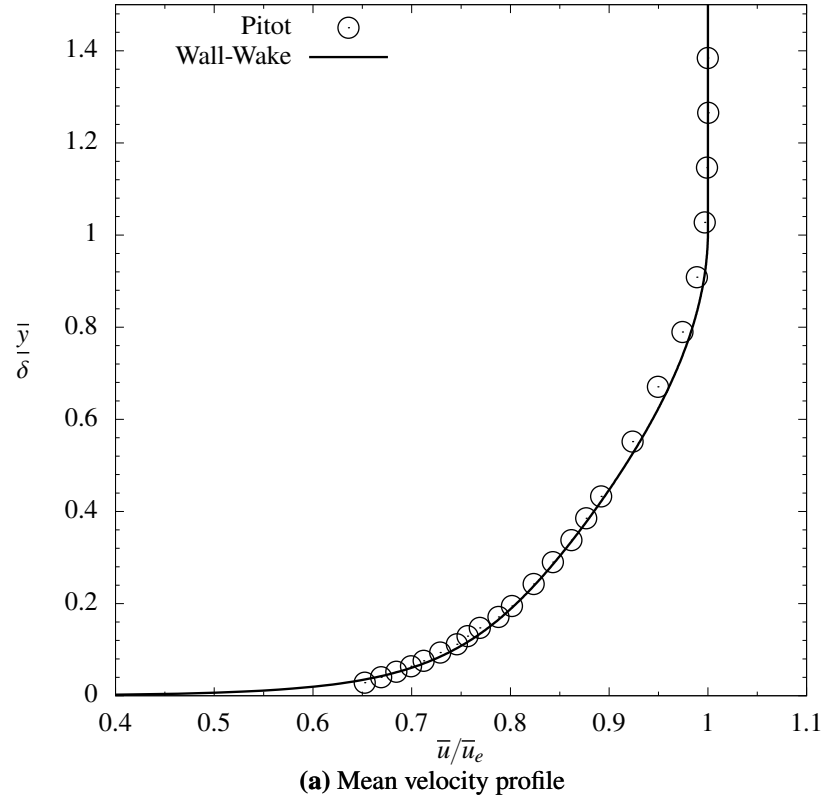


Figure 11 Incoming boundary layer mean velocity profile based on Pitot measurements.

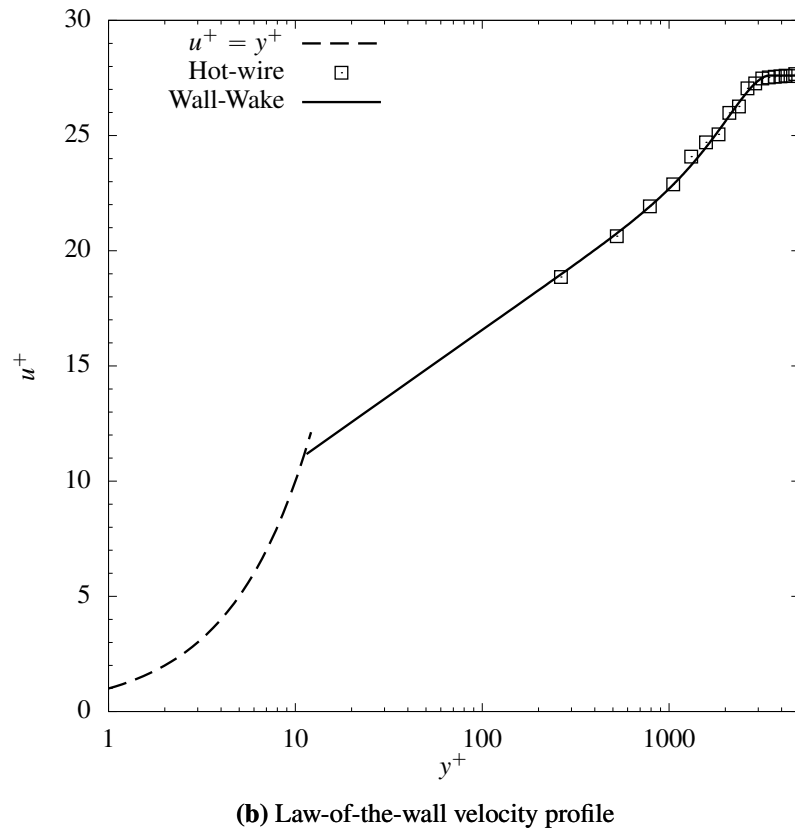
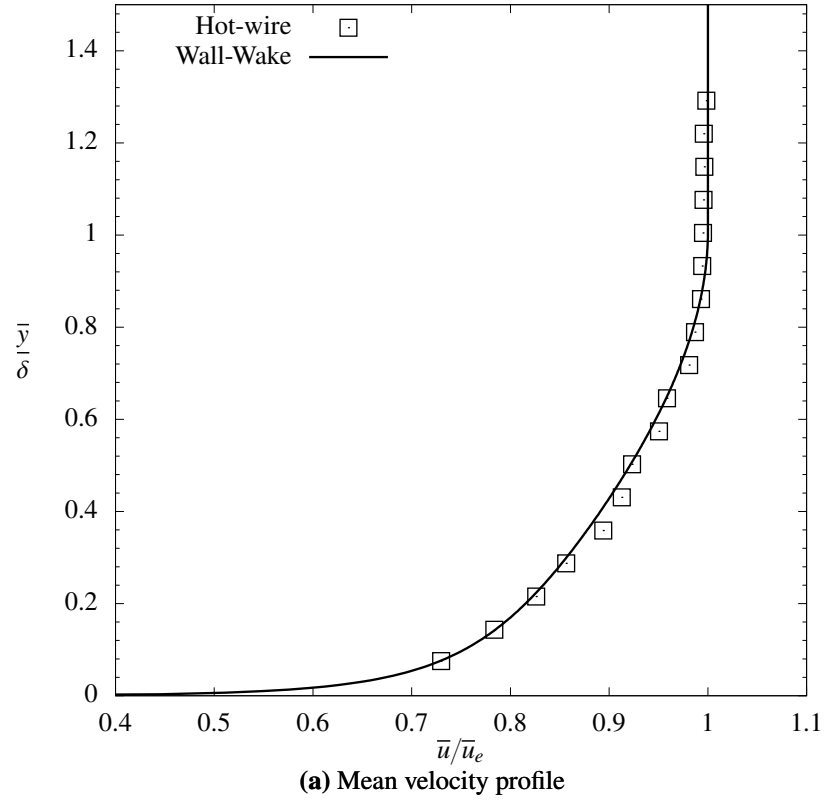


Figure 12 Incoming boundary layer mean velocity profile based on hot-wire measurements.

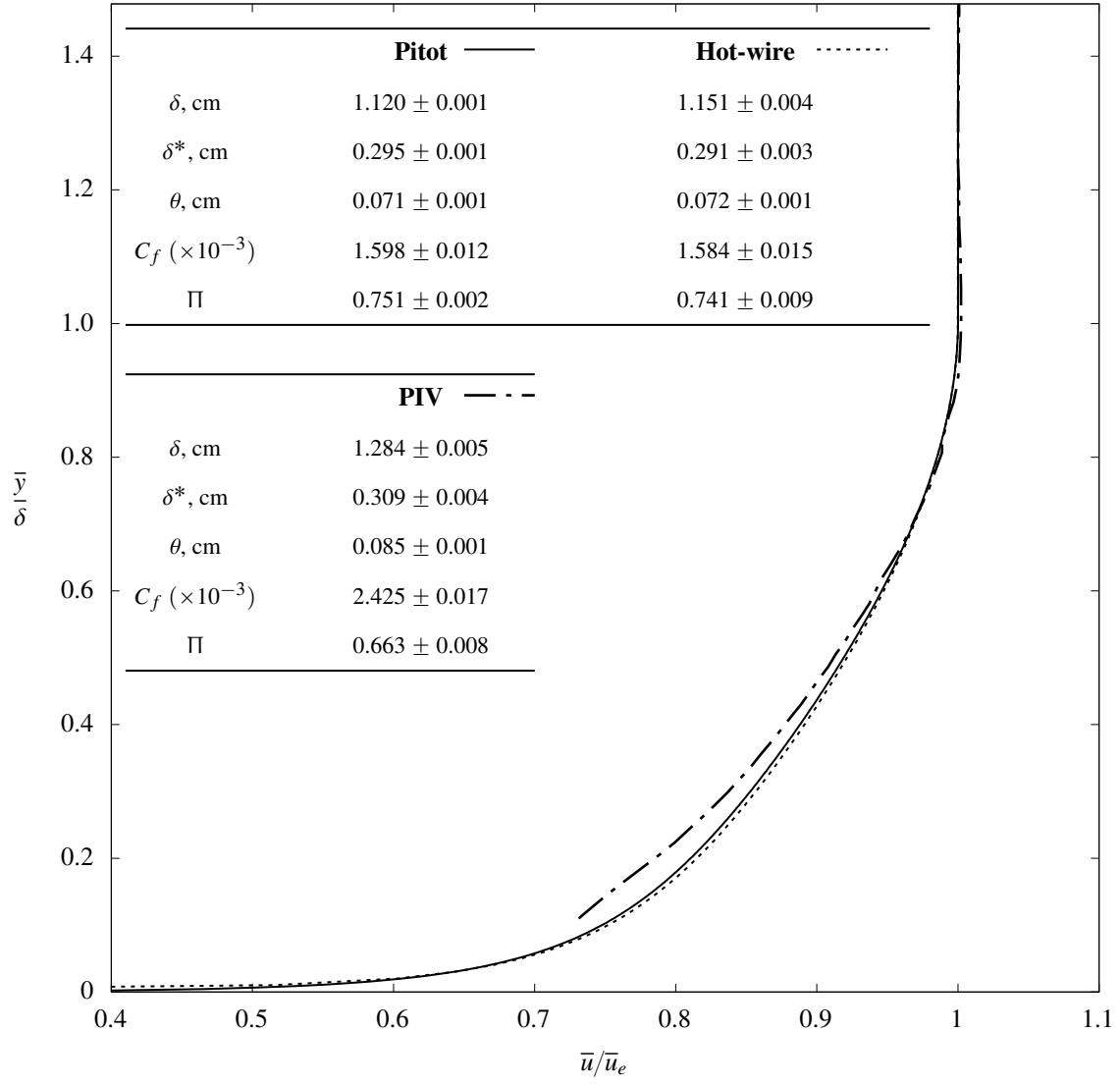


Figure 13 Comparison of incoming boundary layer mean velocity profiles.

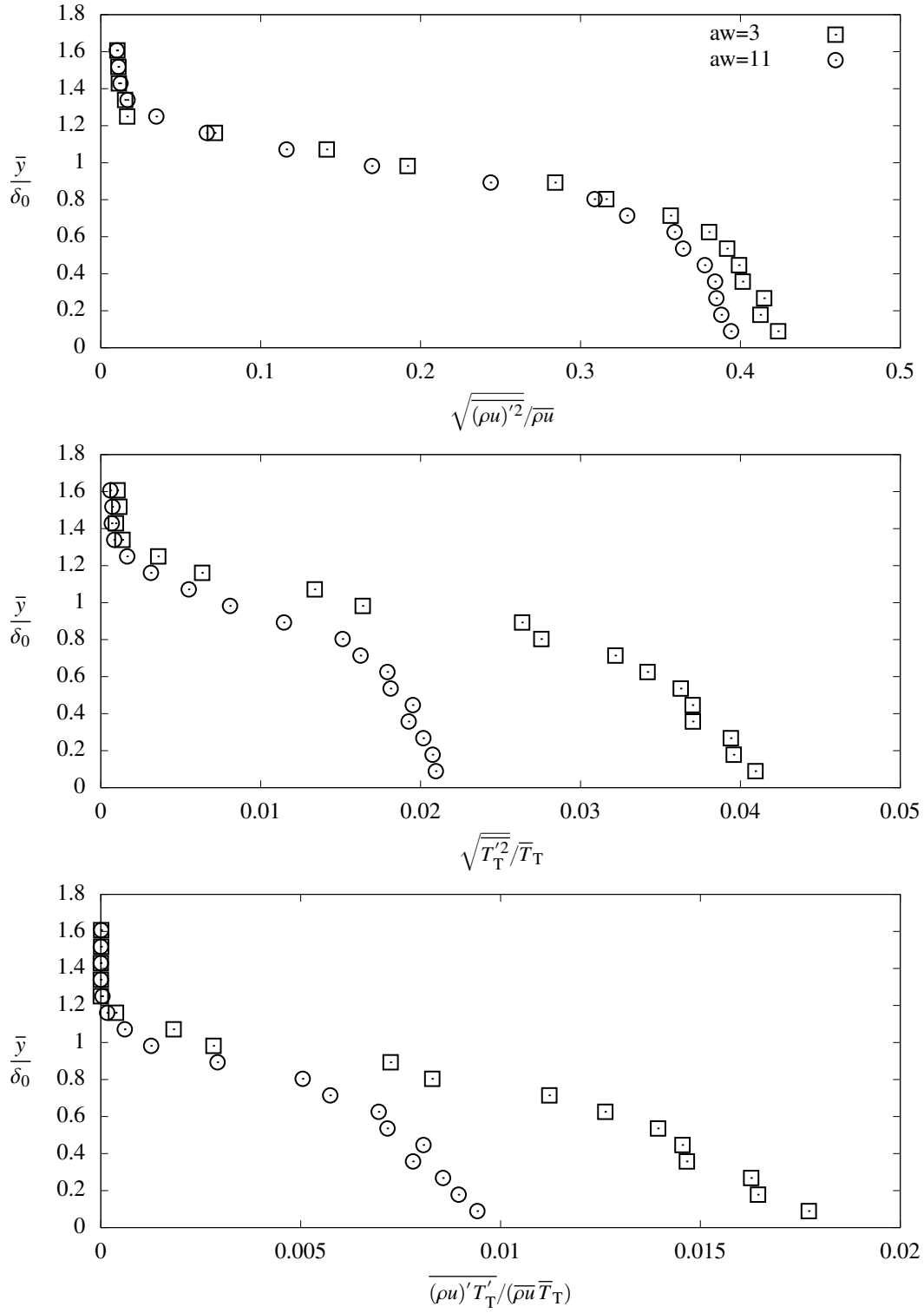


Figure 14 Incoming boundary layer turbulent intensities.

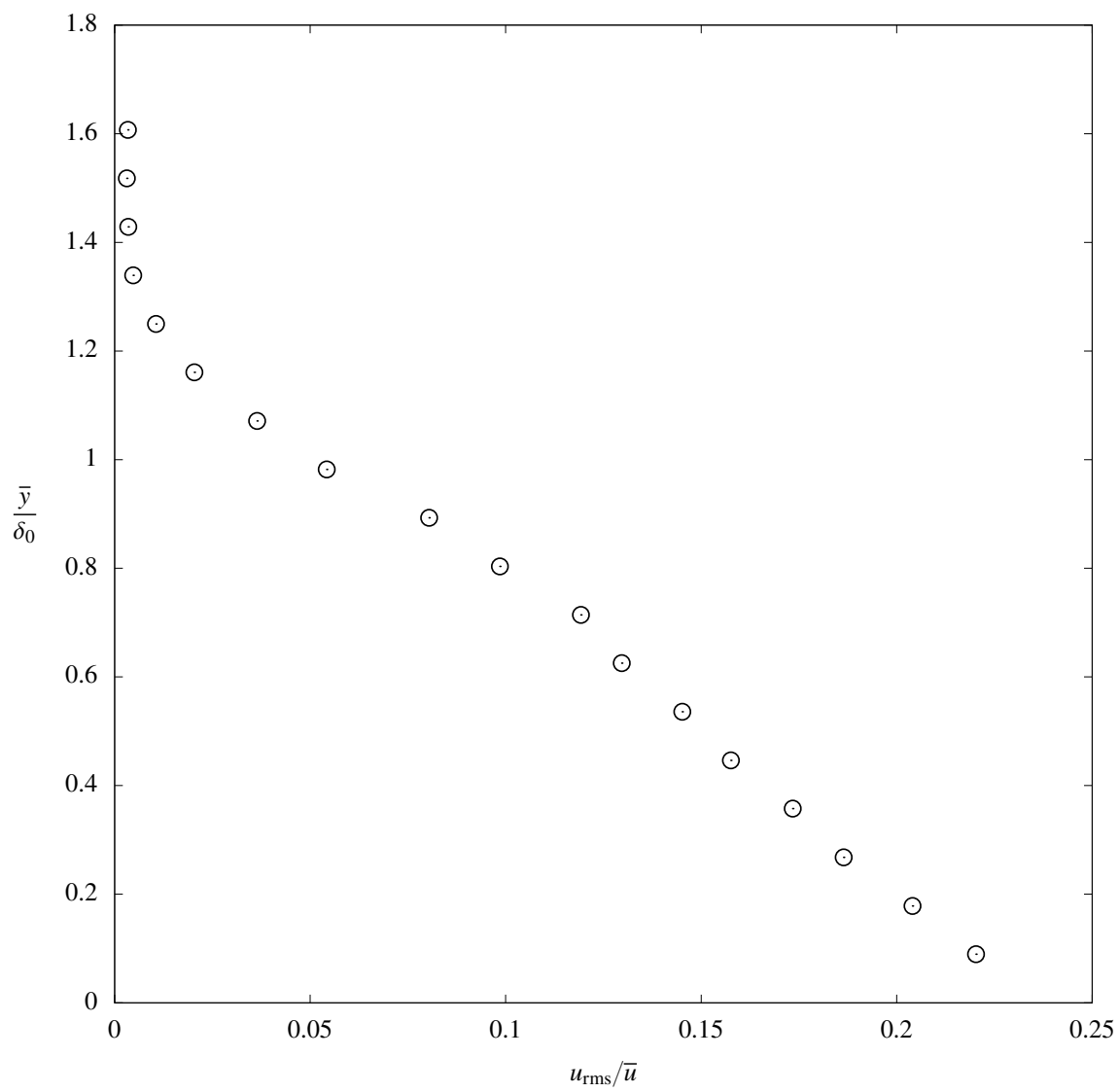


Figure 15 Incoming root-mean-square velocity profile.

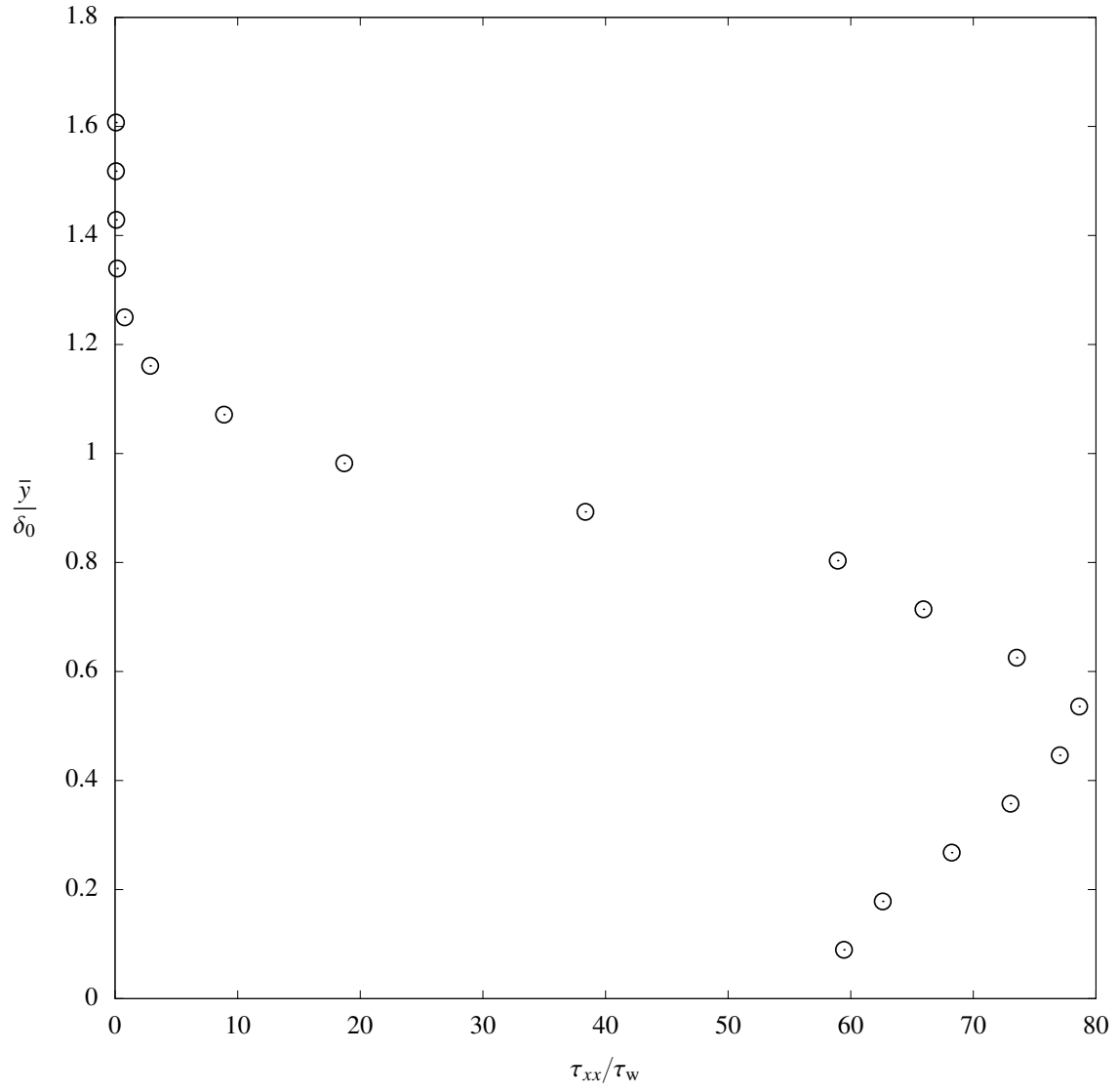


Figure 16 Incoming streamwise normal stress profile.

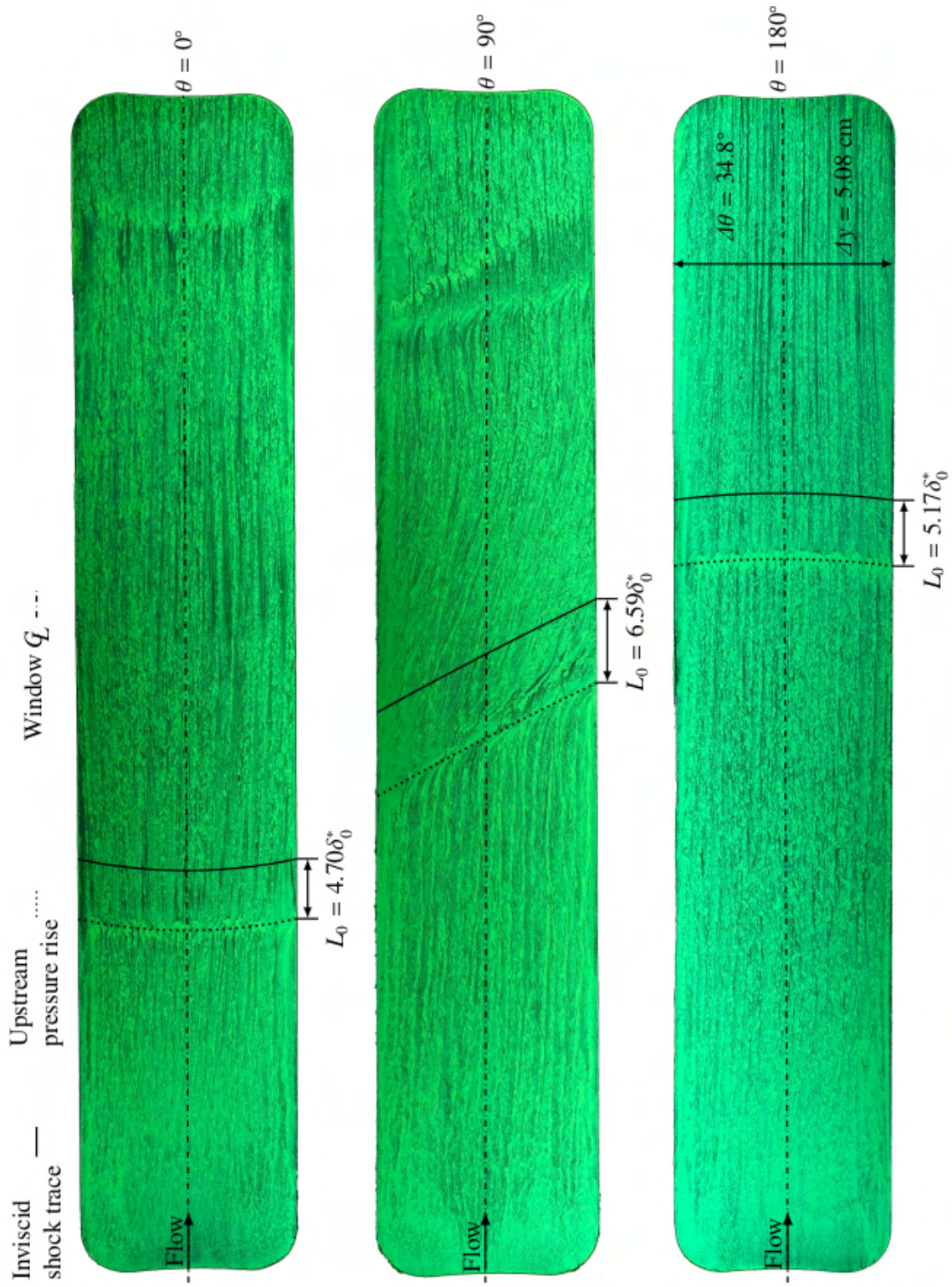


Figure 17 Surface flow visualization for $\alpha = 10$ and ℓ_{OF1} .

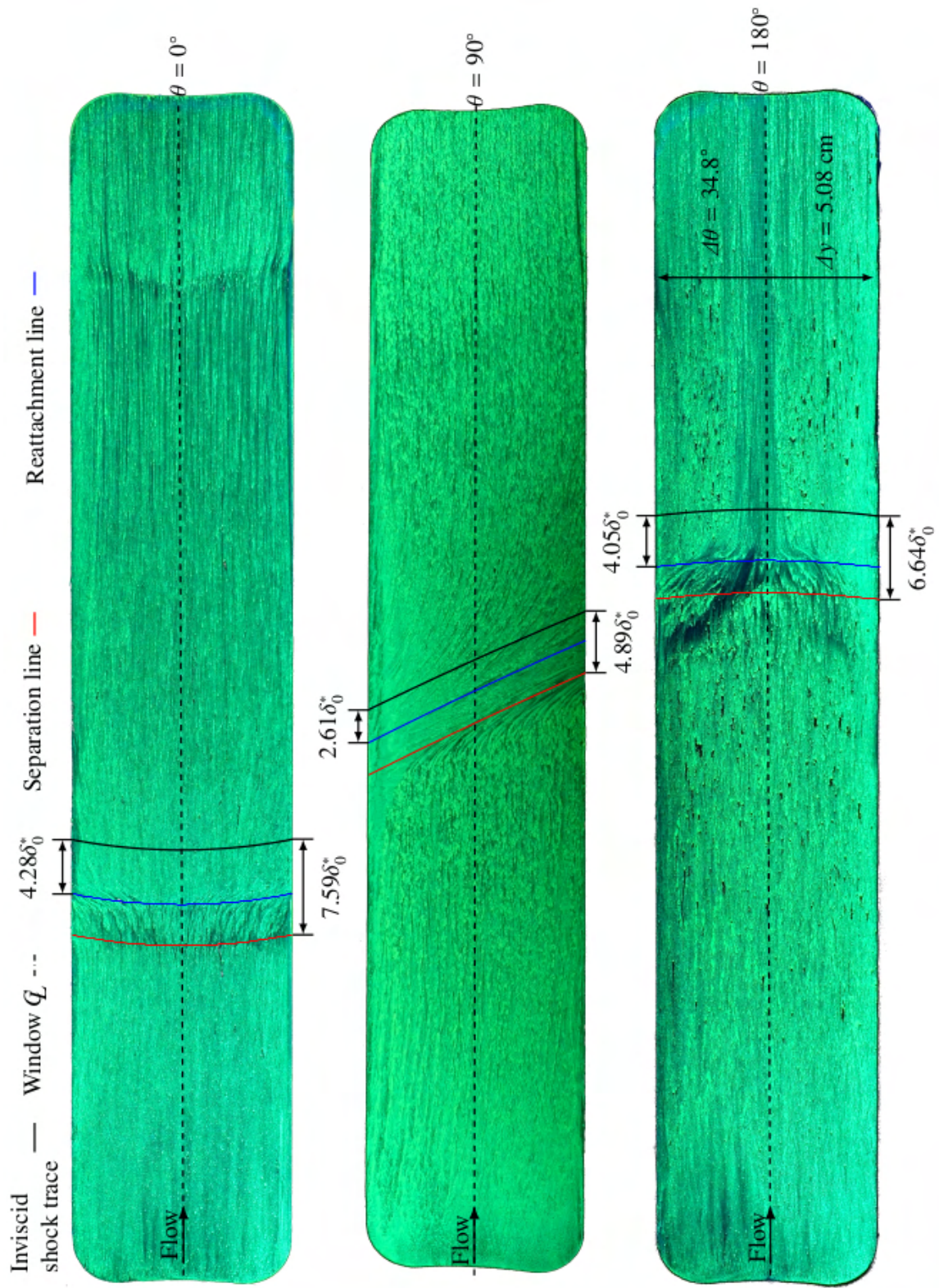


Figure 18 Surface flow visualization for $\alpha = 13.5$ and ℓ_{OF1} .

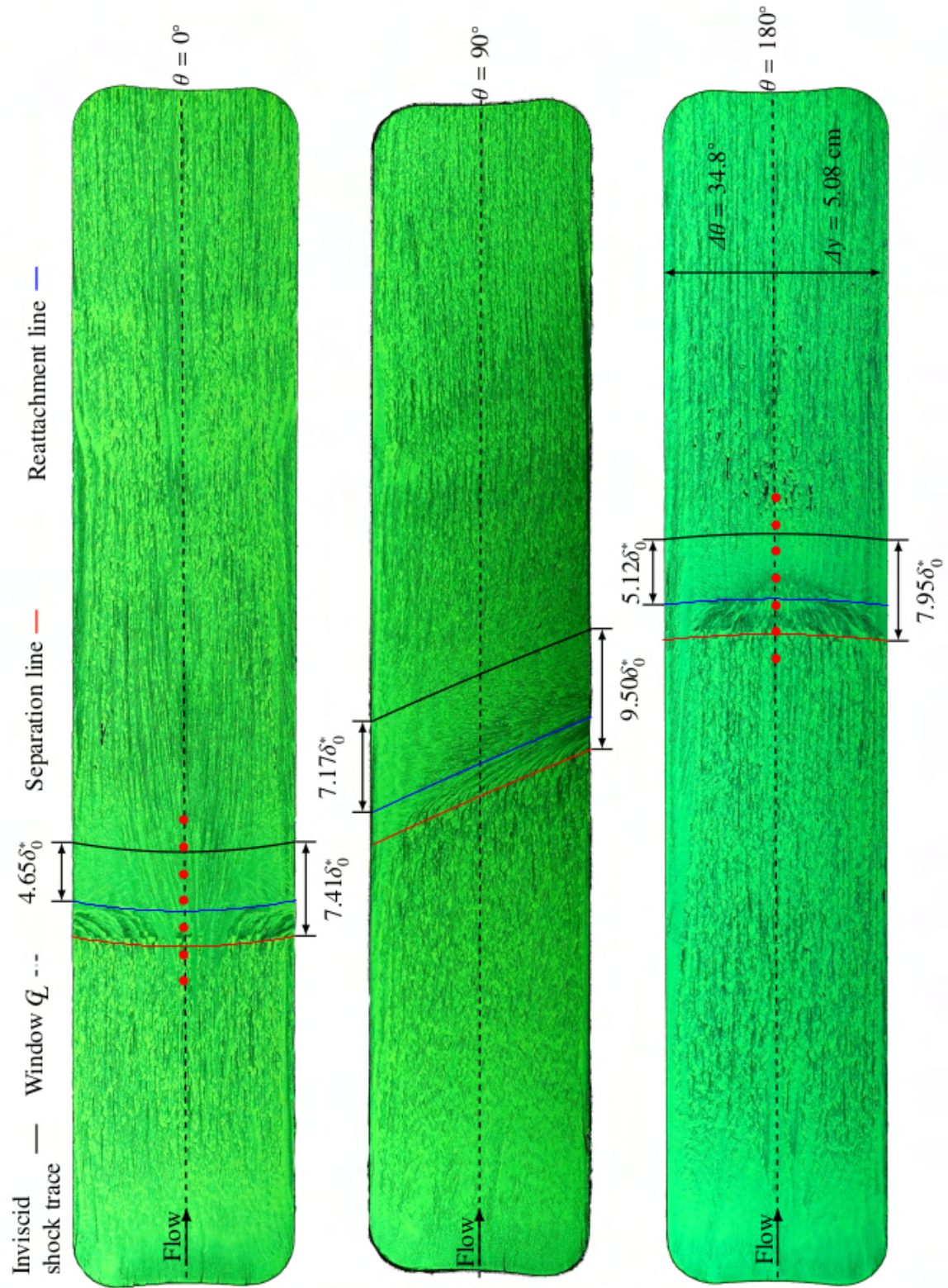


Figure 19 Surface flow visualization for $\alpha = 16$ and ℓ_{OF_1} .

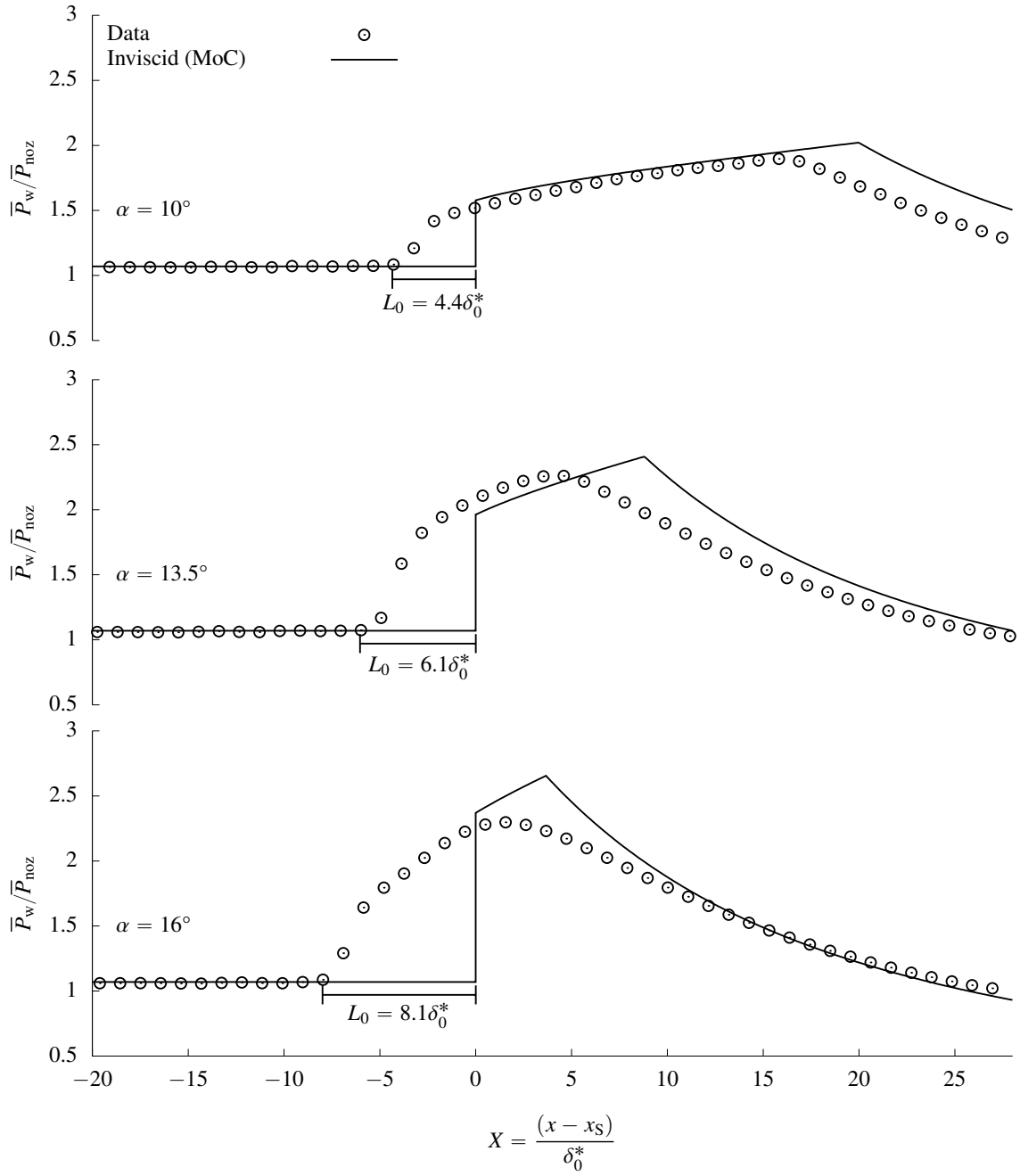


Figure 20 Static pressure distribution with shock generator aligned on centerline.

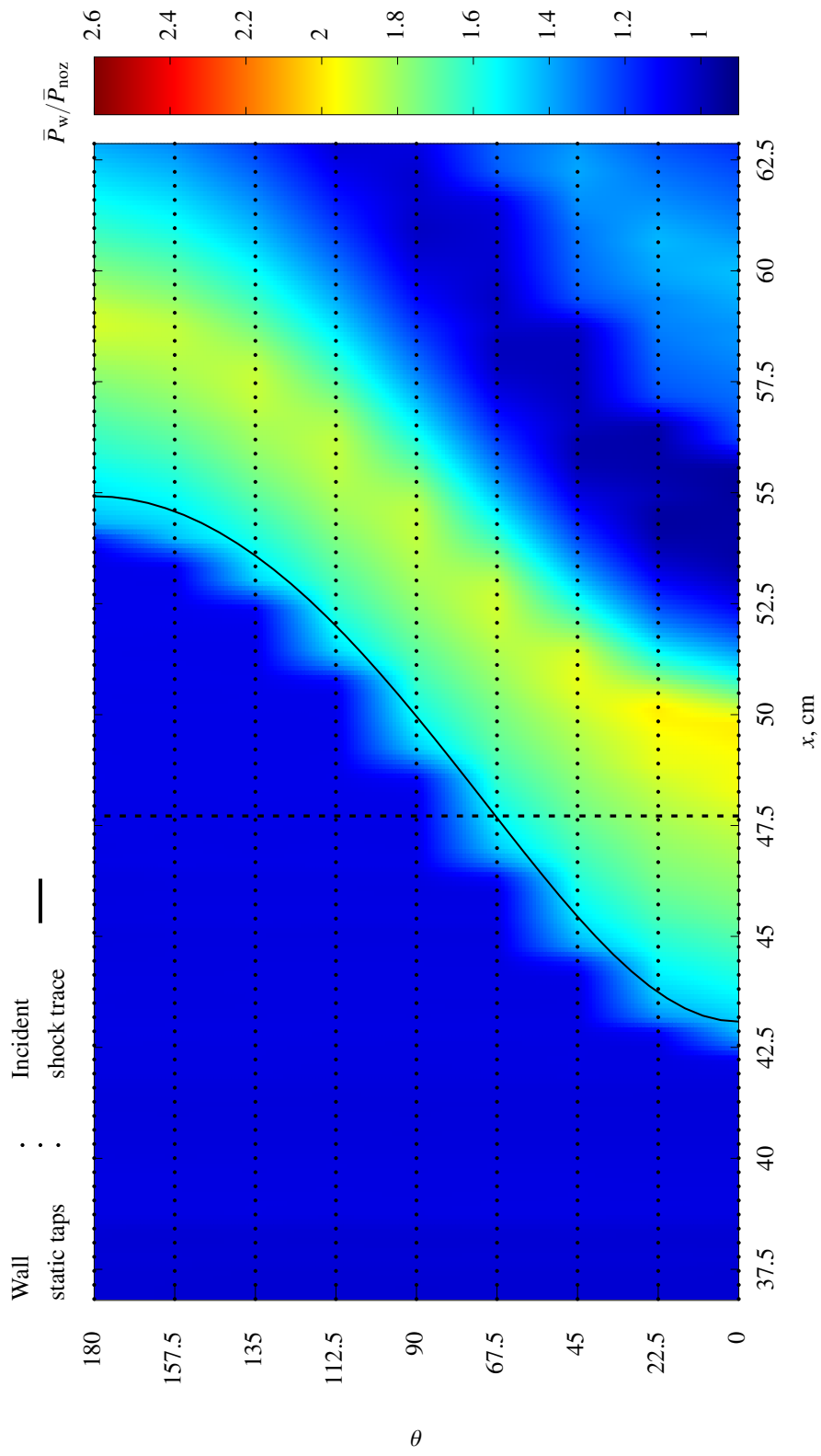


Figure 21 Wall pressure contour for $\alpha = 10^\circ$ and ℓ_{OF_2} .

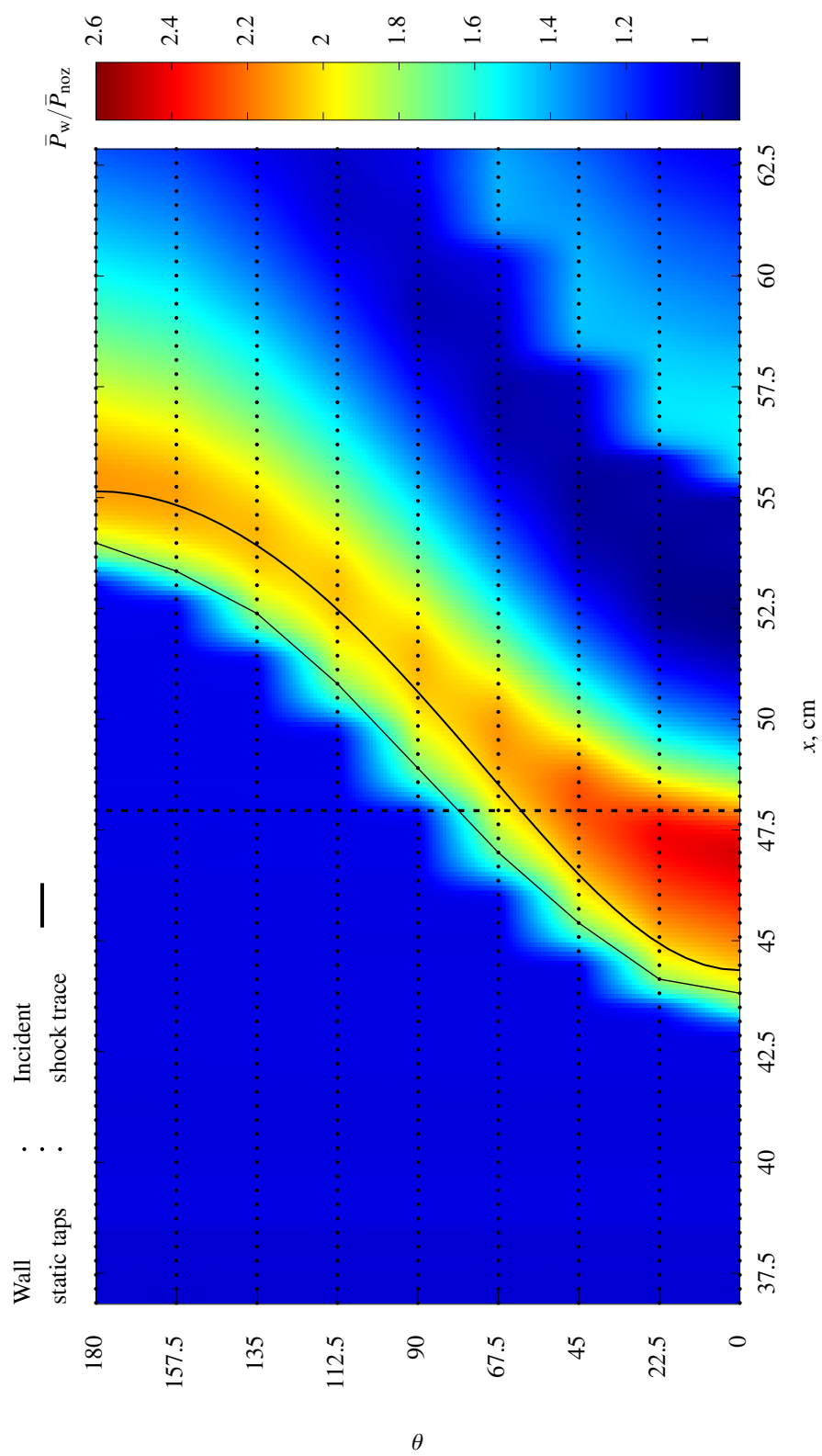


Figure 22 Wall pressure contour for $\alpha = 13^\circ$ and ℓ_{OF_2} .

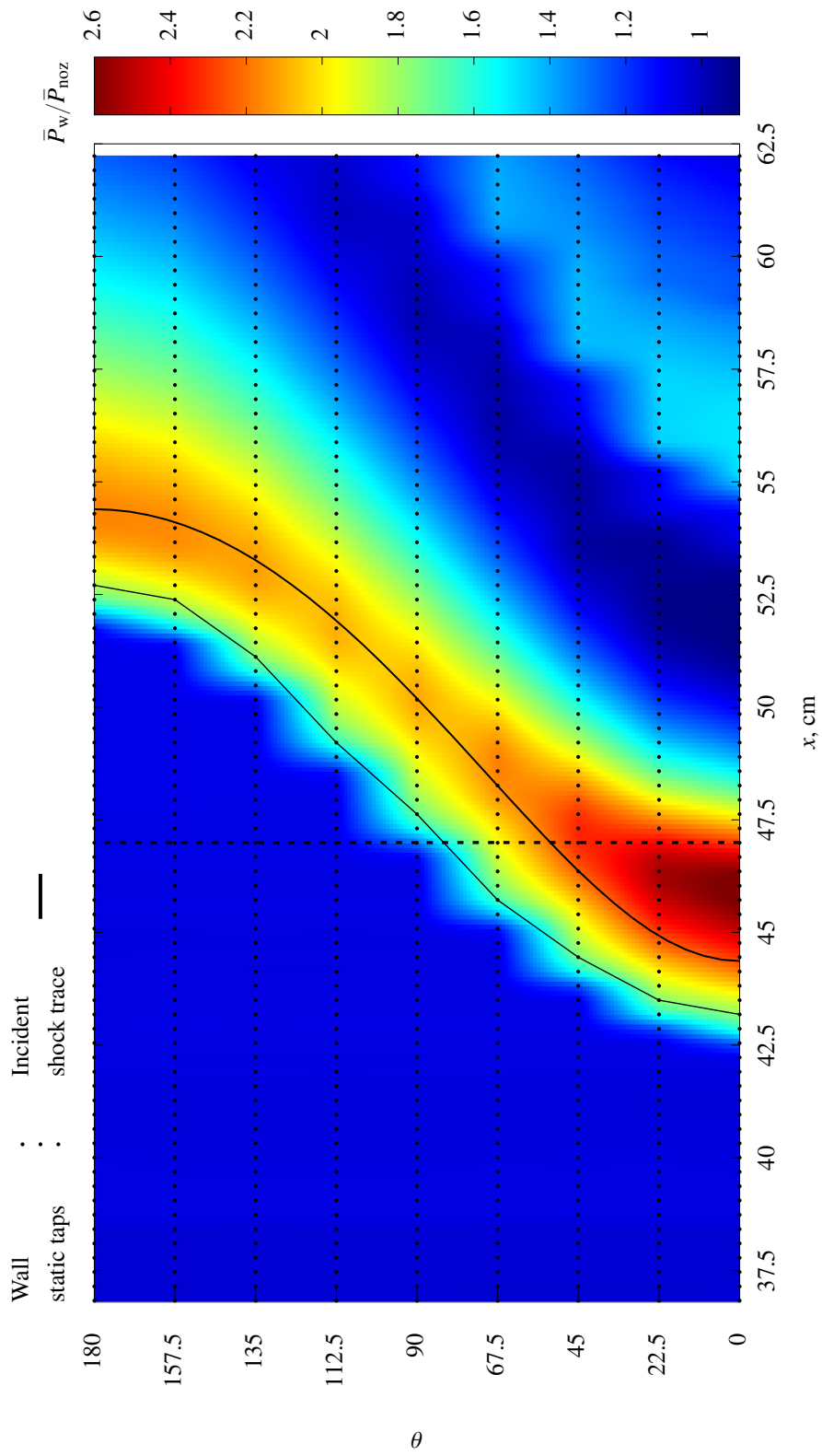


Figure 23 Wall pressure contour for $\alpha = 16^\circ$ and ℓ_{OF_2} .

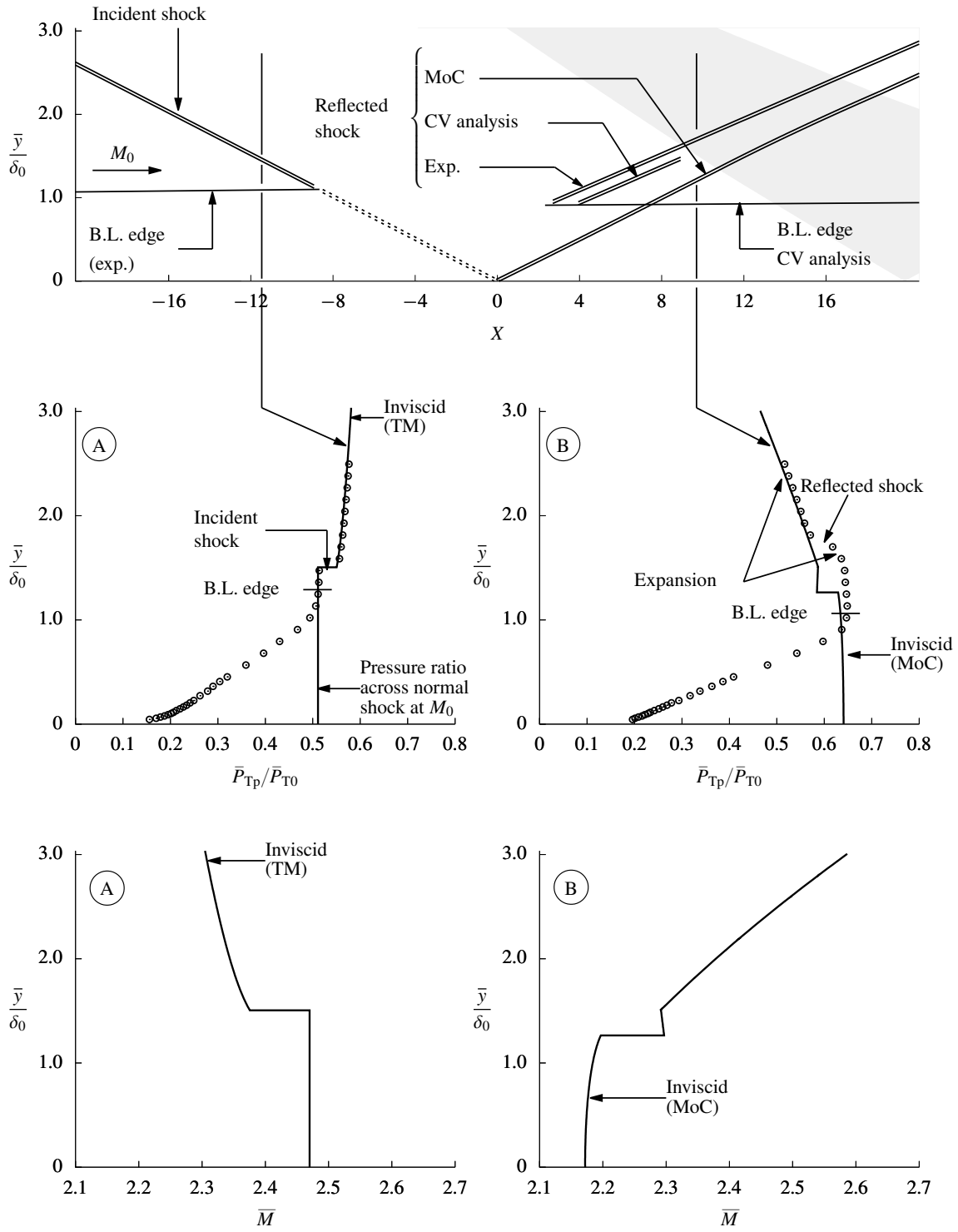


Figure 24 Unseparated Pitot pressure interpretation.

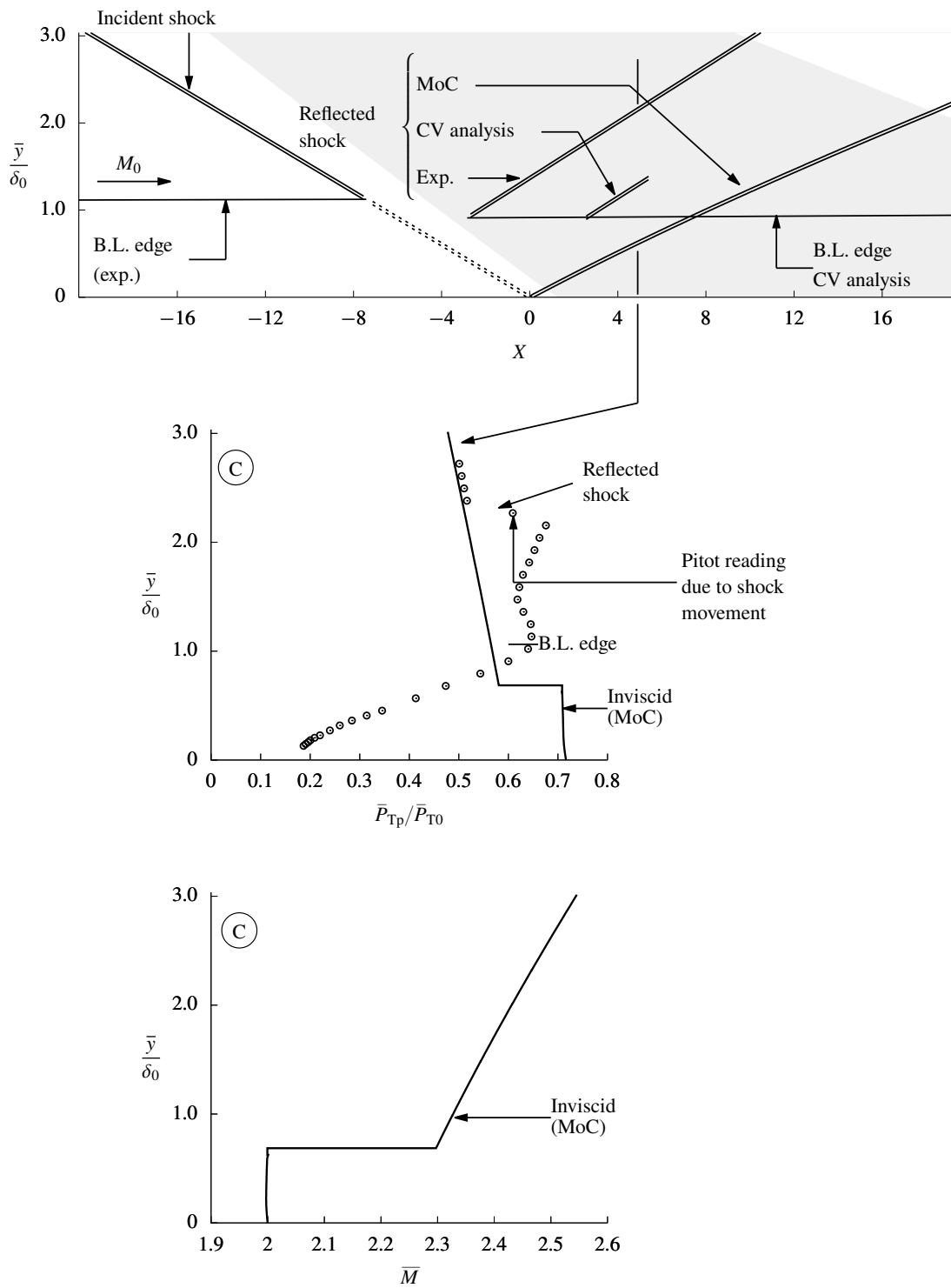


Figure 25 Separated Pitot pressure interpretation.

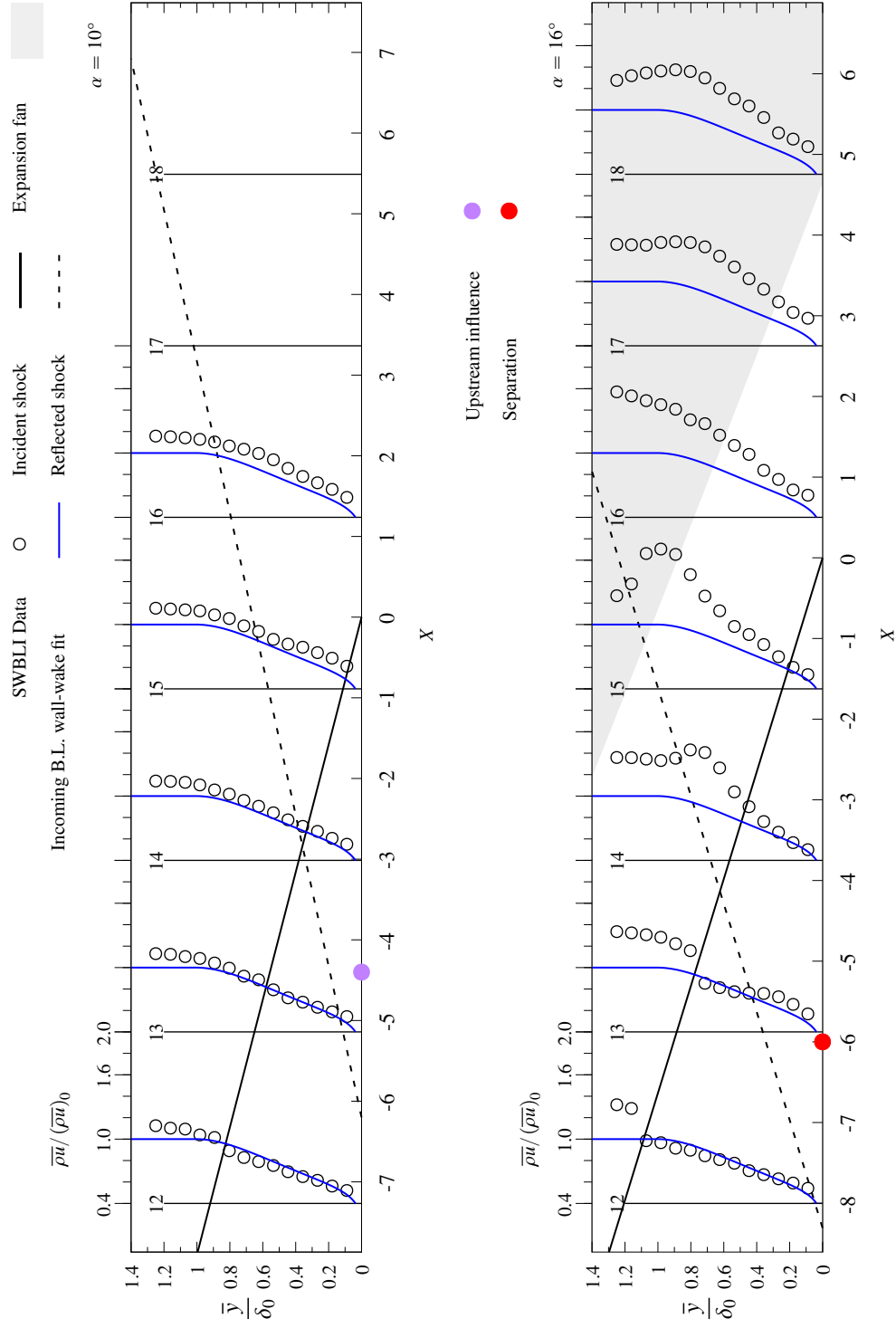


Figure 26 Mean mass flux on center line.

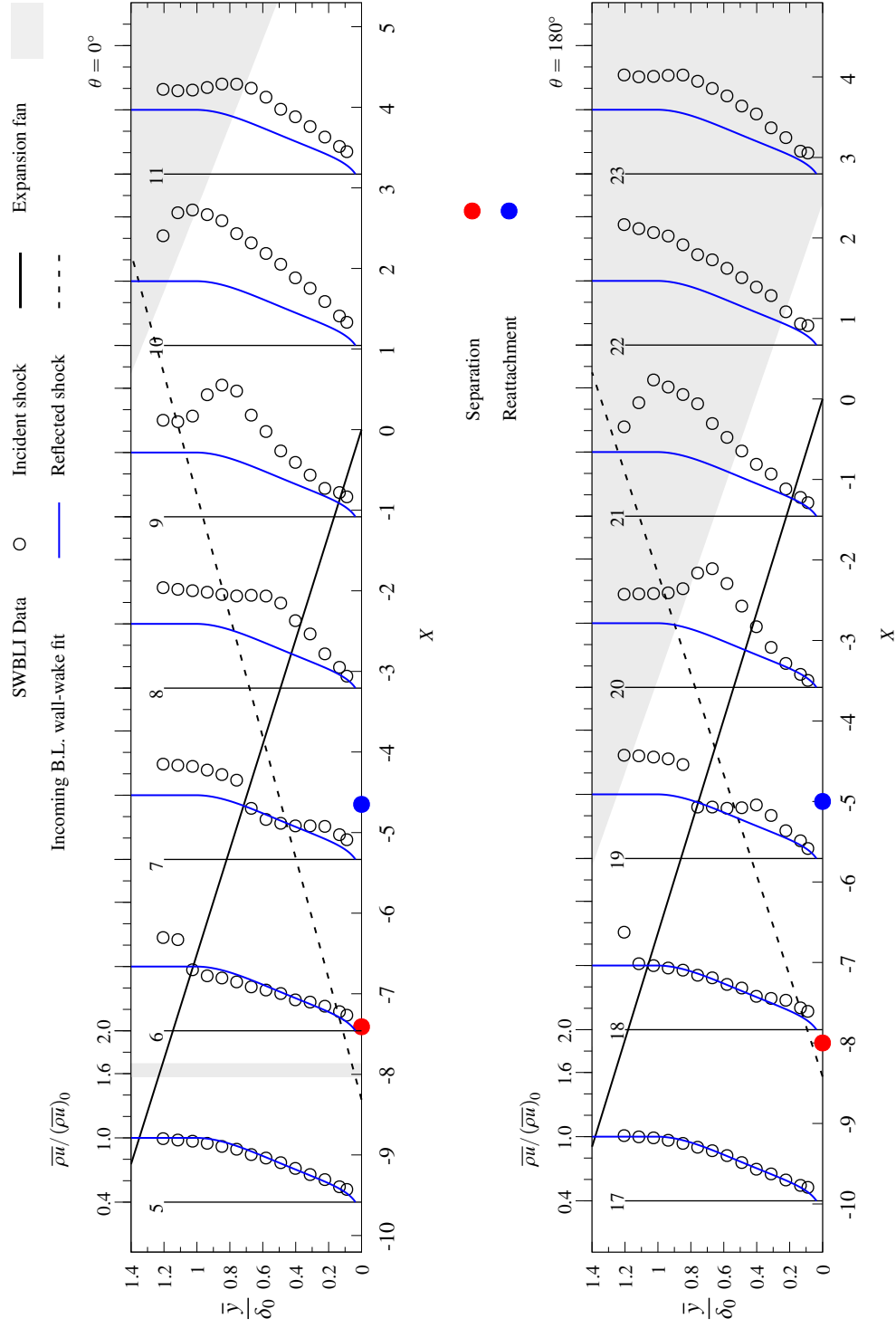


Figure 27 Mean mass flux on offset 1.

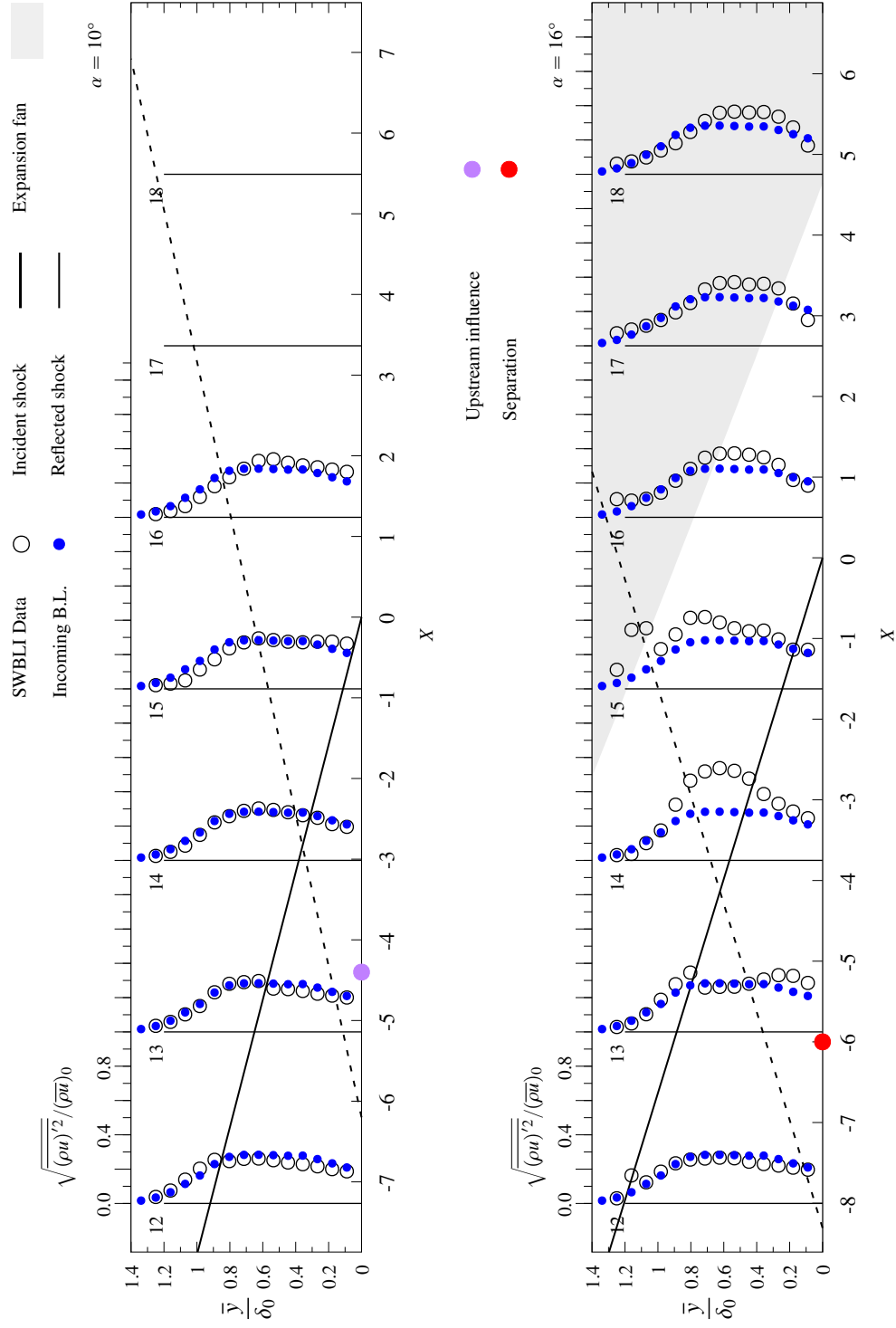


Figure 28 Mass flux turbulent intensity on centerline.

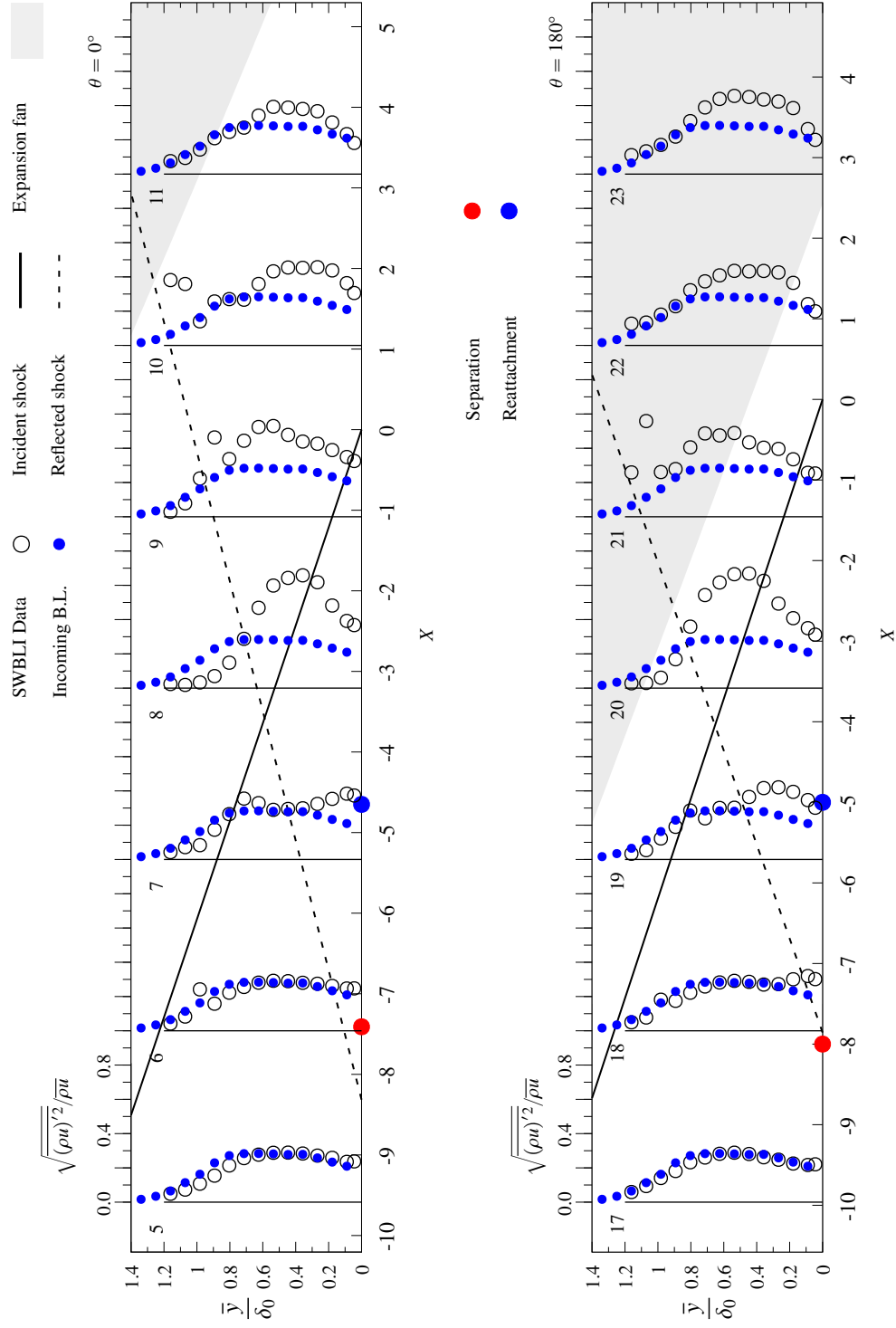


Figure 29 Mass flux turbulent intensity on offset 1.

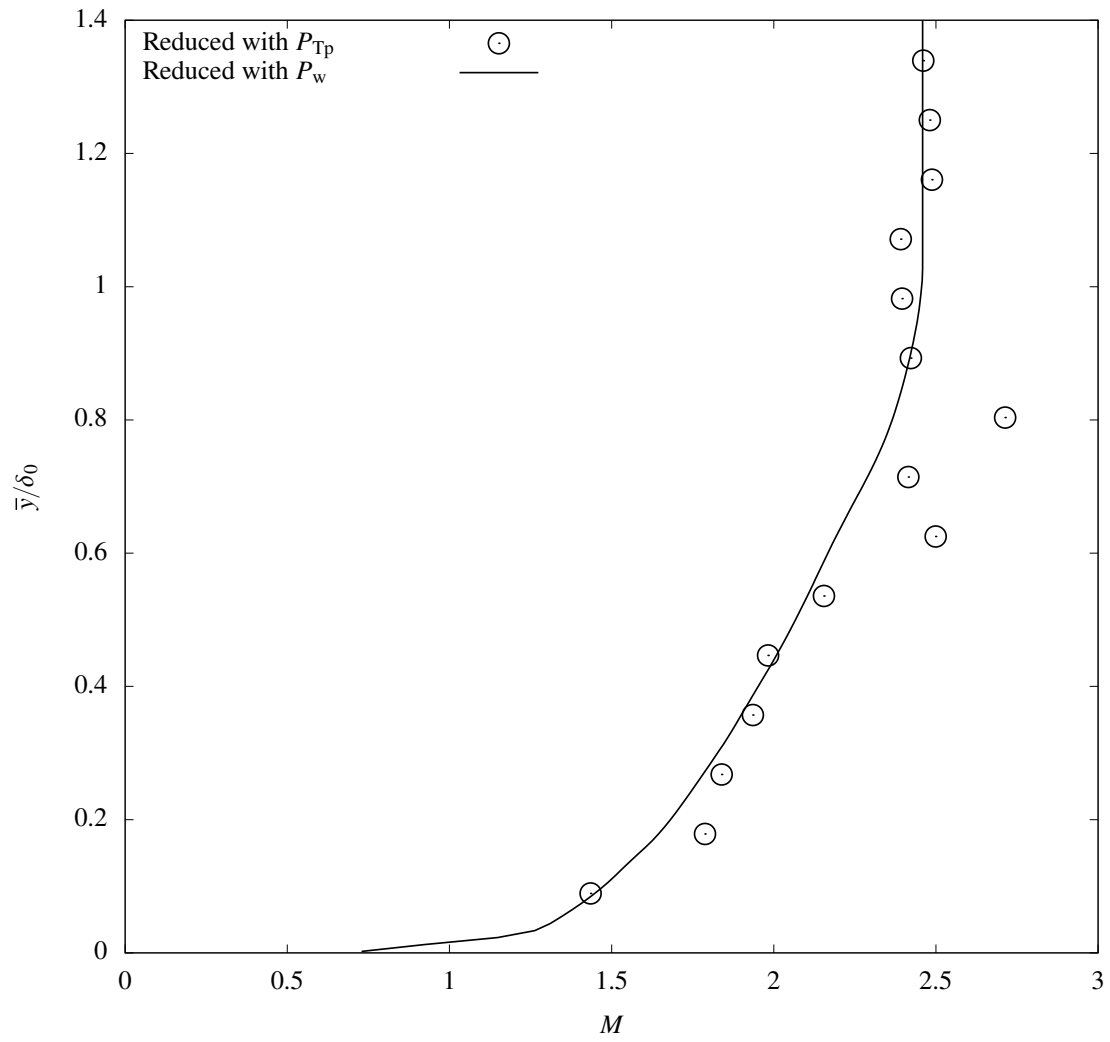
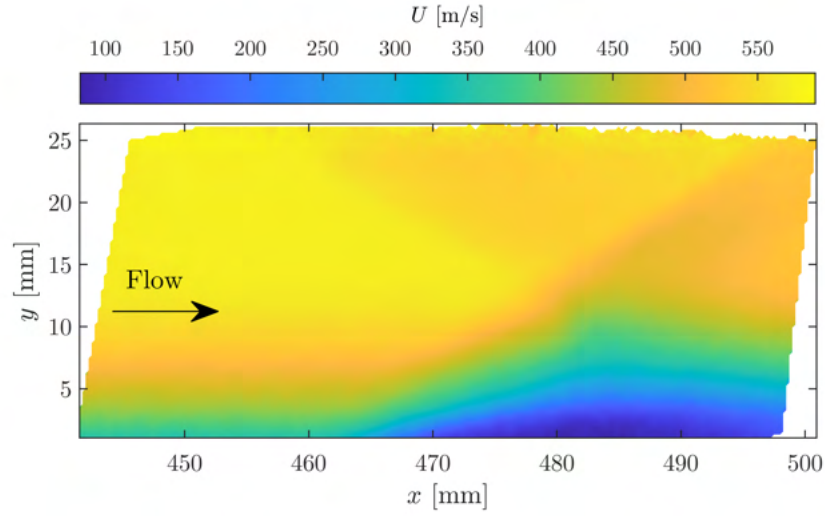
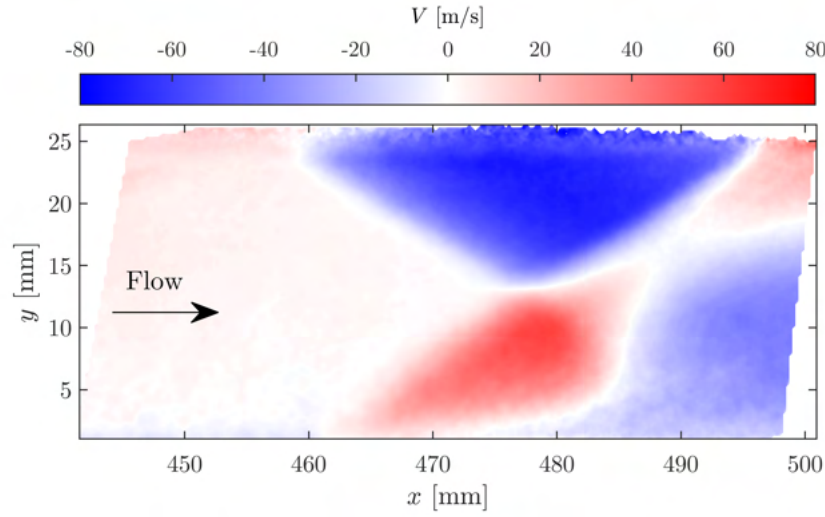


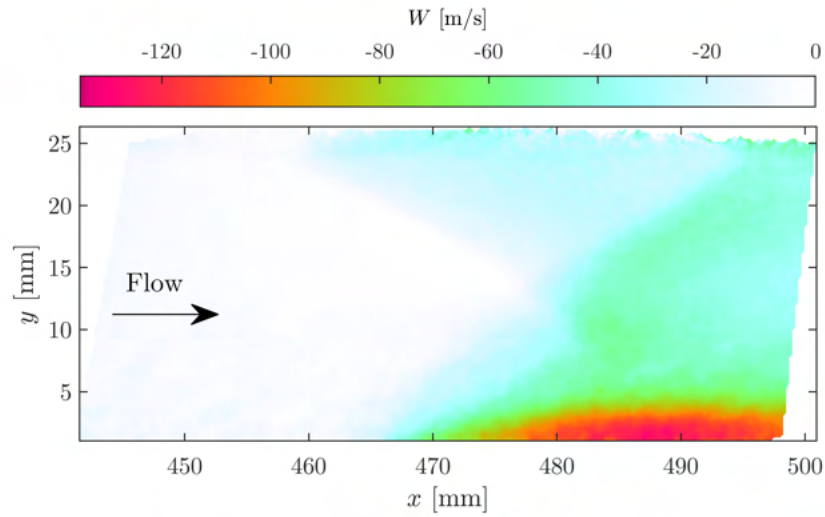
Figure 30 Incoming Mach number profile reduced with pitot and wall pressures.



(a) Streamwise velocity, U .



(b) Wall-normal velocity, V .



(c) Spanwise velocity, W .

Figure 31 PIV mean velocity field example for $\alpha = 16^\circ$ at $\theta = 90^\circ$.