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Visualization of Shock Wave Phenomenon around a Sharp Cone Model at Hypersonic Mach Number in a Shock Tunnel using High Speed Schlieren Facility

M. Saiprakash [1†], C. SenthilKumar [1], G. K. Sunil [2], S. P. Rampratap[2], V. Shanmugam[2] and G. Balu[2]

[1,] *Department of Aerospace Engineering MIT Campus, Anna University, Chennai-600044,India*
[2] *Directorate of Aerodynamics, Defence Research & Development Laboratory, Hyderabad -500058, India*

†*Corresponding Author Email: iamsaiaero@gmail.com*

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ABSTRACT

The flow field around a Sharp cone model configuration has been investigated by means of Schlieren facility in hypersonic shock tunnel. The time dependent evolution of flow around a cone of angle 11.38° with base radius of 150mm has been visualized for a flow Mach number M = 6.5. Experiments have been carried out with Helium as driver gas and air as test gas to visualize the hypersonic flow field. The flow establishment, steady state, and termination process of the hypersonic flow have been visualized for two different angles of attack, namely 0°&5°. Experimentally measured shock angle compares well with the theoretical and the computational study. The measured shock layer thickness compares well with the numerical simulation for both angles of attack.

Keywords: Sharp Cone model; High speed Schlieren facility; Hypersonic shock; Tunnel; Shock layer thickness.

NOMENCLATURE

DRDL	Defence Research &Development Laboratory	ρ	density
M	Mach number	μ	dynamic viscosity
P	pressure	α	angle of attack
T	temperature	∞	free stream condition
		0	stagnation condition
		2	condition behind the Pitot shock wave.
γ	specific heat ratio	5	condition behind the reflected shock wave

1. INTRODUCTION

Advances in diagnostic technique for the visualization of flow field around a cone shaped configuration is a prominent research area in hypersonic flows. The variation in flow properties is extremely large around the vehicle due to the existence of strong shock wave and thin viscous layer. The required flow field over the model is a very complex phenomenon in hypersonic shock tunnel. The flow field in transparent media with complex refractive index can be studied by using various optical techniques. The Gladstone-Dale formula links the refractive index of gas to the density of gas. The relation between refractive index of transparent gas and the density of gas is called Gladstone-Dale equation: n-1 =Kρ where K is Gladstone-Dale constant. This constant value for air varies between 2.239 x 10-4 to 2.33x 10-4 m3 /kg at the temperature of 288K. Merzkirch (1987) discovered the visualization technique and it includes shadowgraph and Schlieren. Until the 19th century, flow field visualization for compressible flow was not established properly and then many techniques were developed to visualize the shock wave around bodies at hypersonic Mach number. This can be done by changes observed in a light beam after passing it through the flow field.

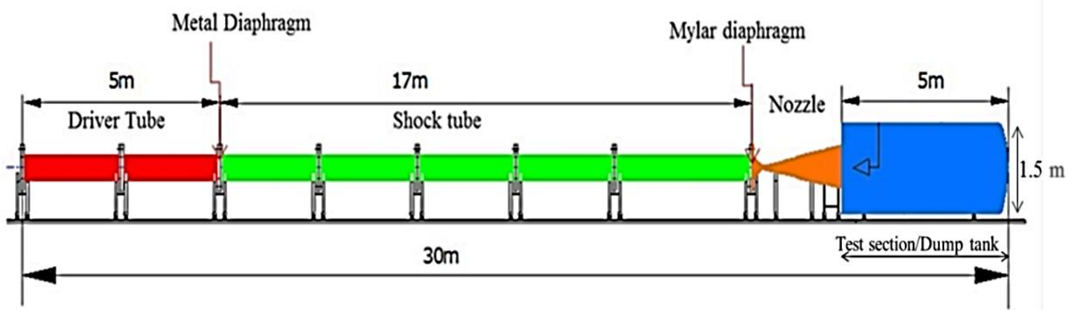


Fig. 1. Layout of Hypersonic shock tunnel.

Interferometer is sensitive to density changes in flow field. Set–up arrangement is not as easy as any other optical techniques. Mach–Zehnder interferometer, holographic interferometer, and differential interferometry are widely used systems. Schlieren, Shadowgraph, Interferometer are standard techniques used for visualization of flows at high speed by (Merzkirch 1987; Settles 2001, Slavica Ristić 2006).Both Schlieren and Shadowgraph techniques are sensitive to the first and the second derivatives of density perpendicular to light propagation direction, respectively. Schlieren method is used to visualize the turbulence phenomena and also used to measure heat transfer rate by Devia *et al.* (1994).

Colour Schlieren technique is used for quantitative measurement under transient flow. Such techniques are reported in (Kleine and Gronig 1991,1992,1993; Elsinga *et al.*2004).The electrical discharge techniques are used to visualize the shock wave around the model by (Jagadeesh *et al.*1996, Nagashetty *et al.* 2000).(Jagadeesh *et al.*2002) investigate the effect of test gas on shock stand-off distance along the large angle cone model The major drawback of the above techniques is that we can take only one photograph of the entire steady state flow in shock tunnel. As a consequence, it will not be possible to clearly compute the dynamics of flow along the model in shock tunnel. The present study is concerned with the characteristics of tunnel starting, steady flow, and flow termination process is critical for shock tunnel testing. Hence, an attempt has been made in DRDL (Defence Research and Directorate Laboratory, Hyderabad) to visualize the shock wave phenomenon in the shock tunnel at flow Mach number of 6.5. The shock shape and shock layer distance are important parameters which influence the heat transfer on re-entry vehicle at hypersonic speeds, in this sense the objective of the present study are as follows 1) to visualize the time dependent evolution of flow around a sharp cone model having an apex angle of 11.38°and base diameter of 150 mm.

2) to compare the shock layer distance obtained using CFD simulation with experimental results at 0° and 5° angle of attack.
3) to validate the use of theoretical method available for the prediction of shock angle in open literature.

2. EXPERIMENTAL FACILITY

Helium was used as the driver gas and air as test gas. The shock tunnel consists of a shock tube made up of stainless steel having the diameter of 88mm which was connected to the convergent divergent nozzle and the test section. The driver and the driven section of shock tube were separated by Aluminum diaphragm. The shock tube and the nozzle were separated by Mylar diaphragm. The pressure transducer (Model Number 113A24, sensitivity 71.79 mV/bar) was mounted very close to Mylar diaphragm, it was used to measure the nozzle reservoir condition of the system. The output signal from transducers were connected to trigger based data acquisition system through signal conditioning. The optical quality glass window was used for flow visualization. Once the diaphragm got ruptured, test gases expanded through convergent divergent nozzle from the inlet of nozzle and hypersonic flow was generated in the test section. The free stream properties along the axis of the test section were calculated by using isentropic relation from Anderson(1989).The conventional shock tunnel is capable of producing the stagnation enthalpy of ~ 3 MJ/Kg. The free Stream Mach number could be varied by using different throat sections which could produce a Mach number ranging from 6 to 7 and 6 to 10 for a nozzle exit diameter of 590 mm and 1000 mm respectively. The schematic view of shock tunnel is shown in Fig. 1.

3. EXPERIMENTS AND TEST CONDITION

The experiments were carried on a sharp cone model at 0^0 and 5^0 angles of attack with air as the test gas to simulate the free stream condition. The tunnel was operated at lower enthalpy of 1.2 MJ/kg. Table 1 gives the test condition of hypersonic shock tunnel at $\alpha = 0°$.The calibrated Mach number was used to determine the free stream conditions. The test flow Mach number was estimated to be 6.5 by using Pitot rakes (6 Pitot tubes) traversing along the axis parallel to the flow where it was mounted 115mm from the Nozzle exit. The model was placed at the center of the test section to visualize the shock wave clearly over the sharp cone model. The test condition in the shock tunnel was computed from isentropic relations of gases expanded through the conical Nozzle. The Dynamic pressure of the atmosphere and flight Mach number correspond to an altitude of approximately 31km as from reference Heiser (1994).

Table 1 Nominal Test condition during the present experiment

Driver gas	Helium
Stagnation pressure, P_{05} (bar) (±8.7%)	24.12
Shock Mach number, M_s (±1.2%)	3.12
Free stream static pressure, P_∞(Pa)(±7%)	889
Free stream static temperature, T_∞(K) (±2%)	132
Free stream Mach number, M_∞(±0.1%)	6.5

*The percentage uncertainties inside the brackets.

4. DYNAMIC SCHLIEREN SET-UP

A Z-type Schlieren technique was used for visualizing inhomogeneity's in transparent gaseous medium at hypersonic flow field on a model. The Schlieren system consisted of a light source, two collimating mirrors, knife edge, and high-speed camera. In the present investigation, high quality optical glass window and quality lens were used to increase the sensitivity of Schlieren technique. This technique works on a non-homogeneous refraction of light due to variation in the refractive index of the media. A pair of concave mirrors was used to obtain a parallel beam of light that was allowed to pass through an optical glass window of the test section, after it was focused on the high speed camera through the knife edge. Due to irregularities in the medium, different parts of light in the test area would be deflected by different amounts and thus would not focus at similar points. The knife edge blocks the undisturbed light coming from different portions of the test section, depending upon the orientation and the amount of the refracted light rays. The speed of camera is insignificant for long duration experiments but for short duration facilities like shock tunnel, the type of camera plays an important role. During the shock tunnel testing, it was not possible to click the camera manually since the steady state duration of tunnel was 3 to 4ms. By using an ordinary camera it would not be possible to monitor the dynamics of flow in the shock tunnel. The only way to monitor the hypersonic flow in the shock tunnel was by using a high speed camera. In the present study, Phantom 630, a high speed camera was used to record the Schlieren images. The camera has automatic continuous adjustable resolution and variable frame rate features depending upon the tunnel operating conditions. The high speed camera was capable of recording 1000 frames per second with a maximum resolution of 1632x1200pixels. As the recording speed was increased, the pixels resolution reduced to 64 x64 pixels at 1,44,175 frames per second In the present study, Schlieren images were recorded at a speed of 14035 frames per second with a resolution of 384 x240 pixels. High speed camera operation and shock tunnel flow were synchronized using a trigger pulse generated by the pressure sensors located in the shock tube. The camera was focused on the test model to visualize the flow field. National instruments made PXI-8187 Lab VIEW RT controller with 2.5 GHz processor speed for interrupt based trigger based data acquisition with a digital clock frequency of 10 MHz was found to be most suitable for this purpose. A Z- type Schlieren set-up used for the flow visualization is shown in Fig. 2.

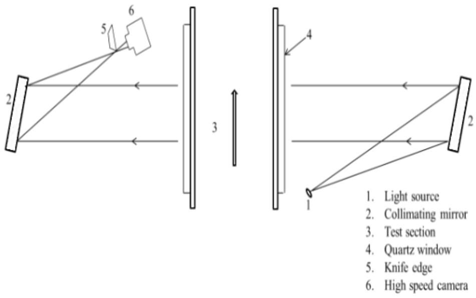


Fig. 2. Schematic view of Schlieren set up used for flow visualization.

5. RESULTS AND DISCUSSION

In the present investigation sharp cone model was used to capture the flow field at hypersonic Mach number M =6.5. During calibration, the variation in Mach number would be 1% up to a distance of 115 mm from the nozzle exit along the axial direction as measured by Pitot rake The Pitot rake was mounted more than 115mm from the nozzle exit, the Mach number increased along the test section more than a design Mach number due to conical flow from the Nozzle. The test model was placed at the same axial distance where it was ahead of Schlieren window. The Schlieren window was located 260 mm from the nozzle exit. As a result, the author was not able to capture flow feature upstream of the model. Figure 3 shows time resolved Schlieren photographs recorded during the tunnel experiments for a test model at α= 0°. The nozzle starting time, steady test flow, and flow termination process of hypersonic flow over the sharp cone model were clearly visible in these Schlieren images. The comparison of Nozzle starting time shown in the Schlieren images matched the Pitot tube measurement (Fig. 4) being carried out in the test section for both the test conditions. The flow reached the steady state of 997 µs. The presence of steady flow for about 3.5 ms was measured using a single Pitot tube which was useful test time in the shock tunnel operation. After the steady flow, the flow properties such as pressure, temperature, density, etc., began to drop. Further, from Fig. 4 it is clear that both the reservoir pressure (P_{05}) and Pitot pressure (P_{02}) start decreasing after a test duration of about 3.5 ms. The Shock structure in Schlieren images remained unaltered of about 9 ms but steady flow in Pitot measurement is 3.5 ms. This could be due to variation of Pitot pressure (P_{02}) to Stagnation pressure (P_{05}) ratios with respect to time [Fig. 5].

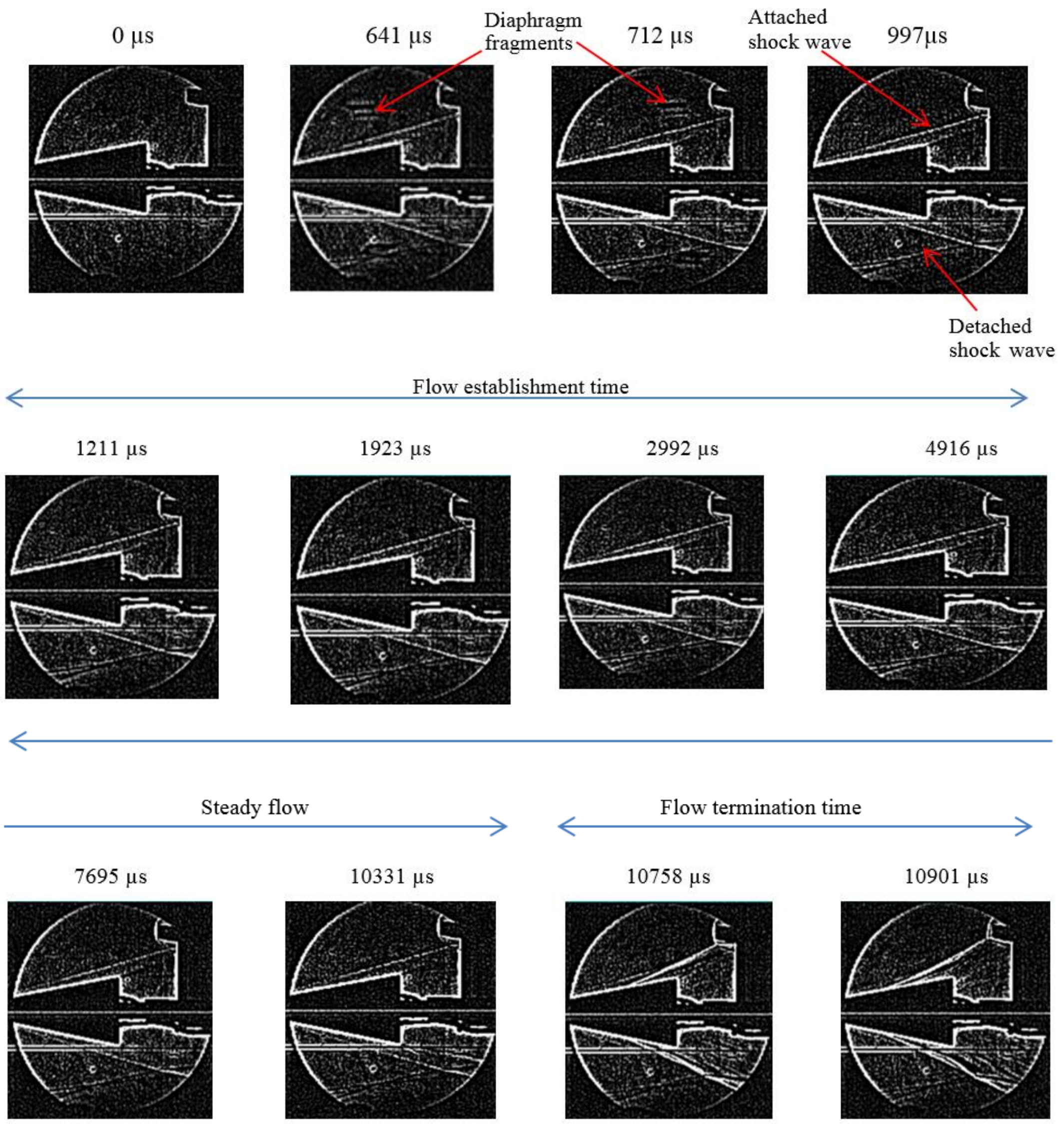


Fig. 3. Dynamics of flow field around a cone model at α= 0°.

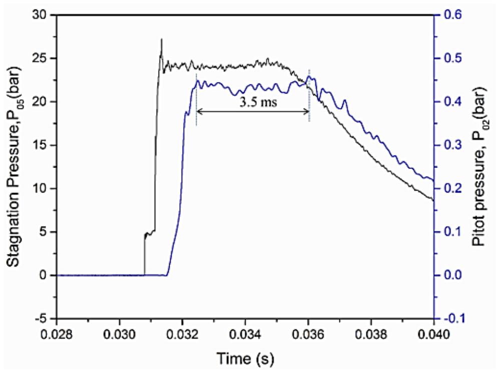


Fig. 4. Variation of Stagnation Pressure and Pitot pressure with respect to time.

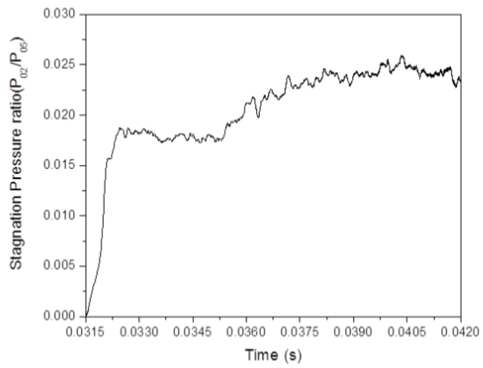


Fig. 5. Variation of Stagnation pressure ratio with respect to time for α= 0°.

This pressure ratio is a unique function of free stream Mach number and the specific heat ratio of the test gas remained steady for about 9 ms. As a result, the test Mach number remained constant during this time.

Once Mylar diaphragm got ruptured, the diaphragm fragments causes disturbance in the flow quality

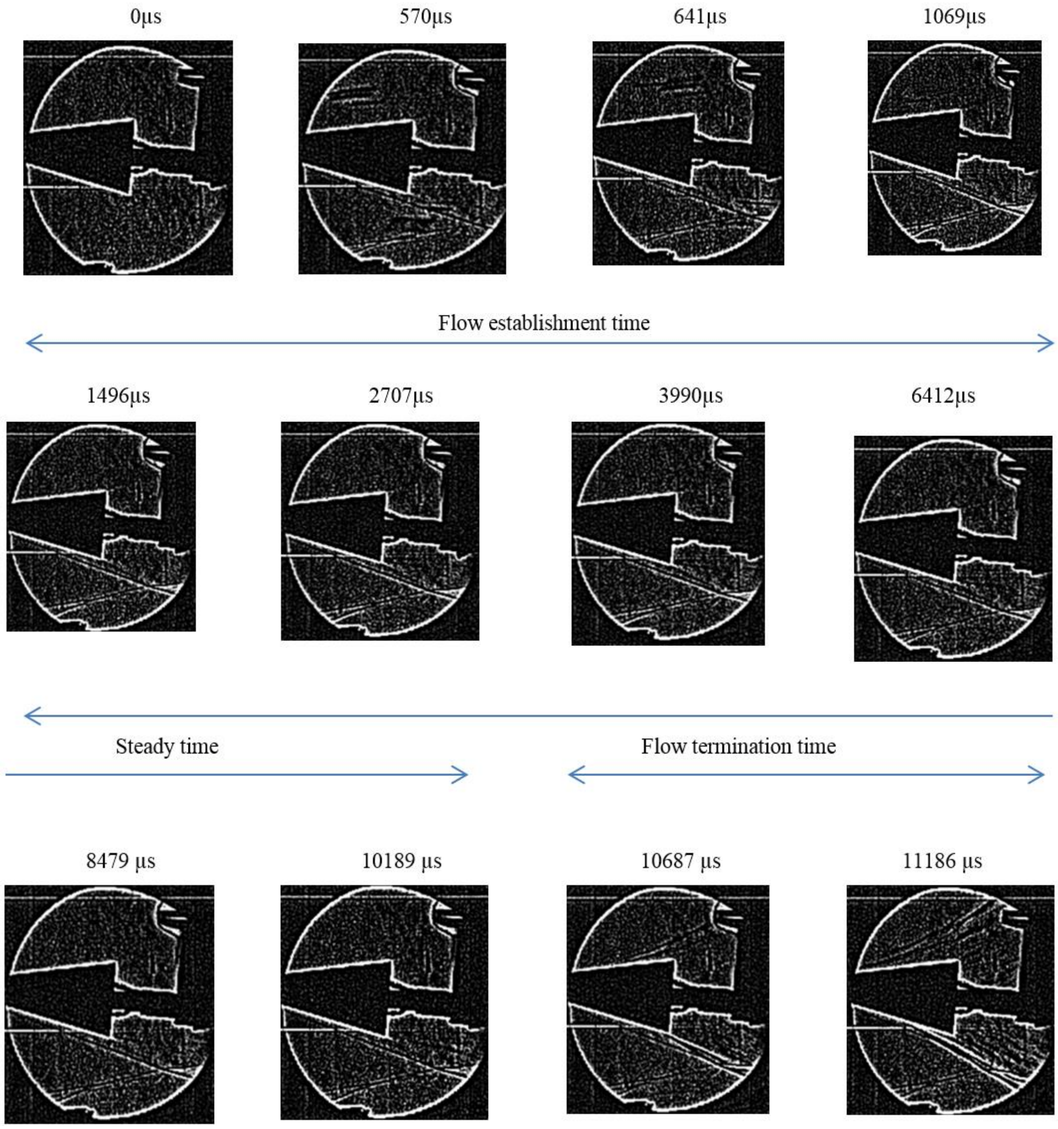


Fig. 6. Dynamics of flow field around a cone model at α= 5°.

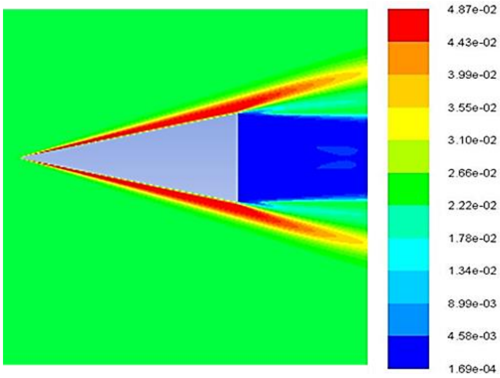
during tunnel testing. This phenomenon was clearly observable in sequential Schlieren images (Fig. 3) for the test duration of approximately from 570 µs to 641 µs. The high speed camera capable of capturing those images. At α= 0°, for M = 6.5 a similar pattern of shock wave existed over the top and bottom surface. The attached shock wave was observed in visualized image over the top and bottom surface. The flow remains steady from 997 µs to 10331 µs, showing the quality of flow in shock tunnel. The measured shock angle compares well with the Taylor and Maccoll theory from NASA SP-3004 table .This shows during the steady test time for all test condition, the flow over the model is steady of Mach 6.5 for which it has been designed. This ensure that flow morphology in a cone model for unsteady flow feature observed during run. The test model was above the Pitot rake by 260 mm, detached shock stand in front of the Pitot tube in a hypersonic flow. The intersection of left running shock wave emerged from Pitot tube with the right running shock wave from the model surface is clearly seen in the images (Fig. 3). The horizontal distance between nose of model and Pitot tube is 30 mm. The average value of response time of sensor for a free stream velocity of 1496 m/s and distance of 30 mm from nose to Pitot tube would be 20 µs approximately. This time response could be a one of the reason for the shock structure remained unaltered in Schlieren images for about 9 ms but steady flow measured by Pitot probe is about 3.5 ms. After the steady state of 10331 µs (Fig. 3) the stagnation pressure (settling chamber pressure) decreased and then starting shock regressed towards the nose portion. With the successful desegregation of this high speed flow diagnostic with the shock tunnel during the tunnel testing, it is now possible to characterize the hypersonic flow field around any vehicle.

The flow field features were entirely different at an angle of attack compared to 0° angle of attack. Dynamics of flow field around a cone model at α= 5° as shown in Fig. 6. At α= 5°, Schlieren images were recorded at a speed of 14035frames per second with a resolution of 384 x240 pixels, Exposure time was 62 µs, post triggering frames were 8300 and frame delay was 1 µs. A sharp cone model having a semi cone angle of 11.38°, length of 368.5 mm and base diameter of 150 mm was used for α= 0°.The same model was at 5° angle of attack, the windward and the leeward density gradient across the shock wave were 3.0 and 1.4 respectively for M =6.5 at α= 5°. The shock waves for both cases were attached to the surface. The leeward density gradient across the shock wave was too small and hence the author could not capture the shock wave pattern on leeward surface; in other words, the shock shape was not bright. This small density jump across the shock wave to make the leeward flow visualization more difficult. Not only the small density gradient make the flow visualization difficult on the leeward surface but also flaw on the lens and dust on the optical window on the test section, background effects (brightness or darkness) to Schlieren pictures and non-uniform brightness of light source could also affect the Schlieren images. In addition to that the resolution of Schlieren images to capture dispersed shock wave in the leeward surface was inadequate. To get better images, both wind -off and wind-on images were compared by Huang *et al*. (2007). The wind-off images were taken before the start of the experiment and it was reference image. The wind-on image was taken during the steady flow of shock tunnel testing. Post processing was carried out using one reference image and one experimental image.

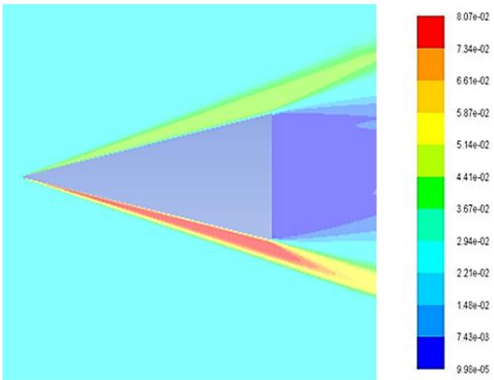
The shock angle and shock layer thickness was measured to quantify the results from high speed visualization studies. The flow field between shock wave and body is called thin shock layer (Anderson 1989). The shock angle and the shock layer thickness from Schlieren images were computed using Fiji software. By using Taylor and Maccoll theory from NASA SP-3004 table, shock angle was calculated for flow Mach number of 6.5 with cone angle of 11.38°. The measured and computed shock wave angle (β) for two different angle of attacks are summarized in Table 2. The measured shock angle from Schlieren images matches well with the theoretical value and CFD simulation. These results clearly demonstrate that Schlieren technique was more suitable for visualizing the hypersonic flow field in short duration test facilities.

Table 2 Shock wave angle (β) for two different angle of attacks

	M=6.5	M = 6.5	
		α= 5°	α= 5°
	α= 0°	Windward direction	Leeward direction
Experiment	15.2°	20.1°	Not able to capture
Theory	15.1°	20.0°	10.9°
Computation	15.1°	20.2°	10.8°



a)



b)

Fig. 7. Computational (density plot) shock wave structure in front of a sharp cone model for M =6.5 a) α= 0° b) α= 5°.

In order to validate and match the experimental results, the hypersonic flow was simulated around a cone model by using the commercial CFD package FLUENT R15. The Fluent Navier-Stoke (NS) code is capable of handling both incompressible and compressible flow problems for both steady and unsteady flow regimes. For CFD simulation SST k-omega turbulence model is used. The structural grid has been generated for the computation, with number of elements being 1.26 millions and 1.78 millions for grid dependent study. The structured mesh with finer grids near the walls is used to capture the shock pattern. The maximum y+ value obtained in the CFD simulation is 0.9. No slip condition is used as wall boundary condition. The flow properties such as static pressure of 889 N/m^2, static temperature of 132 K, and flow Mach number of 6.5, were used for the computation. Initial CFD simulation was carried with first order implicit method and later with second order implicit method to get accurate results. The attached shock wave was captured well by simulation for the both angles of attack. The computed density distribution using CFD code for sharp cone model with 11.38° apex angle at α= 0° and α= 5° at Mach 6.5 is shown in Figs. 7 (a) and (b). The shock layer thickness measured from Schlieren images and the results obtained from CFD studies for 0° angle of attack and 5° angle of attack are shown in Fig. 8. The experimental results are in good agreement with the computational results.

The difference between experimental and computational shock layer thickness is within ±1%. As seen from Fig. 8, when the angle of attack is increased from α = 0° to α = 5° about 4.1% reduction in the shock layer thickness is observed on the windward side for the same fresstream Mach number of 6.5 at 150 mm distance from the nose. Similarly at a distance of 375 mm from the nose, there is a decrease in shock layer thickness of about 4.6%. With the reduced shock layer thickness and hence with increased proximity of the attached shock wave to the sharp cone surface, the convective heat transfer for five degree angle of attack (windward surface) is higher than zero degree angle of attack for same test conditions overall increase in heat transfer could be interaction of shock wave and boundary layer which leads to increase in heat transfer along the surface. And also viscous interaction between of thick hypersonic boundary layer and outside inviscid flow is quite significant effect on heat transfer along the surface of model.

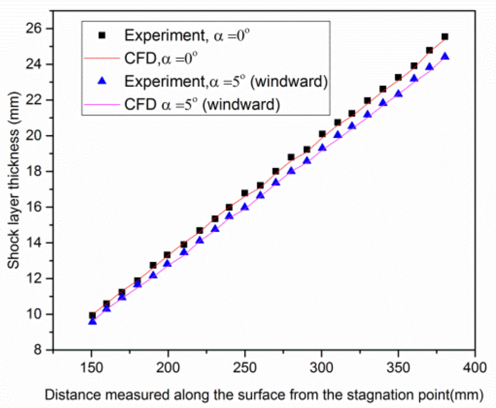


Fig. 8. Comparison of experimental and computational shock layer thickeness from the model surface for α= 0° and α= 5°.

6. CONCLUSION

A sharp cone model was tested using shock tunnel facility with the stagnation enthalpy of 1.2 MJ/kg and free stream Reynolds number based on model length of 1.06 x10^6 for both angles of attack, viz., 0°and 5°. The time dependent development of flow field around a cone model has been visualized for a flow Mach number of 6.5. The Schlieren technique is used to visualize the flow establishment, steady state and flow termination process in shock tunnel. It is observed that steady flow during the tunnel testing is about 3.5 ms but shock structure in Schlieren images remain unaltered for 9 ms. This could be due to the ratio of Pitot pressure (P_{02}) to stagnation pressure (P_{05}) which is constant over the time period. The observed shock layer thickness is lower on windward side for α = 5° compared to α = 0°. About ± 1% difference is observed in experimentally measured shock layer thickness and CFD simulation. For semi cone angle of 11.38°and as expected only attached shock is observed for both angles of attack. For α = 0°, the observed shock angle is 15.1° and for α = 5° in windward side, the measured shock angle is 20.1°.

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