Effects of Angle of Attack on Shock Wave Boundary Layer separations and Shock Wave Vortex interactions for Supersonic Flow around a Triangular Prism

Abstract: High speed aircrafts have been the holy grail of high speed flight research especially Supersonic Transport (SST) aircraft and fighters for drastically reducing the flight time. They fly at a much higher speed compared to other aircrafts by pushing through the air around it which gives rise to shock waves. Thus supersonic aircrafts are triangular in shapes to help them move through these shock waves at much higher speeds. To investigate the effects of shock boundary layer and shock vortex interactions in supersonic flow, researchers in aerodynamics communities uses triangular shaped obstacles inside the flow. Most of the researchers did not consider effects of angle of attack (AOA) in these interactions. The present study aims to fill this gap in research by considering effects of angle of attack (AOA) on shock boundary layer and shock vortex interactions. To perform this study, a triangular shaped body with an apex angle of 400 was considered and simulations were performed with ANSYS, a commercial software package for computational fluid dynamics (CFD) simulations. The approach was validated by solving the problem of Cheng et al and the results were compared. The effects of angle of attack were studied by comparing the effects from 100 to 600 AOA (with a 100 increment), with the 00 AOA to find out the variations in shock boundary layer and shock vortex interactions. Two turbulent vortices were observed at the rear section of the body which became scattered with the increase of the angle of attack. These turbulent vortices initially were found to be superimposed on each other as a single vortex up to the 100 angle of attack and became distinct as two separate entities from the 200 angle of attack. As these vortices were observed to be situated very close to each other, they were found to interact among themselves too.

Index Terms - Supersonic Transport (SST) aircraft, angle of attack (AOA), shock boundary layer, shock vortex interactions, computational fluid dynamics (CFD), turbulent vortices.

I. INTRODUCTION

Supersonic fights/aircrafts have been the holy grail of high speed flights research especially Supersonic Transport (SST) aircraft for reducing the air-travel time. Fighter aircrafts are also supersonic as they need to travel at a much higher speed to address any kind of threats. Any supersonic aircraft travels in a very high speed as compared with the subsonic aircraft. It travels through the air by pushing through the air at a much higher speed which gives rise to shock waves in the airflow. These shock waves have significant effects for the aircraft. These are shock obstacle interactions, shock boundary layer interactions and shock vortex interactions. Huge amounts of work have been done to negate these effects on supersonic aircrafts. As all modern fighter aircrafts are essentially supersonic ones so they are designed in such a way which decreases these effects on them. The shock obstacle, shock boundary layer interactions and shock vortex interactions are unavoidable in high speed flights. Shock obstacle interactions, shock boundary layer interactions and shock vortex interactions [10] involving complex geometries in high speed compressible flows [44, 45, and 46] are an active field of research in fundamental sciences and engineering applications. The interaction of shocks with solid obstacles can occur in various applications like supersonic transition in flight, hypervelocity impact and penetration. These phenomena have received significant attention in the past [37, 38, 39, 40, 43] and remain as an active field of research and development [36]. The flow physics for shock obstacle, shock boundary layer and shock vortex interactions is so complicated that it still requires a lot of detailed research to get the complete and detailed overview [48, 49].

If an external body is inside a high speed compressible flow domain there will be shock obstacle interactions [3]. There will also be boundary layer interactions for flows near boundaries and shock vortex interactions too. These will depend on the shape/geometry and the angle of attack of the external body. For a bluff shaped external body [26] there will be detached shock waves unlike a sharp edged body where there are attached shock waves and these will be different for different angles of attack. The separation of boundary layers will also be different for different angles of attack. There will also be some shock waves which will be reflected after hitting the object; these are known as oblique shocks [13, 16, and 17]. This problem represents one of the simplest occurrences of strong shock/obstacle interactions and therefore an ideal test case for Computational Fluid Dynamics (CFD) [15, 25, 27, and 28]. The present research study is solely focused on the supersonic flow domain and its flow characteristics for shock boundary layer (SBLI) and shock vortex interactions related to the problem of an external triangular shaped body within the flow [58, 59]. Turbulence as such is unavoidable in any kind of high speed compressible flow study howsoever small it might be. According to the father of high speed aerodynamics Theodore von Karman, “Turbulence was, and still is, one of the great unsolved mysteries of science and it intrigued some of the best scientific minds of the day. Arnold Sommerfeld, the noted German theoretical physicist of the 1920’s once told me, for instance, that before he died he would like to understand two phenomena-quantum mechanics and turbulence. Sommerfeld died in 1924. I believe he was somewhat nearer to an understanding of the quantum, the discovery that led to modern physics, but no closer to the meaning of turbulence” circa 1967. As this study considers a supersonic inflow of Mach number 3.5, turbulence has a profound effect on the flow properties. It is because of the presence of turbulence that turbulent vortices and the phenomenon of vortex shedding i.e. Von Karman vortices were observed at the rear side of the body.

High speed aircrafts have been a very active research area the world over. It progressed along with the progression of the knowledge of high speed compressible flows. Germany and USA were the two countries leading the research in this particular field during the early stage
of this research. Ludwig Prandtl considered to be the father of modern aerodynamics, from University of Gottingen Germany and his students were the leaders in this field till the 1940’s. His pedagogical concept of the boundary layer metamorphosed fluid mechanics research as well as aerodynamics research too. The Second World War changed the scenario and more so after the surrender of Germany to the allies. As Germany was split into two parts, East Germany under the Soviets and West Germany under the allies in a way mostly under the control of United States and the UK, a lot of his students went abroad either to the US or to erstwhile USSR. Prandtl himself supervised the construction of the first ever supersonic wind tunnel in University of Gottingen, Germany.

Prandtl’s student Theodore von Karman [60] played a significant role in developing high speed compressible flow research in the United States. In the US NACA the precursor organization of NASA gave a lot of impetus to the high speed aircraft research. Their X plane program contributed a lot to the high speed flight research data. The program is active even today to develop further the present knowledge base for supersonic and hypersonic flights. This was influenced by the invention of the jet engine in pre-World War 2 era in the UK by Sir. Frank Whittle. This fine invention was chosen as the power plant for high speed aircrafts but after a lot of trial and errors. After the World War 2 was over there was a sudden increase in jet powered aircrafts as direct fallout of the German experience in World War 2. All of these things continued in the 1950’s also. The UK in the early 1950’s had initiated works on a Supersonic Transport (SST) aircraft. The only stumbling block was the high cost of development.

In Europe, France was another country where a lot of high speed aviation related research was ongoing. The setting up of Office National d'Etudes et de Recherches Aérospatiales (ONERA) in 1946 heralded the start of a new age in French aviation research post World War 2. They also did some fundamental research in high speed aircrafts which lead to the development of world beating supersonic fighter aircrafts. In the late 1950’s French government asked three aero companies namely Sud Aviation, Nord Aviation and Dassault Aviation for the design of a SST. Among the three designs Sud Aviation’s design was finally selected by the government. They started independently unaware of the British development but since the French had no such heavier jet engine it was decided to purchase a heavier jet engine capable of producing the required thrust for the supersonic flight. People from Sud Aviation came to the UK for discussing the engine but they found that the UK had by then developed the design of a SST very similar to the French one. Two organizations were chosen British Aircraft Corporation (BAC) from UK and Aerospatiale from France. The Concorde had a cruising speed of Mach 2.02 (1.320mph) helping it to travel from Los Angeles to Tokyo in around 3.5hrs flat compared to 7-8hrs taken by conventional subsonic commercial airliners. The Concorde had its first flight in 1969 followed by commercial operations from 1976 onwards till 2003 when they were retired by both British Airways and Air France.

The Tupolev Tu-144 grew out as a Soviet ego of outclassing the west for technological dominance, which is shown in Figure 1.1. Designed by the Tupolev design bureau as a pet project of Alexi Tupolev, the head of the bureau and financially supported by the Soviet Government. The Tu-144 had a maximum cruise speed of Mach 2.4 and was able to fly at a greater altitude. It had a brief commercial life and success unlike the Concorde which had commercial operations of 26 years.

![Figure 1.1 A Tupolev Tu-144 of Aeroflot in flight [62]](image1)

Nevertheless both of these retired SST aircrafts showcased the advantage of supersonic travel i.e. drastic reductions of the travel time. Of particular mention is the US spy plane SR-71 which was developed in the early 1960’s. This particular supersonic spy plane could cruise at a speed of Mach 3.2. Another US aircraft that was faster compared to it in terms of speed was the X-15 which had a cruising speed of Mach 4 and the Soviet Mig-25 Foxbat (see Figure 1.2) but the Soviet plane could achieve a speed over Mach 3 for few minutes only.

![Figure 1.2 The Soviet Mig-25 Foxbat [61]](image2)

The legacy of SR-71 (the Blackbird), refer in Figure 1.3, continues even today and an updated one the SR-72 has been proposed by Lockheed Martin Aerospace. It entered service in the year 1968 and remained active till 2002. Though the X-15 was able to fly at a speed greater than Mach 5, it was an experimental aircraft operated by NASA and USAF. The speed achieved by X-15 is still the highest ever achieved by any manned aircraft as of January 2017.
A renewed impetus have been observed once again in SST aircrafts with Lockheed Martin Aerospace, Aerion Corporation, Boom Technology and Boeing from the US, EADS along with Japanese Space Exploration Agency (JAXA) have proposed new designs for the next generation SST’s. All these proposed aircrafts are in their design stage with JAXA completing the wind tunnel study of their new age SST capable of carrying thrice the passenger load of Concorde (see Figure 1.4) at a quarter ticket price of it. Aerion corporations proposed business jet a smaller SST capable of carrying 12-14 passengers have progressed well and they are currently finalizing the manufacturing breakup of their work among the various stakeholders.

1.1 Introduction

Shock waves or blast waves can harm individuals who are adjacent to it since they have a lot of energy. In early decades, the weakening of shock waves was given more importance because of its significance in numerous handy applications. Many ways to deal with modifying the powers of such waves have been developed. Shock waves obstacle interactions, shock waves boundary layer interactions and shock waves turbulent vortex interactions are essential fields of research in major sciences and engineering as well as for designing applications. For example, the interactions of shock waves with solid obstacles can occur in a few applications incorporating high-speed flows, hypervelocity effect and penetration. These wonderful flow phenomenon have received critical considerations earlier also and still stay as an active field of research. The fundamental essential requirement is to use the properties of these shock waves and obstacle interactions positively, either by centering or weakening the shock wave effects. From the basic point of view a set-up is required for avoiding the catastrophic effects of shock obstacle interactions for complex geometries. These types of interactions, includes various structures of shock waves interacting with obstacles of different geometries like cones, circular cylinders, spheres etc. These particular problems of shock boundary layer and shock vortex interactions, is one of the most fundamental problems of shock obstacle interactions and hence a perfect experiment for Computational Fluid Dynamics (CFD) solvers [29, 32, 35, 65, 67, 68 and 69].

1.2 Experimental work

Simpson et al. [19] investigated the features of two-dimensional and three-dimensional separating turbulent boundary layer flows. They investigated the structure as well as the nature of strong adverse-pressure-gradient separating flows over streamlined surfaces and backward-facing step separations, for the two-dimensional cases. A number of differences from attached flows on normal stress effects in the Reynolds-averaged momentum and turbulence kinetic energy equations were observed. To produce the mean velocity profile, low Reynolds shearing stresses, and the turbulence energy diffusion toward the wall, the backflow dominated by large-scale unsteady motions, which balanced by dissipation and to model these flows, the non-equilibrium turbulent structure required stress transport equation(s). They also discussed about various flow structure features about a 6:1 prolate spheroid at angle of attack and also for a wing/body junction.

Dussauge et al. [21] explained spectral measurements in zero pressure gradient boundary layers, which validated the fact that the longitudinal scales of the energy-containing motions decrease significantly with Mach number. For instant, they noticed at Mach number 2.9, the scales decreased around a factor of two as compared with subsonic flows. That solved by the agreement with spatial correlation data obtained by using Rayleigh scattering. They inferred that these modifications in the scales were due to compressibility effects. They also hypothesized an explanation for the generation of acoustic noise by supersonic boundary layers.

Chang et al. [23] used the analytical and experimental investigations into the phenomenon of flow separations around a triangular obstacle. A summary of flow separation is given for understanding the basic problem, the present knowledge state and for indicating the future development. This covers the complete coverage of this phenomenon i.e. in laminar, turbulent, incompressible and compressible flow domain. Flow separation is one of the most unsolved phenomenon’s of fluid mechanics which leads to energy loss and deviation of stream lines. Chang et al. used an equilateral triangle of base length 20 m, as a finite wage and it is illustrated in Figure 2.1.
Figure 2.1 An experimental setup for shock obstacle interactions in a shock tube [23]

Figure 2.2 shows the experimental setup for Holographic interferometry system used by Chang et. al [23]. In this setup BS, FH, FLS, L, M, SM, TS, and W are for the Beam Splitter, Film Holder, Focussing Lens System, Lens, Mirror, Schlieren Mirror, Test Section and Window respectively. This experimental setup consists of a driving chamber of length 2 m and a driven section of length 8 m. A ruby laser was used with maximum beam energy of 1 J as the light source, Agfa 10E75 films were used for photography and a 25-mW He-Ne laser for reconstruction of the holograms.

Figure 2.2 Holographic interferometry systems [23]

Dupont et al. [34] investigated the organization in a shock wave boundary layer interaction. They focused their attention on the range of frequencies and the space scales present in the flow, along with the beginning of a three-dimensional structure in the separated flow. For achieving their objectives, experimental investigations were performed on an oblique shock wave impinging on a turbulent boundary layer. They observed that the induced pressure gradient on the boundary layer was strong enough to separate it locally, with an unsteady reflected shock. They also investigated space, time properties of the unsteadiness appearing in this flow in different flow zones, creating a problem with multiple time scales, and a three-dimensional separated zone.

In Figure 2.3 the setup of experiments and coordinate system and in Figure 2.4 the position of the camera, used by Dupont et al. [34], are shown. They used a laser sheet, a shock generator to develop the experiment.
Numerical Modelling

2.1 The physical problem considered

Figure 3.1 depicts a triangular prism shaped external body kept in a high speed compressible flow domain.

![Figure 3.1: Schematic of the Physical Model (2D case)](image)

Table 3.1: Dimensions of the Computational domain

<table>
<thead>
<tr>
<th>b (Base of the triangular body)</th>
<th>h (Height of the triangular body)</th>
<th>Length (L_x)</th>
<th>Breadth (L_y)</th>
</tr>
</thead>
<tbody>
<tr>
<td>8</td>
<td>11</td>
<td>120</td>
<td>30</td>
</tr>
</tbody>
</table>

2.2 Governing Equations

2.2.1 Governing equations of fluid flow

The Unsteady Compressible Reynolds-Averaged Navier-Stokes Equations (URANS) [8, 12, 47] can be written as Equation 3.1, Equation 3.2 and Equation 3.3:

\[
\frac{\partial \rho}{\partial t} + \frac{\partial \rho u_j}{\partial x_j} = 0 \quad (3.1)
\]

\[
\frac{\partial (\rho u_j)}{\partial t} + \frac{\partial (\rho u_j u_i)}{\partial x_i} = \frac{\partial p}{\partial x_j} + \frac{\partial \tau_{ij}}{\partial x_j} \quad (3.2)
\]

\[
\frac{\partial (\rho E)}{\partial t} + \frac{\partial (\rho u_j u_i)}{\partial x_i} = \frac{\partial p}{\partial x_j} + \frac{\partial \tau_{ij}}{\partial x_j} + \frac{\partial}{\partial x_j} \left( \rho \beta_i \nabla \theta \right) - \frac{\partial}{\partial x_j} \left( \rho \beta_i \nabla \theta \right) \quad (3.3)
\]

Where, Equation 3.4, Equation 3.5 and Equation 3.6 are

\[
\dot{H} = \dot{E} + \frac{\tilde{P}}{P} \quad (3.4)
\]

\[
\tilde{q}_j = -k \frac{\partial T}{\partial x_j} \approx -c_p \frac{\dot{\mu} \tilde{T}}{P} \quad (3.5)
\]

and the viscous stress tensor is:

\[
\dot{\sigma}_{ij} \approx 2 \dot{\mu} \left( \dot{S}_{ij} - \frac{1}{3} \frac{\partial \tilde{u}_k}{\partial x_k} \delta_{ij} \right) \quad (3.6)
\]
The Reynolds stress term \( \tau_{ij} \equiv -\rho \overline{u_i' u_j'} \) is defined in the literature both ways, as shown here as well as with the opposite sign and sometimes without the density included in the definition. (This different terminology does not matter as long as consistency is maintained throughout the derivation.) The term \( C_P \) is the heat capacity at constant pressure and \( Pr \) is the Prandtl Number (e.g. around 0.72 for air). The overbar written here indicates conventional time-average mean, with the averaging time scale assumed to be long compared to turbulent fluctuations and short compared to unsteadiness in the flow. The hat here represents the Favre (density-weighted) average:

\[
\hat{f} = \frac{\rho \bar{f}}{\rho}
\]  

(3.7)

Also \( f = \bar{f} + f' = \bar{f} + f'^* \), where the over bar indicates conventional time-averaged mean, with the averaging time scale assumed to be long compared to turbulent fluctuations, and short compared to unsteadiness in the mean flow.

For turbulence closure the two equation \( k - \varepsilon \) eddy viscosity model with wall-function boundary conditions were used. A second order implicit Roe-Finite Difference Scheme was employed. For a detailed overview one should refer to the ANSYS Fluent user’s guide [71] and ANSYS Fluent theory guide [72] for release 15.0.

### 2.3 Boundary and initial conditions

A supersonic inflow of Mach number 3.5 is considered as the inflow and the pressure boundary condition for the outlet of the computational domain, as guidelines for the compressible flow. The details of the boundary conditions are shown in Table 3.2. The fluid inside the computational domain was air. The inflow of Mach number 3.5 corresponds to the inlet velocity of 1191.02 m/s. The temperature throughout the computational domain is maintained at 300 K. As interactions in the boundary layers were to be investigated so, no slip boundary conditions were considered at the two walls (upper and the lower one).

#### Table 3.2: Boundary Conditions

<table>
<thead>
<tr>
<th>Inlet Velocity (m/s)</th>
<th>Outlet Pressure (Pa)</th>
<th>Angle of Attack (°)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1191.02</td>
<td>101325</td>
<td>0</td>
</tr>
<tr>
<td>1191.02</td>
<td>101325</td>
<td>10</td>
</tr>
<tr>
<td>1191.02</td>
<td>101325</td>
<td>20</td>
</tr>
<tr>
<td>1191.02</td>
<td>101325</td>
<td>30</td>
</tr>
<tr>
<td>1191.02</td>
<td>101325</td>
<td>40</td>
</tr>
<tr>
<td>1191.02</td>
<td>101325</td>
<td>50</td>
</tr>
<tr>
<td>1191.02</td>
<td>101325</td>
<td>60</td>
</tr>
</tbody>
</table>

#### Table 4.1: Different values of angle of attack (AOA)

<table>
<thead>
<tr>
<th>Cases</th>
<th>AOA (°)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>0</td>
</tr>
<tr>
<td>2</td>
<td>10</td>
</tr>
<tr>
<td>3</td>
<td>20</td>
</tr>
<tr>
<td>4</td>
<td>30</td>
</tr>
<tr>
<td>5</td>
<td>40</td>
</tr>
<tr>
<td>6</td>
<td>50</td>
</tr>
<tr>
<td>7</td>
<td>60</td>
</tr>
</tbody>
</table>
3 Validation

The axial velocity gives an insight of the flow behaviour that takes place in the computational domain. The axial velocity paints the flow features that may be responsible for any changes that hinder the flow path. The present study is validated by considering the previous work of Chang et.al [30]. The parameters considered are axial velocity, density and Mach numbers. These three parameters are the most widely studied ones for supersonic compressible flow past an obstacle and they give first-hand knowledge of the flow physics.

The present work has been compared with Chang et.al [30] as shown in Figure 3.3 and the variation of axial velocity along the flow direction is compared across the upper window of the computational domain. The present study starts with the same trend as that of Chang et.al [30].

![Figure 3.3 Axial velocity distribution along the upper boundary](image)

However as the flow experiences hindrance for the boundary layer growth in the upper boundary, the axial velocity increases till it reaches 350m/s after which it decreases suddenly as that of Chang et.al. However the sudden decrement is at a higher value than that of Chang et.al and drops to 290 m/s compared to 300 m/s of Chang et.al. This point to the shock-obstacle interaction, that takes place in supersonic flow. After this, there is a sudden jump as that of Chang et.al which bears a close resemblance, as it decreases similarly but on a lower value, which continues till the end. Overall the shock-obstacle interaction hampers the boundary layer growth which is verified by the plot.

The variation of density is studied in supersonic flow to observe the flow behaviour. The Figure 3.4 shows the variation of density across the axial direction in the trailing position of the obstacle.

![Figure 3.4 Flow behind the wedge, density distribution from the side perpendicular to the wedge base](image)

The trailing position is chosen for the reason of maximum changes as the flow is intercepted by shock wave interactions and formation of vortices. The present study follows the same trend as that of Chang et.al [30] from the beginning, but slowly moves away due to the flow interactions with the obstacle boundary and shock interactions. Overall it moves away from that of Chang et.al [30] on increasing the distance from the trailing edge.

In Figure 3.5 the Mach number refers to the compressibility effect that is a key feature of compressible flow. The change in Mach number over the trailing edge of the obstacle/wedge is compared with that of Chang et.al [30]. The change observed for the present study follows the same trend starting from the beginning of the trailing edge, but slowly increases sharply at a distance of 40mm till 60mm. After that it drops down to 70mm and increases again to 75mm and then decreases sharply as of Chang et.al [30].
4 Results and Discussion

To study the effect of various angles of attack on the flow properties in the supersonic flow domain, a triangular body was considered. Any supersonic aircraft whether it’s a fighter or a commercial airliner have a triangular shape to move at supersonic speeds. This is so, because for an aircraft to move efficiently through a supersonic flow this particular shape helps it to pass through the shock waves. Researchers in aerodynamics communities also use triangular shaped bodies to investigate various flow characteristics in supersonic flow domain. In our study a Mach 3.5 supersonic flow was considered as the inflow for investigating the effect of various angles of attack on the shock boundary layer and shock vortex interactions. In this study we have varied the angle of attack from 10° to 60° with an increment of 10° among them. We have also considered no slip boundary conditions at the two computational boundaries, upper and lower walls for investigating the effects of angle of attack on shock boundary layer interactions in the two walls. These would give a very clear picture about the shock boundary layer interactions.

4.1 Grid resolution study

To find out the appropriate mesh for the computational domain, grid resolution/independence study was carried out. Based on this grid resolution/independence study, the best mesh was considered for our present study. This was done for accurate simulation of the effects of angle of attack on shock boundary layer and shock vortex interactions. The grid resolution study for the fluid has three types of meshes as shown below in Table 4.2. The table below shows the three types of meshes used to study the change of density at the rear side of the obstacle.

Unstructured meshing has gained popularity due to the advancements in the advection scheme accuracy and the more intense computational stages accessible. Unstructured meshes come in numerous assortments, with the most widely recognized being four-sided tetrahedrons joined with five-sided prisms for boundary layer determination. The unstructured work approach has favourable circumstances for the profoundly complex geometries found in auxiliary stream sections. An ideal opportunity to create a quality unstructured mesh is fundamentally less.

<table>
<thead>
<tr>
<th>Types of meshes</th>
<th>Number of elements</th>
</tr>
</thead>
<tbody>
<tr>
<td>Coarse</td>
<td>235,475</td>
</tr>
<tr>
<td>Medium</td>
<td>312,865</td>
</tr>
<tr>
<td>Fine</td>
<td>359,198</td>
</tr>
</tbody>
</table>

In the Figure 4.1 below, the y axis denotes the location of the turbulent vortex along the flow direction in mm at the rear side of the body and the angle of attack is given in °. The results show that in the mesh containing 235,475 elements, there is an abrupt change in the position of the vortex. The mesh containing 312,865 elements and that having 359,198 elements shows almost the same change in position of the vortex. However for better results the finer mesh of 359,198 elements is chosen for the present study.
4.3 Effects of Angle of attack on Boundary-Layer Separations

It has been found out from these simulations that, as the angle of attack varies the boundary layer separation also changes.

i. For 0° angle of attack

In the first simulation when the triangular body was at 0° angle of attack, four prominent boundary layer separations were observed. Two such were observed in the upper wall and two in the lower wall. A particular characteristic have been observed for these boundary layer separations. The separation which is observed in the upper wall is symmetric to the one observed for the lower wall also. This refers to the nature of the boundary layer separations with respect to their length and breadth. It has been observed that the boundary layer separations near the rear side of the body are much smaller in size i.e. length and breadth as compared to those observed away from the rear side of the body. This is due to the reason that, far away from the rear side of the body more shock boundary layer interactions takes place. This is due to the successive shock interactions with the body, since the flow keeps on impinging on the body. This leads to much more shock boundary layer interactions thereby giving rise to bigger boundary layer separations. In this angle of attack the triangular body is placed in such a position that, the nose of the body is equally spaced from the upper and the lower wall. In this particular position, the incoming supersonic flow impinges with the body head on and an oblique shock is observed which interacts with the boundary layer near the rear side of the body. This occurs for the upper and the lower wall also. As such similar type of boundary layer separations are observed near both these walls. As the incoming flow keeps on impinging on the body which gives rise to multiple shock obstacle interactions thereby giving rise to multiple shock boundary layer interactions. These are more dominant away from the triangular body where there are more variations in total pressure and which leads to more boundary layer separations. So the boundary layer separations observed far away from the rear side of the body are quite bigger as compared to those near the rear side. This is observed in the contour plot of the velocity magnitude in figure 4.2 (a).

ii. For 10° angle of attack

It has been observed that as the angle of attack becomes 10° the boundary layer separations undergoes a change in their characteristics. The boundary layer separation at the upper wall becomes insignificant as compared to that of the lower wall. At the upper wall two such boundary layer separations are observed. The one that is near to the rear side of the body is quite small in size. This separation is due to the interactions of the oblique shock with the boundary layer present near the wall. The boundary layer separation which is observed in the upper wall away from the rear side of the body is somewhat significant and can be identified. This is due to the shock boundary layer interactions occurring near the wall. The boundary layer separations observed at lower wall are quite bigger as compared to the upper walls. The boundary layer separation near the rear side of the body is comparatively much smaller than that observed near the outlet. The boundary layer
iii. For 20° angle of attack

For the triangular body as the angle of attack becomes 20°, the nose becomes more aligned towards the upper wall. So the incoming flow impinges with that side of the body which comes near to the flow. This leads to shock obstacle interactions thereby giving rise to multiple shock boundary layer interactions. These multiple shock boundary layer interactions give rise to four such boundary layer separations. The two boundary layer separations observed in the upper wall are more distinguishable as compared with the two boundary layer separations in the lower wall. These are due to the multiple shock boundary layer interactions as a consequence of the multiple shock reflections from the two walls. As these interactions are not strong enough, so the boundary layer separations are also not significant. This is found out in the contour of velocity magnitude given in figure 4.2(c). As compared with the 0° angle of attack, here also four such boundary layer separations were observed. But in this case only two boundary layer separations observed at the upper wall are significant, the other two at the lower wall are not that significant enough.

iv. For 30° angle of attack

As the angle of attack becomes 30°, the nose of the triangular body becomes more aligned towards the upper wall. Thus the incoming supersonic flow impinges with the side which comes directly in the path of the incoming flow reducing their velocity significantly. This leads to weaker shock obstacle interactions, thereby much smaller shock boundary layer separations. Six such boundary layer separations were observed which were identifiable. Out of these the one which is observed near the upper wall and which is near to the rear side of the body was easily identifiable. The one near the upper wall which was near to the outlet was comparatively much less diffused. Near the lower wall three such boundary layer separations are observed. The separation observed near the rear side of the body is not that diffused and the other one near it is smaller in size lengthwise. The other boundary layer separation which is near to the outlet is comparatively much longer and somewhat diffused. These are due to the successive shock obstacle interactions which forms successive oblique shocks which results in boundary layer interactions occurring in the lower wall. These gives rise to boundary layer separations at three places near the lower wall. The separation near to the rear side of the body is not very big but a significant one. The other separation which is near to the first one is little smaller in comparison but is an identifiable one. The boundary layer separation which is near to the outlet is a quite significant one and much bigger compared to the others. This is due to the reason that successive shock boundary layer interactions leads to this separation. There were more shock boundary layer interactions (SBLI) at that position in the wall which gave rise to a bigger separation. This is so because there were not much boundary layer interactions in the upper wall. Three such boundary layer separations were observed at the upper wall, two of which were near to the rear side of the body. The boundary layer separation that is observed nearer to the rear side of the body is a much diffused one and spread over a longer distance. The boundary layer separation that is little away from it is smaller than the one preceding it but still identifiable. The separation that is away from the rear side and which is near to the outflow is not significant but an identifiable one. This so because there was not much shock boundary layer interactions (SBLI) in the wall near the outflow. Hence the boundary layer separation was quite smaller. This is seen in the contour plot of the velocity magnitude in figure 4.2(d). So for this angle of attack as compared with the 0° case, six such boundary layer
separations were observed. Three such separations were observed in the upper wall and three in the lower wall. But unlike the 0° case not all were significant.

v. For 40° angle of attack

As the angle of attack becomes 40°, the nose of the triangular body becomes more aligned towards the upper wall. In a way the nose of the body is positioned in such a way which obstructs the incoming flow. This has some effect on the incoming flow, reducing the incoming flow velocity near the nose of the body. This is evident from the contour plot of the velocity magnitude given in figure 4.2(e). Incoming flows which impinge on that side of the body that faces the flow, gives rise to shock obstacle interactions. It is these shocks which interact with the upper wall, giving rise to shock boundary layer interactions. These successive shock boundary layer interactions in the upper wall, gives rise to the boundary layer separation in this wall at a place which is near to the rear side of the body. Another separation was also observed near the outlet in the upper wall which was not significant. This was also due to shock boundary layer interactions there, but since the interactions were not strong enough boundary layer separations was not that significant. It was quite diffused one as evident from the earlier figure. In the lower wall near to the outflow another boundary layer separation was observed too. As there were less shock boundary interactions so the separation was not significant. The separation was quite diffused one with and was spread over a longer space. Thus for this angle of attack as compared with the 0° angle of attack three such boundary layer separations were observed but only one was significant (prominent).

vi. For 50° angle of attack

Now as the angle of attack increases further to 50°, the nose of the body aligns itself more towards the upper wall. Thus again the incoming flow is obstructed, thereby the incoming flow velocity is reduced. Still shock obstacle interactions take place, which leads to shock boundary layer interactions in the upper wall. This leads to the boundary layer separation in the wall near to the rear side of the body. This occurs since as the nose of the body is so aligned, thereby successive shock boundary layer interactions occurs for the upper wall but comparatively less than the previous angle of attack. Thus the boundary layer separation in the upper wall is not that much as compared with the earlier case. The separation is quite significant as observed in the contour plot of the velocity magnitude in figure 4.2(f). Another boundary layer separation was observed in the upper wall near to the outflow. As there were not much shock boundary layer interactions so the boundary separation was not that significant. In the lower wall one particular boundary layer separation was observed. Although that was not as significant compared with that observed in the upper wall near to the rear side, but was identifiable. This was due to shock boundary layer interactions, as not much shock boundary layer interactions were observed there the boundary layer separation was not very significant. It was a diffused one and had longer length. Hence as compared with the 0° angle of attack three boundary layer separations were observed, but among them only one was significant and easily identifiable. This was due to the change of the orientation of the body.

vii. For 60° angle of attack

As the angle of attack becomes 60°, the nose of the triangular body becomes aligned more towards the upper wall. This makes the particular side of the body face the incoming supersonic flow, thereby reducing the flow velocity also. This is observed in the contour plot of the velocity magnitude given in figure 4.2(g). Even at this position of the body there are shock obstacle interactions. These successive shock obstacle interactions, gives rise to boundary layer separations in the upper wall near to the rear side of the body. It has a cumulative effect, with every interaction there is some separation and this continues for every interactions. This separation is quite significant and identifiable too. As there are more shock boundary interactions, so the boundary layer separation is comparatively more than the previous case. This is also evident.
from the contour plot of the velocity magnitude in the figure. Another boundary layer separation was observed in the lower wall. Though it was not significant like the one in the upper wall but was identifiable. This was due to successive shock boundary layer interactions which take place there. As these interactions were much lesser and weaker, than those which were observed in the upper wall, so the boundary layer separation was quite small in size. The separation was observed near the outflow and was spread over a longer distance along the lower wall. Thus for this angle of attack, compared with the 0°, three such boundary layer interactions were observed. Of these the one observed in the upper wall near to the rear side of the body was the most significant.

![Figure 4.2](image)

**Figure 4.2** Contours of Velocity magnitude (a) for 0° angle of attack, (b) for 10° angle of attack, (c) for 20° angle of attack, (d) for 30° angle of attack, (e) for 40° angle of attack, (f) for 50° angle of attack and (g) for 60° angle of attack
Figure 4.2  Contours of Velocity magnitude (a) for 0° angle of attack, (b) for 10° angle of attack, (c) for 20° angle of attack, (d) for 30° angle of attack, (e) for 40° angle of attack, (f) for 50° angle of attack and (g) for 60° angle of attack (continued)
Figure 4.2  Contours of Velocity magnitude (a) for 0° angle of attack, (b) for 10° angle of attack, (c) for 20° angle of attack, (d) for 30° angle of attack, (e) for 40° angle of attack, (f) for 50° angle of attack and (g) for 60° angle of attack (continued)

Figure 4.3 shows contours of Mach no. (a) for 0° angle of attack, (b) for 10° angle of attack, (c) for 20° angle of attack, (d) for 30° angle of attack, (e) for 40° angle of attack, (f) for 50° angle of attack and (g) for 60° angle of attack

Figure 4.3  Contours of Mach no. (a) for 0° angle of attack, (b) for 10° angle of attack, (c) for 20° angle of attack, (d) for 30° angle of attack, (e) for 40° angle of attack, (f) for 50° angle of attack and (g) for 60° angle of attack
Figure 4.3  Contours of Mach no. (a) for 0° angle of attack, (b) for 10° angle of attack, (c) for 20° angle of attack, (d) for 30° angle of attack, (e) for 40° angle of attack, (f) for 50° angle of attack and (g) for 60° angle of attack (continued)
4.4 Effects of Angle of attack on Shock Vortex Interactions

Two turbulent vortices were observed at the rear side of the body even though the incoming supersonic flow had a Mach number of 3.5. As the Mach number of the flow is higher, so the vortices were scattered. The position and direction of the vortices varies with the variation of the angles of attack. Another very interesting characteristic that is observed for these two vortices is that they were very close to each other. So there are interactions between these two vortices as such.

i. For 0° angle of attack

As the inflow is considered to be a supersonic one with a Mach number of 3.5 there are bound to be turbulence effects inside the flow domain. In this study as the solid triangular prism is placed near to the inflow so the high speed incoming flow impinges on the body head-on leading to flow irregularities. These irregularities in the flow give rise to turbulence inside the flow domain. It has been observed in the simulations that turbulence has more effect in the rear side of the body. The turbulence effects in the rear side of the triangular body are compounded by the multiple shock reflections occurring from the two walls. These two effects as such give rise to turbulent vortices near the rear side of the body. One such particular and strong turbulent vortex was observed. At the 0° angle of attack the vortex was situated near to the rear side of the body, but in a way which was parallel to the flow direction.
ii. For 10° angle of attack

As the angle of attack becomes 10° the vortex changes its position and direction in a way which is not parallel to the flow direction anymore. In a way the vortex becomes aligned in a direction which is not in line with the angle of attack. The angle of attack is increased in the counterclockwise sense but the direction of the vortex changes in the clockwise sense. This is due to the fact that as the angle of attack changes i.e. increases, the turbulent vortex which was initially observed to be in the same direction as the flow changes its position/direction too. It aligns itself in such a way which is opposite to the direction of movement of the triangular body. This is due to the successive shock waves which interact with the vortex structure. This can be seen clearly from the contour plot of the turbulent kinetic energy for the 10° angle of attack given in figure 4.4(b). So as compared to the earlier case of 0° angle of attack, the vortex undergoes a change in position and direction.

iii. For 20° angle of attack

From the 20° angle of attack another interesting characteristics in the vortex structure has been observed. Up to the 10° angle of attack, a single strong vortex structure was observed. As soon as the angle of attack becomes 20°, two particular vortices were observed for the first time and these two were somewhat parallel to the flow direction. It appears that the vortices were superimposed over one another and as such one strong vortex was observed. Also at this angle of attack another interesting characteristics of these vortices were observed. The upper vortex had more energy than the lower vortex. This is so because as the upper vortex was somewhat nearer to the upper wall, so the reflected shock wave from the upper boundary interacts with it. As the incoming flow is a continuous one, so these shock wave boundary layer interactions are also continuous ones. These kinds of successive shock wave boundary layer interactions in the upper wall, consecutively gives rise to continuous shock vortex interactions. These continuous impingements on the turbulent vortex, leads to increment in the energy of the said vortex itself. This can be seen clearly from the contour plot of the turbulent kinetic energy for the 20° angle of attack given in figure 4.4(c). Thus for this angle of attack, two such turbulent vortices were observed as compared with the 0° case where only one strong vortex was observed. Also among these vortices, the upper vortex was observed to be more energetic as compared to the lower one.

iv. For 30° angle of attack

As the angle of attack becomes 30° another interesting characteristics is observed, for these vortices. The upper vortex which was observed to be more energetic at the 20° angles of attack, losses much of its energy significantly. This is due to the fact that as the angle of attack increases to 30°, the reflected shock waves from the upper wall interact with the vortex as it is nearer to that wall. These interactions lead to the reduction of the energy of the said vortex as evident from the contour plot of the turbulent kinetic energy in figure 4.4(d). So compared with the 0° angle of attack, two vortices were observed. Among these two vortices the lower one was found to be more energetic.

v. For 40° angle of attack

As the angle of attack increases to 40°, the upper vortex gains energy and this is evident from the contour of the kinetic energy as depicted in figure 4.6(e). In this particular angle of attack there are comparatively much more shock boundary layer interactions near the upper wall. So these leads to more shock wave vortex interactions and consecutive increment in the energy for the vortex, which was near to the upper wall. Another interesting characteristic of these vortices, were also observed. The lower vortex which was little far from the upper wall was more energetic than the upper one. This can be observed from the contour plot of turbulent kinetic energy in figure 4.4(e). There are multiple shock obstacle interactions which give rise to multiple shock reflections from the body and the walls. This is because the supersonic inflow keeps on impinging on the body. It is because of these multiple shock boundary layer interactions which lead to the boundary layer
interactions. It is these successive boundary layer interactions/reflections, which gives rise to increment in the energy of the upper vortex. Hence compared with the 0° angle of attack two particular turbulent vortices were observed and they had differences in their energy profile.

vi. For 50° angle of attack

Now when the angle of attack becomes 50°, the nose of the body becomes oriented more towards the upper wall. So the incoming flow impinges directly on that side of the body which is nearer to the incoming flow. The supersonic inflow at first interacts with the body which gives rise to shock obstacle interactions, these shocks are also reflected from the two walls. It is these reflected shocks which complicate the flow as such and gives rise to turbulent vortices. The particular position of the body at this angle gives rise to more shock obstacle and shock boundary layer interactions near the upper wall. These gives rise to more shock wave turbulent vortex interactions. These interactions, ultimately gives rise to the increase in the energy of the upper vortex as shown in figure 4.4(f). So for this angle of attack, compared with the 0°, two turbulent vortices were observed. The upper vortex was found to be more energetic than the lower one.

vii. For 60° angle of attack

Now again as the angle increases to 60°, the position of the body changes also. The nose of the body becomes aligned more towards the upper wall. The new position of the body has a huge impact on the flow properties. So now much of the incoming flow impinges directly on that side which is near to the inflow. These gives rise to shock obstacle interactions, which leads to shock boundary layer interactions and reflections. As the flow is continuous so successive shock obstacle interactions gives rise to successive shock boundary layer interactions and reflections. These successive shock boundary layer reflections gives rise to successive shock turbulent vortices interactions. This ultimately leads to the change of position of the vortex which was near to the upper wall. The vortex becomes more scattered as a consequence of multiple shock reflections from the upper wall. This can be seen clearly from the contour plot of the turbulent kinetic energy for the 60° angle of attack given in figure 4.4(g). The scattered vortex relocates to the side which is more near to the upper wall. So for this particular angle of attack as compared to the 0°, two turbulent vortices were observed. Out of these two, the vortex which was near to the upper wall underwent a drastic change of position.

Figure 4.4 shows various contours of Turbulent Kinetic Energy (k) in (m²s⁻²) (a) for 0° angle of attack, (b) for 10° angle of attack, (c) for 20° angle of attack, (d) for 30° angle of attack, (e) for 40° angle of attack, (f) for 50° angle of attack and (g) for 60° angle of attack
Figure 4.4  Contours of Turbulent Kinetic Energy (k) (m²s⁻²) (a) for 0° angle of attack, (b) for 10° angle of attack, (c) for 20° angle of attack, (d) for 30° angle of attack, (e) for 40° angle of attack, (f) for 50° angle of attack and (g) for 60° angle of attack (continued)
Figure 4.4  Contours of Turbulent Kinetic Energy (k) (m$^2$s$^{-2}$) (a) for 0° angle of attack, (b) for 10° angle of attack, (c) for 20° angle of attack, (d) for 30° angle of attack, (e) for 40° angle of attack, (f) for 50° angle of attack and (g) for 60° angle of attack (continued)

Figure 4.5 illustrated contours of Turbulent Intensity (%) (a) for 0° angle of attack, (b) for 10° angle of attack, (c) for 20° angle of attack, (d) for 30° angle of attack, (e) for 40° angle of attack, (f) for 50° angle of attack and (g) for 60° angle of attack (continued)
Figure 4.5  Contours of Turbulent Intensity (%) (a) for 0° angle of attack, (b) for 10° angle of attack, (c) for 20° angle of attack, (d) for 30° angle of attack, (e) for 40° angle of attack, (f) for 50° angle of attack and (g) for 60° angle of attack
Figure 4.5  Contours of Turbulent Intensity (%) (a) for 0° angle of attack, (b) for 10° angle of attack, (c) for 20° angle of attack, (d) for 30° angle of attack, (e) for 40° angle of attack, (f) for 50° angle of attack and (g) for 60° angle of attack (continued)
5 Conclusion

i. From the simulations performed on the triangular body it has been observed that the boundary layer separations decrease with the increment of the angle of attack. This particular characteristic is evident from the 20° onwards when the boundary layer separations become insignificant.

ii. A single high energy turbulent vortex was observed at the rear side of the body up to the 10° angle of attack. From the 20° onwards, two such turbulent vortices were observed at the rear side.

iii. The turbulent vortices that were observed becomes scattered with the increment of the angle of attack. At 60° the vortices becomes more scattered and the position of one particular vortex changes drastically. The vortex that was near to the upper wall changes its position from the rear side to another side (one of the two equal sides) which was also near to the upper wall.

iv. The two turbulent vortices were found to be very closely situated (side by side) to one another. As they were situated very close to each other, they were found to interact among themselves. This is due to the incoming supersonic flow which has a Mach number of 3.5. In the work by Chang et al [31] the two vortices were far apart as their inflow had a Mach number of 1.34.

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