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Computational Analysis of Shock Wave Boundary Layer Interaction using Shock Impingement Method

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Shock Wave Boundary Layer Interaction (SWBLI) is the most prominent phenomenon affecting supersonic flows, transonic aircraft wings, hypersonic vehicles and aero engines. The project employs Shock-wave Impingement Method to study the interaction of an externally generated shock-wave with the boundary layer of the surface it's made to impinge upon. A considerable range of wedge angles from 10° to 24° have been taken to analyse the interaction strength and the corresponding separation regions. Similar phenomena are responsible for increase in drag and decrease in efficiency of supersonic aircraft, the future design improvements of which, is the motivation of the study.

Keywords: Shock Wave, boundary layer, interaction strength, separation, supersonic flow, shock impingement

I. Introduction

One of the most basic phenomenon associated with any supersonic flow is of Shock Wave Boundary Layer Interaction (SWBLI)[1]. It has been seen in many high speed flow applications that shock waves interact with other stationary or moving surfaces. Since a surface inside a flow is characterized by some thickness of boundary layer, analyzing the interaction of shock wave generated, with the boundary layers is of prime importance from the view of design engineering.

The present work decodes the physics around the SWBLI region using the Method of Shock Impingement. In this method, an obstacle is placed in the path of the supersonic flow leading to generation of an oblique shock wave known as an externally generated shock wave. This wave is made to impinge onto a surface consisting a boundary layer.

Substantial success has been achieved in describing the phenomenology of low frequency unsteadiness, including correlations and coherent structures in the separation bubble, through complementary experimental and numerical studies on nominally 2-D interactions. In this contribution we shall be concerned with SWBLI produced by shock impingement due to a wedge and analyse the interaction strength and the corresponding separation region.

Apart from increase in drag and decrease in efficiency of the aircraft, the interaction of shock wave with boundary layer can also separate the boundary layer which makes the high speed flow more complex. For this study, ramps with four different angles (10°, 12°, 16° and 24°) are placed on the surface of the CD nozzle designed for Mach 2. The 2D geometry of the nozzle and obstacles have been made using the software Catia.

The analysis is carried out using density based implicit solver with SST k-omega model. This method basically solves for kinematic eddy viscosity and turbulent kinetic energy. The method is successful and efficient in analysing supersonic flow, shock wave regions and the separation regions. The computational results obtained are in accordance with both their experimental and theoretical counterpart [2][3], as reviewed in the literature. The changes in flow properties thus created, are successfully analysed and the theory behind these changes are appropriately understood and accounted for.

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II. Background

A CD nozzle (de Laval nozzle or con-di nozzle) is a tube that is pinched in the middle, making a carefully balanced, asymmetric hourglass shape. It is used to accelerate a hot, pressurized gas to a higher supersonic speed in the axial direction. Because of this, the nozzle is widely used in some types of steam turbines and rocket engine nozzles. It is also used in supersonic jet engines. A CD nozzle will only choke at the throat if the pressure and mass flow through the nozzle is sufficient to reach sonic speeds, otherwise no supersonic flow is achieved.

A. Method of Characteristics (MOC)

Method of Characteristics (MOC) was used to design the gradual expansion nozzle for Mach 2 [4]. It is a technique for solving partial differential equations which typically applies to first-order equations. The method is to reduce a partial differential equation to a family of ordinary differential equations. For a first-order PDE the method of characteristics discovers curves called characteristic curves along which the PDE becomes an ordinary differential equation (ODE). It can then be solved along the characteristic curves and transformed into a solution for the original PDE. Characteristics are 'lines' in a supersonic flow oriented in specific directions along which disturbances (pressure waves) are propagated.

B. Shock Wave-Boundary Layer Interaction

Air consists of molecules that move out of the way for an object to pass. When the speed of an object is greater than the speed of sound, the air molecules ahead of it have no warning and don't have time to move out of the way. This creates a compression region known as a shock wave. Shock waves propagate faster than the speed of sound have discontinuous and abruptly changing properties. Boundary Layer is the layer of the fluid in the immediate vicinity of the bounding surface where the effects of viscosity are significant. Boundary layer has a pronounced effect upon any object which is immersed and moving in a fluid. Drag on an airplane or a ship and friction in a pipe are some of the common manifestations of boundary layer. On the other hand, a shock wave is generated when the speed of an object is greater than the speed of sound.

Shock wave–boundary-layer interactions occur when a shock wave and a boundary layer converge and, since both can be found in almost every supersonic flow, these interactions are commonplace. These interactions also can be produced if the slope of the body surface changes in such a way as to produce a sharp compression of the flow near the surface – as occurs, for example, at the beginning of a ramp or a flare, or in front of an isolated object attached to a surface such as a vertical fin. SWBLIs can occur at any Mach number ranging from transonic to hypersonic, but it is in the latter category that the shocks have particularly dramatic consequences due to their greater intensity.

C. Shock Impingement Method

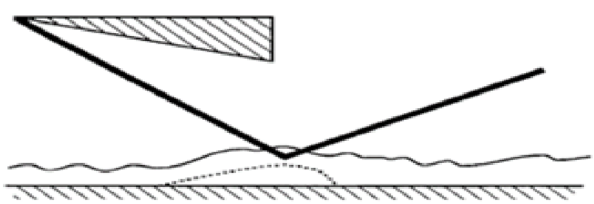


Figure 1: Shock wave boundary layer interaction through shock impingement [5]

In shock impingement method, a shock wave is generated by a wedge placed in a supersonic free stream. The shock impinges and reflects from the boundary layer. In the process the boundary layer thickens and for higher shock strengths (equivalent to higher wedge angles) it will separate [6]. This change to a separated flow occurs for shock strengths lesser than those that lead to a change of a regular reflection to a Mach reflection in inviscid flow. The incident and reflected shock both contribute to the overall pressure rise.

III. Model and Meshing

Meshing is done in the ansys workbench module, the unstructured grid has been generated. The element size for the meshing is taken as 0.2 mm and the mesh has further refined near the wall and near the obstacle (i.e. wedge), that is present inside the nozzle by the factor of 10, so the element size is 0.02 mm. The total number of elements are 360800 for constant area nozzle, about 450000 for 10°, 12°, 16° and 24° wedge angles present inside the nozzle.

For numerical simulations, ansys 19.0 is used. The implicit based density model is used for solving the governing equations of continuity, momentum and energy for compressible and supersonic flow. Standard wall treatment, no slip condition and stationary wall has been used for the wall. The SST k- turbulence model is a two-equation eddy-viscosity model. The flow, turbulent kinetic energy and the specific dissipation rate is solved by using the second order upwind equations, for better results and better accuracy. The fluid is modelled as air with ideal gas behaviour, temperature based specific heat (piecewise polynomial), constant thermal conductivity and sutherland law for viscosity. The constant value of thermal conductivity is 0.0242 W/m-K.In piecewise polynomial the value of the specific heat (C_p) is defined as the function of independent variable temperature. The temperature range that is given to the piecewise polynomial is between 300 K and 1000 K. Sutherland law shows a relationship between the dynamic viscosity, μ, and the absolute temperature, T, of an ideal gas.

IV. IIllustrative Results

Our first analysis was on the plain supersonic test section without any obstacle. The results verified that the nozzle indeed has been built for a mach number 2 supersonic flow. From the literature review it was known that the Inlet to Outlet pressure ratio of 4 Bar was required for achieving the desired mach number throughout the tunnel. Similar results were obtained by us as well.

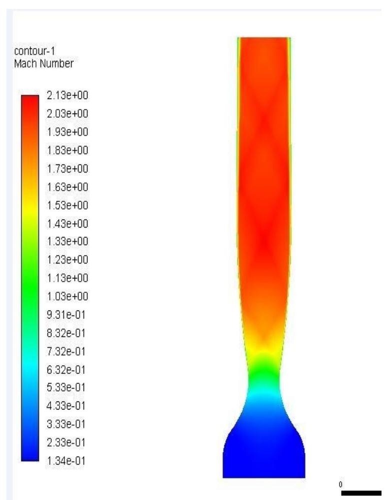
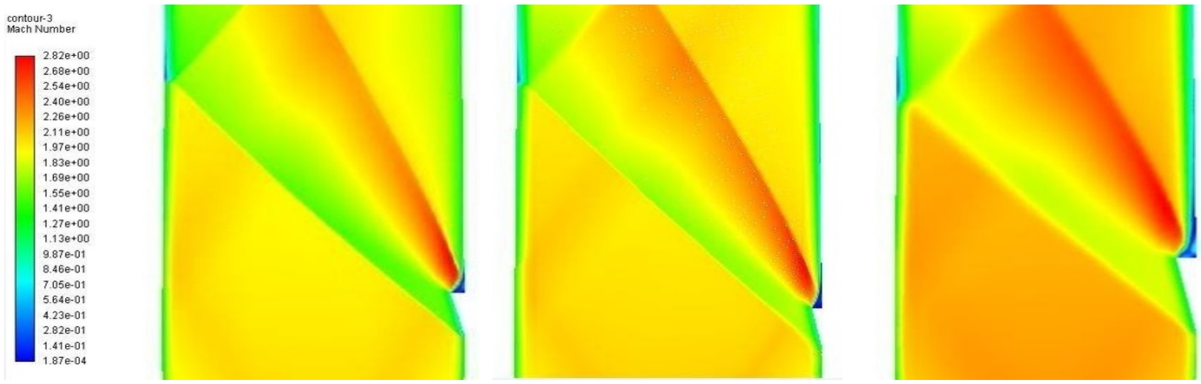


Figure 2: Contour of mach no. for 2-Dimensional CD nozzle

Moving forward, wedges with different wedge angles were introduced in the flow to analyse the effect of wedge angle over the strength of the oblique shock so generated. We have also looked at the attachment of shock wave thus created with the wedge, which again depends on the wedge angle. We have seen in the literature that the shock is supposed to detach from the wedge if the wedge angle goes beyond 22.5°. The same has been visualized through the simulations as well. It is known that for mach 2, the oblique shock will separate the boundary layer if the wedge angle is 6° or more[3], [7], [8], therefore, all the wedge angles taken for this study are greater than 6°.

The first wedge has the wedge angle of 10°. This is the smallest angle taken into consideration in the present study. It is evident from the results that, though the wedge angle is high enough to cause boundary layer separation, the interaction of shock with the boundary layer is quite weak. The separation region is quite small and the flow reattaches itself not long after the separation point. The separation bubble so formed is very small. For the subsequent cases of wedge angles 12° and 16°, the results indicate a little bit stronger interaction of shock wave with the boundary layer with the interaction in the 16° case being the strongest of the three. The separation region is still considerably small though, the size of the separation bubble consequently increases with the wedge angle. Comparing the pressures in the bubble we are able to justify that, the pressure increase in the separation region of a higher wedge angle case is more. Since the wedge angle is more so the shock created is stronger and therefore the drop in mach number across the shock is higher.



a)Wedge Angle 10° b)Wedge Angle 12° c)Wedge Angle 16°

Figure 3: Shock wave strength and separation bubble region comparison for wedge angles less than 22.5°

Till now, all the three cases had wedge angles less than 22.5° and therefore the shocks in all cases were attached to the wedge. But for the case of wedge angle 24° it is revealed that the shock wave is detached from the wedge. This is in complete acceptance with the theory[9]. Also, since the wedge angle is quite high, therefore, the strength of the shock is high as well. This case has resulted in a larger separation region which leads to a large separation bubble noticeable with the naked eye. Further since the shock is quite strong, so it is seen that the drop in mach number, and thus the pressure increase, after the shock is considerable and way higher than the previous cases.

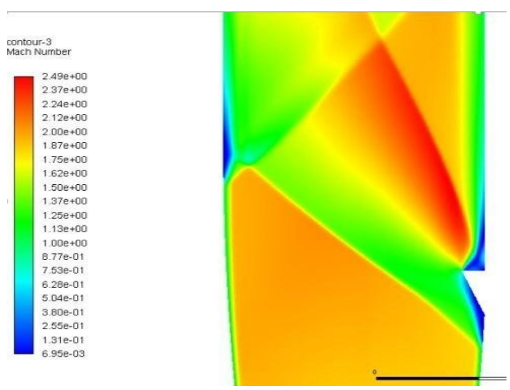
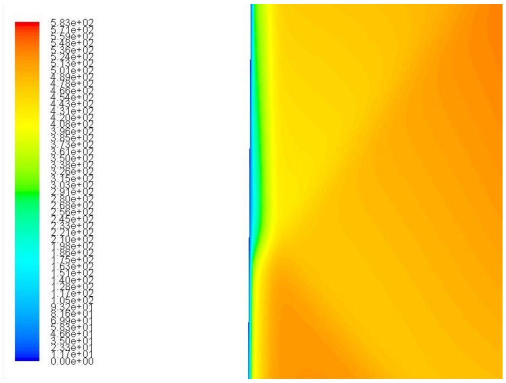


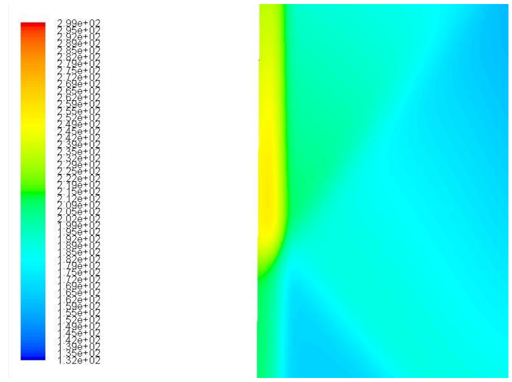
Figure 4: Separated shock wave formed at 24° Wedge Angle with a big separation bubble

Another observation which is eminent in all four cases is of early separation of boundary layer. It basically means that the separation region begins a bit ahead of the contact point of shock wave with the boundary layer. It is prominently visible in the case with wedge angle 24°. The reason for such behavior is that when shock wave interacts with the boundary layer, the presence of shock is felt upstream of its impact point. The shock gets transmitted upstream through the subsonic part of the boundary layer. This influence phenomenon, results in the pressure rise. At the same time, a boundary layer profile with high velocity has higher momentum, hence a greater resistance to the retardation imparted by an adverse pressure gradient. Since retardation effect is larger in the boundary layer inner part, a situation can be reached where the flow is pushed in the upstream direction by the adverse pressure gradient so that a separated region forms.[1].

Moving forward, the study further explores various properties and their variation in the SWBLI region. Velocity Magnitude, Static Temperature, Static Pressure and Skin Friction Coefficient, have been specifically focused upon.



a)Velocity Magnitude Contour



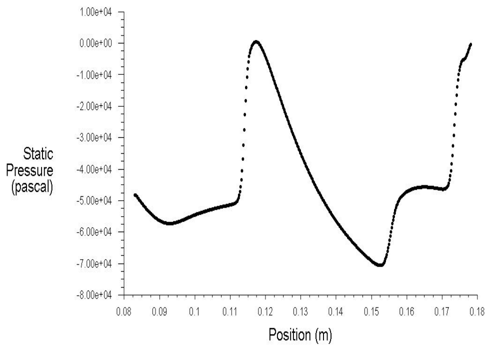
b)Static Temperature Contour

Figure 5: SWBLI Contour corresponding to wedge angle 10°

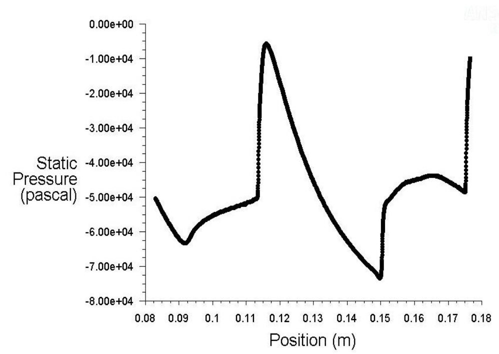
Figure 5 shows the Velocity Magnitude and Static Temperature contours for the case with wedge angle 10°. The primary reason to include these is that these contours validate our work with the past established works in this field. Though, the work presented here is unique and doesn't find any overlap with the literature with respect to the test case and the corresponding results, the works of Humble[2] and Sandham[5] provide the crucial parallel needed to validate the results. In his work, Humble did an experimental study of SWBLI for Mach 2.1 with a wedge with angle 10°, using PIV. The velocity contour in *Figure 5a* shows definite resemblance to the observations made using PIV. The sudden decrease in velocity in the SWBLI region is, as discussed before, because of the separation of the boundary layer. This makes the flow turbulent. The flow then reattaches to the wall, leading to velocity magnitude recovery. This leaves a separation bubble at the SWBLI contact point.

The same can also be observed using the Static temperature contours as shown in *Figure 5b*. Since the flow velocity decreases due to separation, one can expect the static properties to rise in this region. Sandham's work produced a temperature contour for SWBLI with turbulent boundary layer for Mach 2.3 with wedge angle 8°. Similar patterns are observed in the presented work as well. Though, none of the literature hold the exact data presented here, the two mentioned works are close to at least to one of our test cases and hence can be used for validation, apart from the general theory. The validation of this one case along with the theoretical understanding of SWBLI, provides a good argument to justify the rest of our results.

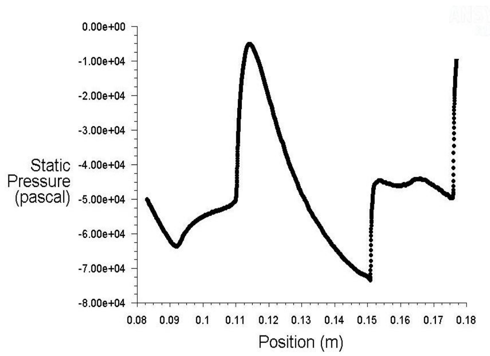
Now, as for the Static Pressure plots, the physics involved is relatively simple. Since, in the separation region the velocity of flow decreases, then assuming isentropic flow in the separation region, it is expected for static pressure to peak with a sudden rise. The results obtained are in line with this theory and hence are justified. Furthermore, it can be observed that as the flow reattaches and the separation region ends, the static pressure drops again with the increase in flow velocity. The interesting part here is the rise bump that comes afterwards. In all four plots, the partial rise in the plot is due to the expansion wave which is generated from the end of the wedge and hits the SWBLI wall(Wall on which we are observing the Shock wave Boundary layer interaction). Moving further along the wall, we see another peak building up as the wall comes to an end. This is actually the reflected shock coming back and hitting the surface. This phenomenon is better visible in the plot for wedge angle 24° because in this case, due to high wedge angle, the incidence angle of the shock with the normal of the SWBLI wall is less. Consequently, the reflection angle at the opposite wall is also low. These angles are low enough to allow the reflected shock to hit the SWBLI wall before the end of the wall. This is the reason why the second peak is completly visible only for the 24° wedge angle case. The contour plot for the same can also be seen in *Figure 7*.



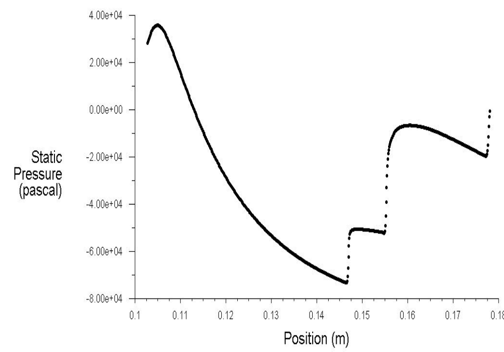
a)Wedge Angle 10°



b)Wedge Angle 12°



c)Wedge Angle 16°



d)Wedge Angle 24°

Figure 6: Static Pressure Plots showing spike in SWBLI region

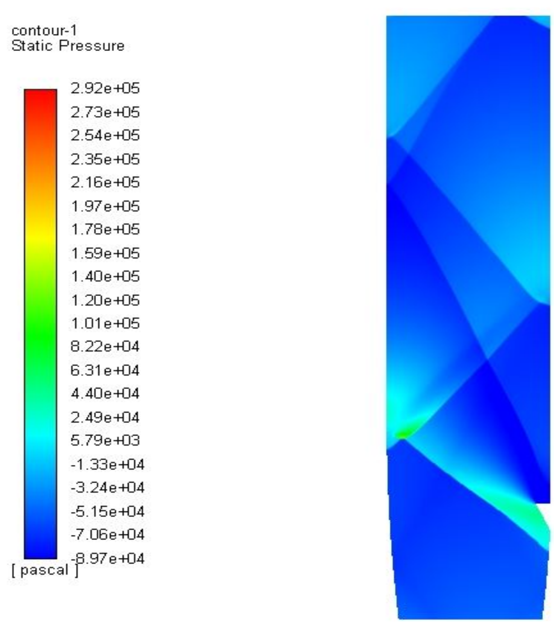
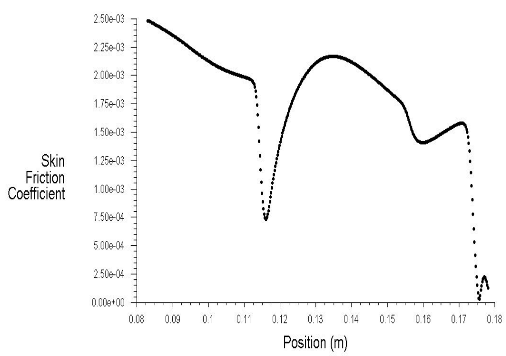
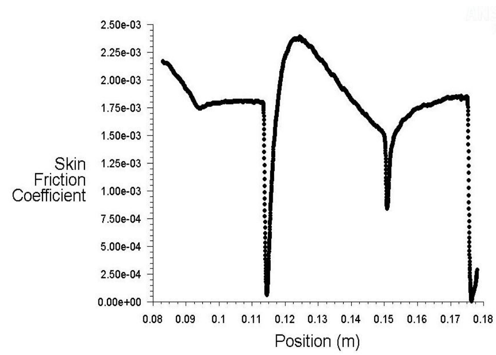


Figure 7: Contour of Static Pressure for 24° wedge angle case

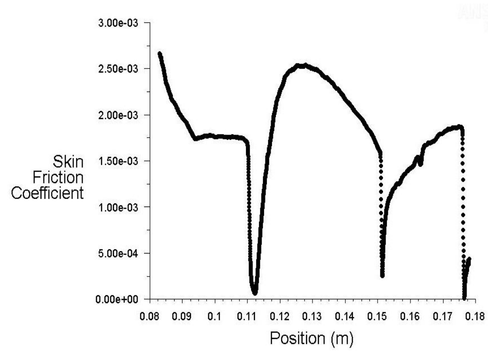
Now, coming to skin friction coefficient, it is dimensionless skin shear stress which is non dimensionalized by the dynamic pressure of a free stream. *Figure 8* shows the plots of skin friction coefficient along the SWBLI wall, for three different cases of wedge angle 10°, 12° and 16°. Coefficient of Skin Friction gives the frictional/resistive force offered to an object moving in a fluid. But, since we have a stationary wall, so in our case it will yield the hindrance offered by the wall, due to it's No slip condition, to the fluid flow. This mainly happens because of the viscous nature of the fluid. The resistance of the boundary layer to shock-induced separation can be identified by the distribution of the skin-friction coefficient, displayed in *Figure 8* for all the shock-impingement cases. In the interaction region, the wall shear stress is characterized by a remarkable drop, caused due to the retardation imparted by the adverse pressure gradient on the near-wall profile, followed farther downstream by a gradual recovery. For higher wedge angles the separation bubble region has mainly turbulent flow, which for the majority part, is not along the wall. Since, c_f gives the frictional resistance offered to the flow along the wall, therefore, for these high wedge angles it drops to almost zero. However, it can be seen in *Figure 8a* that for small wedge angles like 10°, where separation region is extremely small, the c_f doesn't quite drop to it's lowest, since we can still expect some component of the flow along the wall. Another observation is that, since the separation region increases with the increase in shock strength (due to increasing wedge angle), c_f value stays near minimum/Zero for a longer length along the wall because the separation region increases. It is visible that the drop for 12° case has a sharper peak than 16° case and that the peak for 16° case has a smoother curve. Lastly, the recovery region is basically where the separated flow starts to attach to the boundary again. This yield to a peak in the c_f plots whose heights are again directly proportional to the wedge angles and by extension, the strength of the shock.



a)Wedge Angle 10°



b)Wedge Angle 12°



c)Wedge Angle 16°

Figure 8: Skin Friction Coefficient Plots for SWBLI Wall

Skin Friction Coefficient gives us the Skin Friction Drag, which is a part of the Profile Drag. Since the different test cases have yielded different variation in c_f, therefore, we can deduce that the Drag on an object travelling with supersonic speed would greatly be affected by these consideration.

The reason we are emphasising on drag due to SWBLI is because it is one of the most dangerous phenomenon associated with supersonic flows. A few of the most basic examples of hazardous effects of drag being that, the drag increases on the control surfaces affects the control effectiveness of the control surfaces by increasing the stick force required and a drag increases on the lifting surfaces contributes directly to the overall drag. It also increases stress on the structure and leads to structural fatigue and failure. Therefore, all the designs for supersonic aircraft revolve around the necessity to reduce drag.

V. Conclusion

This work outlines a computational study to gain the insight into the shock wave boundary layer interaction, to check the varied interaction with wedges of different angles and to compare these results with theoretical and experimental data.

The results presented here demonstrate a stronger interaction between shock wave and boundary layer as the wedge angle increases. Consequently, the separation region increases as well. The detachment of shock wave for the deflection angle greater than 22.5° is also established through the simulations. The variation in properties like Static Pressure and Skin Friction Coefficient is also documented. The corresponding implications of these observations in practical aspects have also been explored.

In the future, the study can be extended to understand phenomena like glancing shock which is a major source of instabilities in modern supersonic aircraft and engines; design improvements and suggestions being the ultimate aim.

VI. Acknowledgement

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