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CFD Studies of Flow over a 2D Supersonic Compression Corner

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Abstract—*Shock wave and boundary layer interactions in presence of a thermal wall have been considered in the present study. The work comprehensively details the effect of wall temperature in the nature of shockwave produced and is an attempt to study the flow parameters aft of the shock. A two dimensional finite volume approach using a commercial CFD solver have been used to study the pressure variants in a 25° compression corner .To juxtapose the results the simulation was done in three different temperature conditions of the wall viz. adiabatic wall, 300 K and 400 K. Results show that the pressure increases with increase in wall temperature, and the obliqueness of the shock decreases with increase in the wall temperature,.*

1. INTRODUCTION

Shock waves are basically pressure waves produced by an object moving faster than the speed of sound. When an object moves with a speed greater than the speed of sound, the sound waves are piled up and compressed behind the object. The waves protruding are confined to a cone that narrows as the speed of object increases and the waves bunch up, creating high pressure regions outside the compressed waves. The strength of a shock wave dissipates greatly with distance, much more so than a regular wave, as heat and other energy are more quickly transferred into the surrounding environment. Once enough energy has dissipated, the shock wave will become a regular wave such as a sound wave.

The boundary layer of a flowing fluid is the thin layer close to the wall where the viscous stresses and velocity gradients are very prominent. The boundary layer theory was developed by the well-known German scientist, Ludwig Prandtl in 1904 which enabled ground breaking developments in the field of aerodynamics. This theory states that when a free-stream approaches parallel to a sharp-edged, thin, smooth flat plate boundary layer is formed. The fluid, with a uniform free-stream velocity U, gets retarded in the vicinity of the solid surface due to frictional effects and boundary layer region begins at the sharp leading edge. At subsequent points downstream the leading edge, the boundary layer region increases because the retarded fluid proceeds and it is further retarded, also known as growth of the boundary layer. Within this thin *boundary layer*, the velocity rises rapidly from zero at the wall to the freestream value at its edge.

According to Newton's shear–stress law, which states that the shear stress is proportional to the velocity gradient, the local shear stress can be very large within the boundary layer even if the viscosity were small.

Near the leading edge of the solid surface, where the thickness is small, the flow in the boundary layer can be laminar and the layer is said to be laminar boundary layer. As the thickness of the layer increases in the downstream direction, the laminar layer becomes unstable and the motion within it is disturbed and irregular which leads to a transition from laminar to turbulent boundary layer. Turbulent flow is characterized by greater interchange of mass, momentum and energy within the fluid particles than in the case of laminar flow.

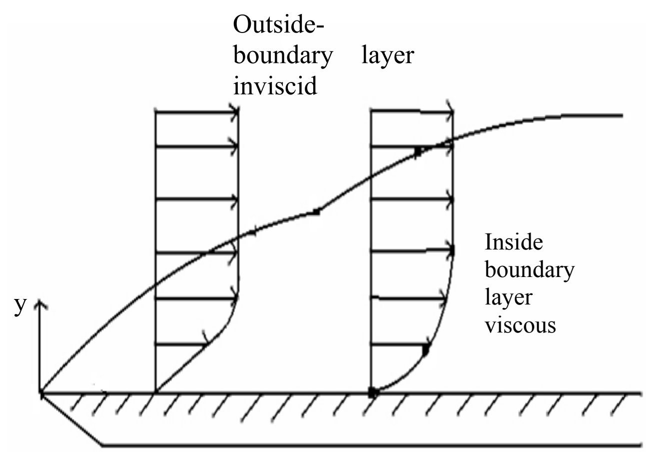

Fig. 1: Boundary layer region over a flat plate

2. PROGRESS IN SWBLI

From 2007, [2]H.D.Kim[a] and T.Setoguchi[b] international symposium presentation: Shock-wave boundary layer interactions occur in all practical transonic, supersonic and hypersonic flows. When shock waves encounter with boundary layer, it results in many diverse types of flow phenomena such as flow separation, unsteadiness, vertical flow, pressure waves, complicated mixing, turbulence etc. The effect of SWBLI on flow properties depends on the strength of interaction. The parameters that determine the strength of interaction are: 1) Mach number 2) Reynold's number 3) Characteristics of boundary layer flow. For a weak interaction ($1 < M_1 < 1.2$), the boundary layer thickens due to reduce in flow velocity caused by adverse pressure gradients. In moderate interaction ($1.2 < M_1 < 1.32$), a small separation bubble is formed cause of flow separation and reattachment. In strong interaction ($1.32 < M_1 < 1.5$), a sizable separation bubble is formed and for $M_1 > 1.5$, large separation bubble and shock separation is observed which is very complicated wave system (or multiple shocks).

EFFECTS OF SWBLI IN EXTERNAL & INTERNAL FLOW:

EXTERNAL FLOW: 1. Increased aerodynamic drag and loss of lift

2. Aerodynamic heating

3. Increased instabilities such as

inlet buzz and buffeting

INTERNAL FLOW: 1.Total pressure load and unsteadiness

2. Loss of flow control performance

3. Increasing turbulence level

It was studied that as Mach number increases, the minimum of wall shear stress decreases and reaches a zero point where tiny separation bubble is formed. Also, the separation pressure rise (P_s) is insensitive to variation of Reynold's number. Separation pressure rise depends on Reynold's number only for high Mach numbers and it is independent of Reynold's number for minimum Mach number (i.e. Mach number required to separate boundary layer).

In 2012, [3] Argyris G. Panaras[a] and Frank k. Lu[b] have studied the effect of micro vortex generators (MVGs) on SWBLI. The computational and experimental work showed that the use of MVGs has significant effect in the suppression of shock induced separation. Vortex generator is an aerodynamic device which generates stream-wise vortices to increase the low-momentum boundary layer region near any wall surface and there by delaying local flow separation. They used forward forcing ramps (called micro ramps) and pairs of tapered vanes (vane pairs) for computational simulations. Micro ramps and vane pairs are type of vortex generators used to increase boundary layer momentum near the wall surface. The results showed that the vane pairs are slightly more effective than micro ramps in suppressing flow separation. Nonetheless, micro ramps are featured by structural sturdiness, which is quite effective for supersonic air intakes. Hence, the split ramp and the ramped vane were proposed as they combine the high performance of a vane device along with the sturdiness of micro-ramp device.

In 2012, [4]Man Sun Yu[a], Jiwoon Song[b], Ju Chan Bae[c] and Hyung Hee Cho[d] have studied about the major effect of SWBLI – aerodynamic heating. Aerodynamic heating causes damage to the external skin as well as internal skin of scramjet/ramjet engine structures. The increase in heating value may lead to burn-out of important units and it also increases the thermal stresses on engine structure which is undesirable. Hence, it is necessary to study the heat transfer and flux rates across the separation region. They have reviewed many scholarly articles on aerodynamic heating due to SWBLI. The reviewed articles measured the heat flux rates surface and around the inclined ramp/sharp fin, along the length of separation bubble and surface temperature images were taken using TLC (Thermochromic liquid Crystal) thermography technique on a laminar flow separation. In their study, they have measured the surface temperature images on cylinder solid body using infra-red thermography when boundary layer separation occurs.

The measured data is used to determine convective heat transfer coefficient, surface shear flow and shock wave structure. From data and determine values, it was concluded that sweeping of cylinder is an effective way for the suppression of heat transfer increase around cylinder and reduction of separation area.

In 2013, boundary layer interactions in hypersonic flows have been done by [5] Bibin John[a], Vinayak N. Kulkarni[b] and Ganesh Natarajan[c]. They considered the study of parameters that effect the separation of laminar boundary layer. The factors effecting laminar boundary layer separation are Mach number (M), Reynold's number (R_e), Wall temperature and boundary layer stability.

They also conducted CFD simulations to predict the length of separation bubble from pressure and heat transfer rate measurements. The effect of very small leading edge bluntness on the size of the separation bubble is also considered. The results from the simulations have been summarized below:

1. The numerical data of separation bubble length can be obtained by analyzing wall shear stress distribution.
2. The separation bubble length increases as the ramp angle and the wall temperature increases.
3. The length of separation bubble decreases as the Mach number and free stream stagnation enthalpy increases.

Finally, the effect of leading edge radius on separation bubble length verifies that there is a strong correlation between the parameters measured and length of separation bubble and hence, require more attention on these parameters for suppressing flow separation in laminar boundary layer.

3. COMPUTATIONAL METHOD

A 2 dimensional model of length 0.75m and a ramp of 25° located at 45% of the total length , with the inlet having 0.25m has been designed using commercial software, ANSYS 15.The model has been meshed using the same software. Automatic mesh method was considered and fine meshing was done near the walls, with mesh size of 2e-003 m. The final mesh was transferred to Fluent solver of the ANSYS 15.The representation of the mesh is shown in *Fig.* 2.

Numerical investigations have been carried out to study the effect of wall temperature on aft shock flow properties. The simulation was steady state density based and the fluid was ideal gas .K-epsilon turbulent model has been used. Free stream Mach number (M_{∞}) of 4 was used in all the different conditions of wall temperature (Tw) viz. adiabatic, Tw=300K and Tw=400K. The relation between coefficient of pressure (Cp) and length (X) was plotted in Mat lab and is shown in *Fig.* 3.

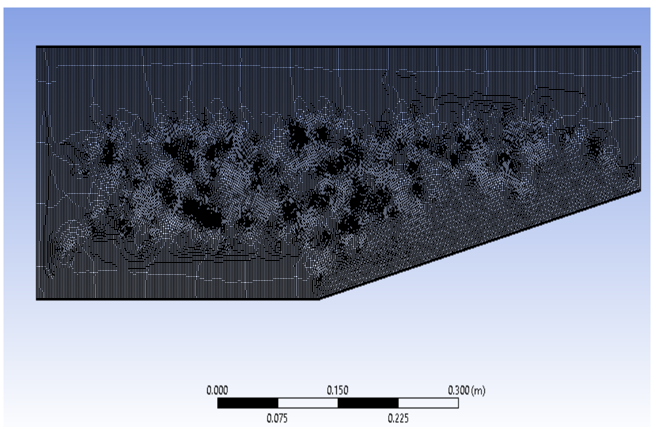

Fig. 2: Meshed Computational domain

4. RESULTS AND DISCUSSION

The simulation was carried out using ideal gas, and the viscosity was calculated using Sutherland's Law. Courant number in the range of 0.2-0.4 was considered for numerical stability. Isentropic condition was assumed fore and aft of the shock wave. The simulation was carried for 1000 iterations and the results were achieved.

The first simulation was carried with adiabatic wall condition, where in no heat was generated or considered in the wall. The contours of the Pressure (*Fig.* 4) were observed and the variation of Cp along X was plotted. The next simulation involved the raising of wall temperature Tw, to 300K ,followed by Tw = 400K.The Contours could not be differentiated visually , however numerical values of Cp along X for each condition shows the effect of wall temperature to pressure .The pressure increases when wall temperature increases.

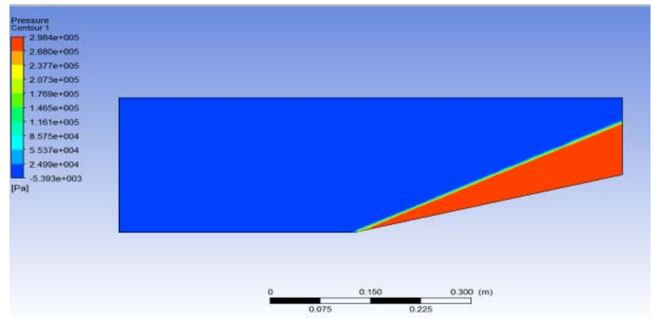

Fig. 3: Adiabatic Wall Pressure contour

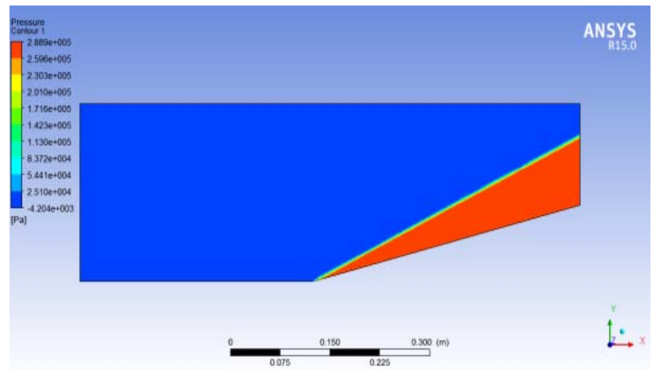

Fig. 4: at Tw= 300K Pressure Contour

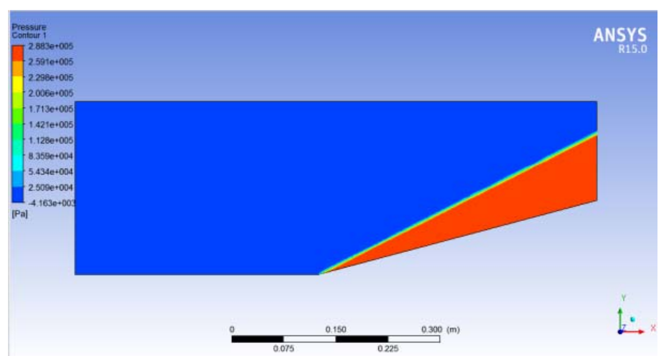

Fig. 5: Tw= 400K Pressure contour

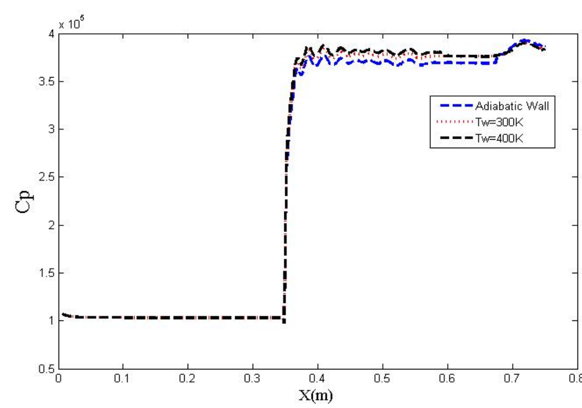

Fig. 6: Effect of Wall temperature on Pressure distribution

5. CONCLUSION

From the result it can be concluded that the increase in temperature results in increasing the pressure across the shock wave. The numerical values of Cp at different Tw condition shows the change in obliqueness in the shock, with lowest Oblique angle at Tw=400K and highest when the wall is adiabatic. This results finds vast application in the Aerospace industry, the results can be inferred to understand the flow parameters in a scramjet engine, the heat produced during combustion may account for the increase in wall temperature which in turn increase the pressure post Shockwave.

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