Numerical study of a shock-wave boundary-layer interaction: compression ramp flow-fields

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Abstract. Shock-wave boundary-layer interaction (SBLI) causes a serious aerodynamic and structural problems, therefore understanding its complex flow phenomenon is essential. Flow properties at the interaction region were numerically predicted via solving steady, compressible, 2D Navier-Stokes Equations with Roe-FDS flux type and a second order upwind flow discretization. Two compression ramps with 12° and 24° angles were investigated. The flow over a 12° ramp were fully attached with a λ-shock type and a limited upstream influence. Wall temperature increases to 2.55 times the edge temperature value and affected region increases with the downstream distance. The pressure field for the 24° ramp case reveals a substantial increase in the upstream influence due to the strong shock. A larger shock-leg at the wall provoke a significant flow separation and the wall temperature increases to 2 times the edge temperature value. Separation bubble is generated at the corner, with a separation length of 3.25 times the boundary layer thickness. This leads to a noticeable increase in the total pressure drag. Simulated results were fairly agreeing with the experimental result at the separation region.

1. Introduction

Shock wave turbulent boundary layer interaction and shock reflection results in a complex flow structure. Interaction has an adverse consequence on the transonic, supersonic and hypersonic flights and must be considered at the early vehicles design stages. SBLI influence aircraft external structure via losses of controllability and extreme heat and mechanical surface loading in an unsteady manner. Also, it increases the pressure losses and distortion in the internal flow. Viscous-inviscid interactions will take place at this flight regime and basically leads to flow separation, vorticial flow structures, turbulence intensification and structure alterations. Research of the SBLI started in the mid of the twentieth century.

Inside the boundary layer shock bends as it meets a relatively low Mach numbers, breaks into compression fan and shock reflection may be generated [1]. Further if the shock is strong enough boundary layer may separate. The compression corner flow has a single major shock associated with the ramp. In the mid-1990s extra experimental measurements were conducted to provide validation data for numerical codes [2].

The standards chosen were that the data be turbulent, on relatively simple geometries, applicable in the sense that they are good enough to explore turbulence modelling, those studies shows that experimental boundary conditions and error bounds, were self-consistent and adequately resolved. Similar researches of surface pressure, field Pitot survey, skin friction and heat transfer at Mach 8.3 on impinging shock, single-fin and double fin interactions were conducted [3-4]. In the same research area [5-6] published skin friction coefficient results for the double fin. These data were useful to predict
surface properties using turbulence modelling aspects. [7] compile RANS result from simulations for numerous 2D and 3D interactions.

The models started from algebraic to full Reynolds Stress Equation formulations for the single fin, double fin and the hollow cylinder flare. They concluded that skin friction and heat transfer predictions were poor, with up to 100 percent difference for the strong interactions showing a substantial region of separation. There are still significant quantities that cannot be predicted correctly, such as peak heating in strong interactions [8].

Direct numerical simulation codes were used by [9] to study the SBLI on a flow at Mach 2.25 with an impinging oblique shock wave with a shock angle of 33.2°. The study reveals that due to the vortex-shedding near separation point, along with the vortical structures diffusion in the mixing layer, a pressure waves was produced due to shock interaction at the foot of impinging shock, and upstream propagation of acoustic frequencies of cavity tones, caused the low-frequency unsteadiness. [10] evaluated SBLI unsteadiness by studying a shock reflection using plate angles of incidence from 7° to 9.5° with a free stream flow of Mach 2.3. Experiments reveals fluctuation shock motion produces a relatively lower frequency than the incoming boundary layers characteristic frequencies. The shockwave acted as a low pass filter, which selected only the lower frequency excitation. Velocity fields was measured in case of shock impinging boundary layer at Mach 2.3 on two wedges with angles 8° and 9.5° using particle image velocimetry [11]. It reveals that flow reattachment downstream the shock wave depends on the fluid properties in the mixing layer. The low frequencies of the separation shock were associated with the separation bubble dilatations and contractions.

The dominant frequencies for the 8° and 9.5° configuration were close to the theoretical value. The calculated Strouhal number agreed with the values obtained in experiments conducted by [12] and [13] for different configurations. [14] in his review paper declared that, correlations and coherent structures in the separation bubble, were successfully described on nominally 2D interactions. Heat transfer can be measured and, in several cases, predicted, accurately for axisymmetric laminar situations. [15] simulated Mach 2 turbulent compression corner flows with 22.5° and 37.5° sweep angle and found no indications of a low-frequency unsteadiness. They argue that compared to unswept interactions for swept interactions the reverse flow intensity is not a good indicator for the onset of instability since disturbances are convected outward in the spanwise direction. [16] carried out simulations of a hybrid turbulence model with 1-equation renormalization group turbulence model as well as LES with wall-adapting local eddy-viscosity model of turbulent swept and unswept interactions. The simulations revealed ripples of shock separation for the unswept case that appeared weakened in the presence of sweep.

In this work a steady 2D numerical simulation of SBLI flow-field phenomenon at supersonic flow regime was conducted. Shock-wave boundary-layer interaction flow-filed features, such as pressure loads (reflecting the shape and criteria of the SBLI), velocity profiles, wall shear stress distribution and heat transfer were predicted. All those considerations are vital for supersonic engine inlet design, control surfaces functionality and understanding of the internal supersonic nozzles flow.

2. Mathematical Model

Navier-Stokes equations were solved numerically. Spalart-Allmaras turbulence model was selected to account for turbulence. The equations for conservation of mass, momentum and energy are given below:

\[ \frac{\partial \rho}{\partial t} + \nabla (\rho \vec{v}) = 0; \]

\[ \frac{\partial (\rho \vec{v})}{\partial t} + \nabla (\rho \vec{v} \vec{v}) = -\nabla p + \nabla (\vec{r}) + \vec{F}, \]

where \( p \), \( \vec{r} \) and \( F \) are static pressure, stress tensor and surface forces, respectively. The stress tensor \( \vec{r} \) is below shown:

\[ \vec{r} = \mu \left[ (\nabla \vec{v} + \nabla \vec{v}^T) - \frac{2}{3} \nabla \cdot \vec{v} I \right], \]

where \( \mu \) and \( I \) are the molecular viscosity and unit tensor respectively. Energy transport equation is:
\[
\frac{\partial (\rho E)}{\partial t} + \nabla [\tilde{v}(\rho E + p)] = \nabla [k_{\text{eff}} \nabla T + (\tilde{v}_{\text{eff}} \tilde{V})],
\]

where \(k_{\text{eff}} \nabla T\) is the conduction and \(\tilde{v}_{\text{eff}} \tilde{V}\) is the viscous dissipation terms. Energy \(E\) per unit mass is defined as:

\[
E = h - \frac{p}{\rho} + \frac{u^2}{2},
\]

where \(h\) is enthalpy. The viscous stress tensor \(\tau_{ij}\) is related to the Reynolds stresses through the eddy viscosity \(\mu_t\), effectively modelling the momentum transfer by turbulent eddies, written as

\[
\tau_{ij} = 2(\mu_1 + \mu_t)(s_{ij} - \frac{1}{3} \frac{\partial u_k}{\partial x_k} \delta_{ij}),
\]

where \(\delta_{ij}\) is Kronecker delta and \(s_{ij}\) is the strain rate tensor given by

\[
s_{ij} = \frac{1}{2} \left( \frac{\partial u_i}{\partial x_j} + \frac{\partial u_j}{\partial x_i} \right).
\]

The Spalart-Allmaras turbulence model is effective for flows at high Reynolds number and shows good results for boundary layers subject to adverse pressure gradients. Spalart-Allmaras model was used in this study to consider the effect of turbulence. The SA model utilizes one equation to solve the transport variable where \(\tilde{v}\) is identical to the turbulent kinematic viscosity except in the near wall region. The transport equation for \(\tilde{v}\) is:

\[
\frac{\partial}{\partial t} (\rho \tilde{v}) + \frac{\partial}{\partial x_i} (\rho \tilde{v} u_i) = G_v + \frac{1}{\sigma_{v}} \left( \frac{\partial}{\partial x_j} \left[ (\mu + \rho \tilde{v}) \frac{\partial \tilde{v}}{\partial x_j} \right] + C_{b2} \rho \left( \frac{\partial \tilde{v}}{\partial x_j} \right)^2 \right) - Y_v + S_v,
\]

where \(G_v\) is turbulent viscosity, and \(Y_v\) is the near wall turbulent viscosity destruction, \(\sigma_v\) and \(C_{b2}\) are the constants and \(\nu\) is the molecular kinematic viscosity. \(S_v\) is the source term.

3. Numerical Setup

Shock wave originates at a discontinuity in the wall direction, through which the supersonic flow experiences a deflection equal to the corner angle. The flow structure depends on the corner angle, therefore two ramps with a 12° and 24° angles were selected for this numerical study. The structure meshes generated over the two-dimensional domain of total length \(L = 0.6\) m and ramp located at \(L = 0.2\) m from leading edge. The mesh for 12° and 24° ramp cases consisted of 8891 nodes and 17790 faces as shown in figure 1.

![Figure 1](image-url)
Grids were clustered within the boundary layer and at the expected shock regions to ensure a proper flow resolution. The values of $Y+ = \frac{Y}{\mu} \sqrt{\rho_{\text{ref}}}$ in the wall-adjacent cells ranges between 0.3-0.6 and 0.15-0.35 for the 12° and 24° corner angles respectively. Those ranges indicate a laminar sublayer resolution was achieved. Flow domain boundary conditions are illustrated in figure 1.

4. Results and discussions

Results of a supersonic flow at free-stream Mach number $M = 2.95$, Reynolds number based on total length equal to $Re = 3.9 \times 10^7$ and boundary layer thickness at the edge corner $\delta_i(L=0.2) = 2.79$ mm are discussed below.

4.1. Wall Pressure Distribution

Pressure distribution over the 12° and 24° ramps are shown in figure 2 (a) and (b) respectively. For the case of 12° the pressure at the wall starts to rise by the upstream propagation mechanism reach a value of 2.35 times the free stream value. This result shows a very good agreement with the theoretical value which found to be equal to 2.3136. This trend indicates a weak shock-boundary layer interaction.

Figure 2 (b) shows the pressure distribution of the 24° ramp. The pressure curve exhibit three inflection points. Those respectively associated with separation, the onset of reattachment and the reattachment compression. This result indicates a good agreement with experimental results of [17-18] and excellent agreement with inviscid solution outside the interaction region.

4.2. Pressure contours over the flow field

When a shock-wave propagates through a boundary-layer, it sees an upstream flow of a decreasing Mach number as it approaches the wall. The shock adapt itself and becomes vanishingly weak when it reaches regions of sonic flow. The pressure is transmitted in the upstream direction through the subsonic inner part of the boundary-layer. Thus, a pressure rise is felt upstream of the point where the shock would meet the surface.

Figure 3 (a) show the pressure distributions over the 12° ramp. It reveals that a distinct shock-wave arise from the corner location with the shock wave angle of $\mu = 31.5^\circ$, this value is 6% higher than the theoretical value. Within the boundary layer the shock is curved due to its propagation and the entropy changes from one streamline to the other. Outside the boundary-layer, the shock is linear. The upstream influence is evidence but relatively take place over a small region in front of the ramp.

Pressure distribution for 24° corner angles case is shown in figure 3 (b). Again, a single major $\lambda$ shock associated with the ramp is evidenced. A substantial increase of the upstream influence length due to an intensified shock strength is noticed along with a large spreading of the shock near to the wall is visible. This is due to the compression waves induced by the thickening of the low velocity portion of the boundary-layer. In this case the pressure rise is strong enough to encourage significant separation with a length of 9 mm, leads to a high-pressure rise.
4.3. Velocity profile and streamlines distribution

Velocity profiles for the 12° and 24° ramps are shown in figure 4. The velocity profile of the incoming flow for both cases shows a trend of a typical turbulent velocity profile. The transition to zero velocity at the wall takes place over a very short normal distance, hence the subsonic layer is extremely thin, and the shock originates from a region very close to the wall. Incoming turbulent boundary layer, $\delta_i$ is found to be 2.77 mm. Right behind the shock, the boundary layer thickness increases to 3.3 and almost an order of magnitude for the 12° and 24° ramps, respectively, which indicates a large region of separation, especially for the 24° ramp. For the latter case, the pressure rise is strong enough to provoke significant flow separation clearly shown by the reverse flow portion near the ramp leading edge as illustrated in figure 5 (a). Streamlines distribution of the flow shown in figure 5 (b) illustrates clearly formation of separation bubble stating at point $S$, with a length of $L_{sep}$ equal to 9 mm, generated at the corner. Flow reattachment is taking place right behind this bubble at point $R$. 

Figure 3. Static pressure contours flow over (a) 12° and (b) 24° ramps.

Figure 4. Velocity profiles over the 12° and 24° ramps.
4.4. Heat transfer and near wall temperature distributions
Wall adjacent temperature profiles at the corner region is presented for 12° and 24° ramps in figure 6. The near wall temperature is directly reflected into heat transfer to the wall as well as the skin friction at the wall. Temperature profiles at the corner region for 12° ramp shows an increment at an order of 2.55. Affected region is increasing with the downstream distance. Same trend is evidence for 24° ramp case with smaller order of increment due to flow separation.

![Wall adjacent temperature profiles](image)

**Figure 6.** Wall adjacent temperature for (a) 12° and (b) 24° angles ramps.

4.5. Skin friction coefficient
Skin friction coefficient of the 12° and 24° ramps are shown in figure 7 (a) and (b) respectively. For the case of 12° ramp the skin friction coefficient decreases to its minimum value in front of the corner where $C_f = 0.00015$ at the ramp corner. This value grows sharply up to 0.0025 at the shock region at the corner leading edge and remains nearly constant over the rest of the ramp. This skin friction coefficient distribution reveals attached flow at the corner. While for 24° ramp the skin friction coefficient plummets to the value of $C_f = 0.0005$ near separation point $S$, then reverse its direction in a rather complex variation and further decreases to zero indicating reattachment of the flow $R$. This complex variation indicates flow separation at the corner. Further skin friction coefficient recovers to a value 0.003 reattachment point.
5. Conclusion
Turbulence, compressibility and viscous-inviscid interaction phenomena are present in the SWBLI flow. For all that, numerical simulation is providing a further insight of those phenomena. SWBLI is numerically simulated for a compression ramp flow with corner angles of 12° and 24°. Propagation through a boundary layer results in a \( \lambda \) shape curved shock for the case of 12° corner with fully attached flow at the corner. Temperature profiles shows an increment of 2.55 times the edge temperature value. Affected region is increasing with the downstream distance. For the 24° ramp the upstream influence is relatively higher due to the increased shock strength with a wide spreading near the wall that provoke significant flow separation. The streamlines of the flow reveal that a separation bubble is generated at the corner, which leads to the boundary layer separation. Same wall temperature trend is evidence as for 12° ramp but, with a smaller increment due to flow separation in this case. Simulation and experimental results in the shock impinging region are found to be in a fair agreement.

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