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A Review on Effect of Shock Wave Boundary Layer Interaction in different Flows Fields

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Abstract: Shock wave boundary layer interaction (SBLI) review based on different concepts of interacting profiles in different boundary-layer conditions on different models and applications. From past 50 years of research shock wave boundary interaction (SBLI) has played the most crucial role while designing the Air vehicle (like Rocket, Airplane, Missile, etc.,). Here considering ten different papers and deeply understanding the concept and reviewing the literature. The shock wave boundary layer interaction has various kinds of usages in the aerospace industry (like transonic and supersonic flight vehicle surfaces, and surfaces of rockets, missiles, transonic turbine blade passages, and re-entry vehicles, transonic gas turbine blade tip gaps, scramjet isolator ducts, supersonic aircraft engine intakes). Considering various applications mentioned above and taking the research paper for each application and reviewed which has been done in the paper through which medium. Considering the numerical method to determine the concept of shock boundary layer interaction (SBLI) and comparing the result with CFD to provide the data that attain in an efficient manner. And Conducting experiments through different wind tunnels (like LENS XX, transonic and supersonic wind tunnel) and comparing the result with CFD. Determining the dimension flow while conducting the experiment.

Keywords: Shock wave, Boundary layer, SBLI, Transonic flow, Supersonic flow, Hypersonic flow, CFD, 2D flow and 3D flow.

1. INTRODUCTION

When going to design any air vehicle that flies through the Earth's atmosphere, the fluid flow acts on the vehicle to opposes the vehicle is a drag. Drag is generated in the direction fluid flow is moving when it encounters a solid object (Air vehicle). To minimize the drag and other factors that act on an air vehicle, should consider the Shock wave boundary layer interaction (SBLI) on the vehicle. There are different types of experiments conducting on the shockwave boundary layer interaction in the last 50 years for better interaction to attain minimal drag and other factors in the air vehicles (e.g.; Aircraft wing, Rocket Nose, Re-entry vehicle nose, etc). Getting the points more clearly from the next paragraph should start from shock waves.

The type of propagating disturbance that moves faster than the local speed of sound in the medium is known as a Shock wave. Generally, a shock wave carries energy and can propagate through a medium but is characterized by a sudden, change in pressure, temperature, and density of the medium. The shock wave is divided into three types:

- Normal Shock wave
- Oblique shock wave
- Bow shock wave

Normal shockwave is generated perpendicular to shock medium (usually 90̊). When comes to oblique shock wave is generated at an angle to the direction of flow. Whereas bow shockwave occurs upstream of the front of the blunt object when the upstream flow velocity exceeds (Mac 1). The boundary over which the physical condition undergoes an abrupt change because of the shockwave is called the shock front. This is a small note of a shock wave. Continuing the boundary concept from the next paragraph [1].

A boundary layer is a thin layer of fluid in contact with a surface. Generally, the flow acting on the fluid of the boundary layer is subjected to shearing forces. The velocity that exists beyond the boundary layer is maximum to zero when the fluid is in contact with the solid surface. Boundary layers are normally having a thinner at the leading edge and are thicker toward the trailing edge of the aircraft. The fluid flow in the boundary layers is laminar at the upstream portion of the aircraft wing and turbulent at the downstream portion of the aircraft wing. Boundary layer interaction or Boundary layer flow separation is the process when the fluid (gas/air) comes in contact with a solid surface (e.g.: aircraft wing, rocket nose, re-entry vehicle nose, etc) the fluid produces the shockwave, the process also known as Shock wave boundary layer interaction (SBLI). Up to now learned the shock wave and boundary layer things can undergo the things of shock wave boundary layer interaction from the next paragraph.

Shock wave boundary layer interaction (SBLI) is a primary phenomenon of gas dynamics, aerodynamics. Generally, this is observed in some practical situations, which differ from transonic aircraft wings to hypersonic vehicles and engines. SBLI has the possibility to create serious problems in a flow field; consequently, proves that critical or even design limiting issues are raised for many aerospace applications. shock waves are present in a variety of engineering usage in environments, like transonic gas turbine blade tip gaps, transonic turbine blade passages, scramjet isolator ducts, supersonic aircraft engines, transonic and supersonic flight vehicle surfaces, and surfaces of rockets, missiles, and re-entry vehicles. Many proved a state-of-the-art explanation of this phenomenon. It ranges from the transonic, supersonic, and hypersonic velocities [2].

The flow fields where the fluid velocity is much larger than the velocity of propagation of small disturbances, the velocity of sound ($M<0.8$) is Subsonic flow, whereas the velocity of sound ($M=0.8$-1.2) is Transonic flow, whereas the velocity ($M>1.2$, $M<5$) is Supersonic flow, whereas the velocity of sound ($M \geq 5$) is Hypersonic flow.

The transonic assumption is based on the Mach number is the only important factor that plays a major role. This is the reason that potential theory has shown that smooth transonic (that is, subsonic-supersonic-subsonic) flow is possible with local Mach Numbers well in excess of unity. The exact range depends upon the critical Mach number. General issues caused by transonic airflow are unsteadiness occurrence and large-scale downstream separation. It also has another consequence that it causes a rapid increase in drag [3].

Air acts much more variety at supersonic speeds than it does at subsonic speeds. When an aircraft approaches the speed of sound, the airflow over the wing reaches supersonic speed before the airplane itself does and shock waveforms on the wing. The airflow beyond the shock waveforms on the wing. The airflow beyond the shock wave split up into a turbulent wake, increasing drag. When the airplane exceeds the speed of sound, a shock wave forms just the speed of sound, a shock waveform in the wing's leading edge is ahead. The shock wave formed on the wing in the trailing edge after the leading edge.

The von Kármán has pointed out that in many methods the dynamics of hypersonic flows is similar to Newton's corpuscular theory of aerodynamics.

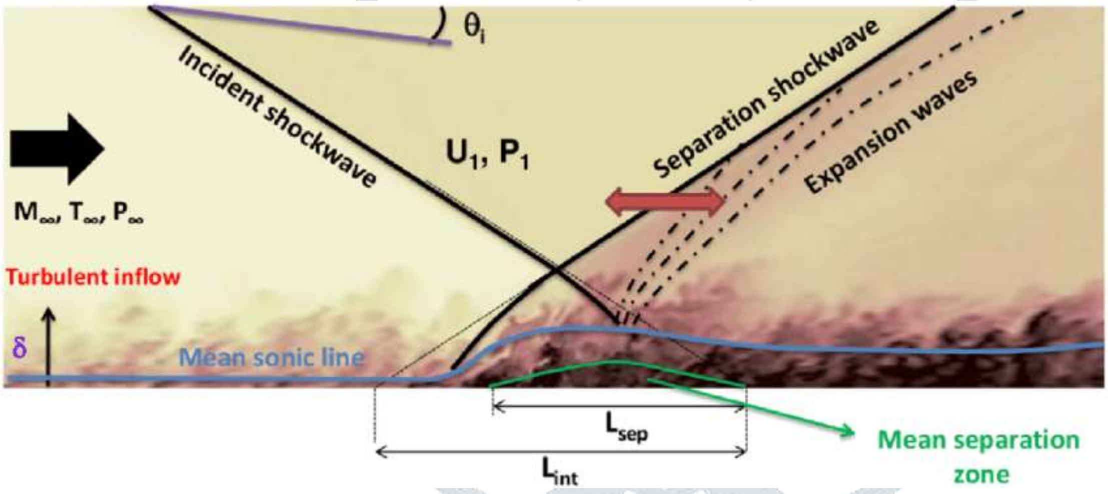
Hypersonic flow having the following characteristics;

1. Shock layer:
2. Aerodynamic heating

3. Entropy layer
4. Real gas effects
5. Low density effects
6. Independence of aerodynamic coefficients with Mach number.

It is the field of study of a very specific class of flows that develop around aerodynamic bodies moving in gases at exceedingly high velocities compared to the speed of the sound waves. The descriptive explanation of the gas environment surrounding a hypersonic vehicle is important for the calculation of thermomechanical loads on the body. These notes provide a high-quality characterization of hypersonic flows in terms of characteristic scales experienced in engineering applications. The wealth and characteristics of the gas-dynamic phenomenology emerging around hypersonic flight systems are summarized schematically and elaborated in the remainder of these notes [4].

This paper is about the Shock wave boundary layer interaction (SBLI) in different research conditions and models have done up to now from past 50 years of research not considering all the papers until now, only considering the ten different papers which have done different types of research in different models. In some papers, they have done CFD (Computerized fundamental dynamics) analyses and some of them only done numerical analysis and some have done experiments through supersonic and lens xx wind tunnels for different purposes to achieve accurate results.



Shock wave/ Turbulent boundary layer interaction on flat plate[5]

2. Detailed Review on Shock wave boundary layer interaction

It is an experiment conducting an existing shockwave boundary layer interaction having sufficient quality to guide turbulence modeling and code validation. By using CFD we find solutions for Hypersonic problems. Various boundary layer interaction experiments are to be done to known about the various properties in turbulent shock situations. These tests can't be done fully in-ground base situation. It should be done by using CFD methods. So, before installation of the body to be designed in a manner that could eradicate this vibrational disturbance. For this proper experimentation to be done. Here the analysis is taken from the Mac 3 then after the section, it is observed that Mac 3-5 is the operating range for supersonic and above Mach 5 is the operating range for Hypersonic shock interaction boundary conditions [6].

Collecting numerical values from large eddy simulation of shock boundary layer interaction to know the complex mechanisms which would play a major role while designing propulsion system. By using the CFD code validation of supersonic shock boundary interaction has been done. We observed the dynamic interaction of boundary is observed from the data. We also able to predict the wall pressure, temperature fluctuation, density profile, root mean square of velocity, Reynold's shear profile.

Comparing the flow of shock wave boundary layer interaction from numerical data to experimental data of 3D flow [7].

Considering some fundamental properties of the interaction are considered for a 2D adiabatic flow developing on the flat surface. Conducting an experiment to examine the following: upstream interaction length, incipient shock-induced separation, and evolution of the boundary layer properties. Here they considered both supersonic and transonic flows. Controlling the shock wave/boundary layer interaction is classified into two categories; those acting on the boundary layer properties before it enters the shock region and shock foot region. Conducting numerous experiments and analyze data to find various properties of the shock wave boundary layer while interacting. By using two methods of control techniques, it has been done [8].
Finding the better solution for a mathematical problem of impinging the shock wave and laminar boundary layer. By using the Prandtl-Meyer formula pressure is determined. Comparison of the experimental data and theoretical data to various properties of the boundary layer on the plate. They conclude that the compressible boundary layer equation has an approximate result for the shock wave laminar boundary layer equation. Theoretical data is tough to get due to the parabolic equations involved [9].

Shock wave boundary layer interaction is reviewed in four different areas: i) understanding low-frequency unsteadiness, ii) heat transfer prediction capability, iii) phenomena in complex (multi-shock boundary layer) interactions and iv) flow control techniques. Change the design to achieve maximum accuracy and conduct the experiment to find the various properties in mentioned areas accordingly. Here they achieve maximum accuracy for predicting and solving the problem through two-dimensional flow interaction and they can’t achieve in three-dimensional flow interaction due to RANS/LES methods have shown promising results. This can be achieved through numerical data and experimental data and by using CFD data validation has been done [10].

Conducting an experiment to find the glancing interaction between oblique shock wave and thermal boundary layer through a supersonic wind tunnel. It has two different viscous layers. Whereas, (i) the side-wall boundary layer growing along the flat surface; (ii) the induced layer originating on the shock-generator surface near the root and crossing the path of the wide-wall layer. Comparing the theoretical data with experimental data to achieve accuracy. It also provides information about various properties and characteristics of the flow which includes oil flow pictures, vapor and smoke-screen photographs, wall-pressure distributions, and local heat-transfer measurements. Conducting the experiment by mounting the wedge in the supersonic wind tunnel [11].

Conducting an experiment in the transonic wind tunnel to find the transonic flow of the shock wave pattern and pressure distribution of the boundary layer. And also find the change of flow from laminar to turbulent in the same Mach number while conducting the experiment. In this paper, they are conducting the experiment to find the various properties of the transonic flow while interacting between the boundary layer and shock wave [3].

Comparing the result of numerical solution to CFD solution for a given problem. Conclude that we also find an accurate solution using the numerical solution. Although it will take time to do the whole calculation gives the accurate solution in linear and non-linear problems. Although the solution can proceed at larger time steps, is the computation time per step correspondingly greater also [12].

Modifying the design to double cone configuration and conducted the experiment to find the various properties of flow stream through Lens XX tunnel. Free Stream has been selected through CFD calculations. Properties which include pressure, Mach number, and temperature of flow field should be predicted through this experiment. It also includes experiments on flow chemistry. It combined both experimental data and numerical to find an appropriate technique for the double cone configuration. By the experiment, we also find the chemical properties and reactions. It provides information effects of flow chemistry on the characteristics of the laminar region of shock boundary layer interaction. Provide exact data related to the model which they have taken. Design the double cone configuration and tested in Lens xx tunnel to find various things and by using CFD find some chemical characteristics [13].

A shock tunnel experiment has been conducted to study the interaction of boundary layer developed along with a rocket with a bow shock generated by a booster. Booster configurations were employed to change the strength of the bow shock. The distribution of heat flux and static pressure were measured along the rocket surface in order to examine the character of the interaction region and correlated both peak values. By the experiment, the three-dimensional shock boundary layer interaction flow field was also visualized by the oil flow method and schlieren photography. Hereby using this we understand the heat flux and static pressure [14].

3. Observations

They mentioned that the high-speed data collection and assessment effort create a few critical issues directly relevant to turbulence modeling. Their purpose in this effort is to define a database for the specific goal of the advancement of modern turbulence models, not to conduct a broad-based survey of all previous work in the field of hypersonic [6].

The present work focuses on the utilization of large-eddy simulations to the study the properties of an oblique shock interacting with a turbulent boundary layer above a flat plate. To provide more understanding into the computed results, the experimental data provided by previous research papers and study the unsteady aspects of the 3-D shock wave boundary layer interaction (SWBLI), with specific attention on the origin of the low-frequency oscillations associated with wall pressure fluctuations. It also stated addresses the inquiries of the three-dimensionality of the flow in the presence of sidewalls, and the possible effect of the spanwise confinement on the flow organization together with the associated low-frequency unsteadiness [7].

An attentive presentation is needed to some control techniques applied to shock wave boundary layer interaction on transonic aerofoils was published nearly a quarter of a century ago. However, the application of interaction control on airplanes was not seriously examined until the recent development of a new generation of large civil transport aircraft equipped with advanced supercritical profiles. Within the present low-budget context, reduction of the airplane drag may lead to a sizeable and perhaps decisive improvement of performance in terms of range, fuel volume/cost, or speed. Hence, we have a considerable renewal of interest in drag reduction techniques and, in this context, the reduction of the drag rises due to strong shock forming on aerofoils at off-design conditions as well as the increase of buffet boundary can be of vital importance [8].

The process of replacing the partial differential equations with finite-difference equations was chosen for finding a better solution. This method has good pliability in that it allows an arbitrary law for viscosity and heat conductivity and very general boundary conditions. In theory for a well-chosen implicit finite-difference estimation to the differential equations, the solution of the finite difference equations converges to the solution of the partial differential equations as the stop size of the limited-difference grid approaches zero [9].

It is essential to limit the scope of the paper to maintain completeness of the paper. Although recent improvements in diagnostic and computational tools have played an important role in recent research, these are not the focus of the paper. Rather, the primary emphasis of this paper is on the physics of SBLI, though some importance is given to the accuracy of numerical methods in the context of heat transfer rate prediction hence, this is a major thrust of some current research programs. The data is employed to illustrate salient points are taken from the work of the author and his collaborators mainly because of their ready availability. Transitional interactions are not mentioned although such SBLI interactions have been examined experimentally (for example near a blunt fin in a supersonic stream by Murphree et al26) and computationally (for example an impinging shock case by Teramoto27). Finally, in the shortness of time, many insightful efforts could not be included and some conclusions are stated without elaboration of some of the qualifications noted in the cited work: there is thus a necessary if the undesirable degree of generalization [10].

The present paper makes an effort to show the results from a wide range of variety of experimental techniques applied to a fairly large test-flow field (23 x 23 cm) at a free-stream Mach number of 2.3. (Some of the results are compared with those from pilot experiments made in n 6 x 6 cm intermittent tunnel at M = 2-4.) The measurements include oil-flow pictures, vapor and smoke-screen photographs, wall-pressure distributions, and local heat transfer. Our results, like Oskam's, show that separation on the sidewall does not always appear when the surface-flow deflection angle exceeds the inviscid-shock angle and hence McCabe's criterion for incipient separation is conservative. Our measurements can be interpreted by the double-viscous-layer flow field model outlined later [11].

The ‘investigations stated that the ultimate goal is the understanding of the formation of shock waves in transonic flow. The reasoning defines above led to the belief that boundary layer effects are of paramount importance in transonic flow phenomena. Hence, the interaction between the boundary layer and the shock wave was the object of the first investigations [3].

To illustrate the concepts of the previous section’s numerical solutions of the two-dimensional interaction of an oblique shock wave with a laminar boundary layer at a Mach number of 2.0 will be presented and compared with the experimental data of Heikkinen. Some characteristic features of this interaction are illustrated schematically Heikkinen’s data were chosen because the boundary layer is laminar throughout the interaction region and the experimental data include both plate surface pressure and skin friction. Considering two cases in the test, one in which the incident shock was not strong enough to cause flow separation, and one with a shock sufficiently strong to cause separation, were solved numerically [12].
The predictions of flow chemistry indicate that the levels of atomic oxygen and nitric oxide should be sufficient to obtain quantitative measurements over and in the shock interaction regions. Based on previous papers with LENS XX high enthalpy flows with larger models than are simulated here, the flow in the interaction region will stabilize within the test times available. A typical set of analysis with the DPLR code for a 5 km/s (16.5 kft/s) flow over this blunt double-cone configuration. The predictions for the pressure, Mach number, and temperature demonstrate that this configuration generates a well-defined separated interaction region with attached flows both upstream and downstream of the separated region [13].

The detailed flow mechanisms in the interacted region, which are different between the three shapes of the booster nose. In addition, a three-dimensional effect is noted by comparing the transverse changes in heat flux and static pressure. In the case of the three-dimensional interaction flow, the main body model is curved in the transverse direction. Heat flux depends not only on static pressure outside the boundary layer but also on the temperature gradient inside it. This characteristic produces three-dimensional effects in the interaction that is of interest in this study [14].

4. Future Enlargement

It should be a focus on the Mach and Reynold number instead of completely focuses on the Mach number. Further experimentation should be conducted on Reynolds number and also conducted on complex shapes (like double fins and cross-shock type interaction) [6]. Experimented in the more unsteadiness of 3 D flow in future although it focuses on more 2 D flow concepts and also done a comparison between computed analyses and numerical solutions [7]. The boundary layer interaction is experimented and proven the controls and 2 D flow is determined and more to find in further in 3 D flow should be carried [8]. The parabolic equations are achieved better results to find accurate solutions of shock wave boundary interaction in laminar conditions and further experimented on the different problems raised in the interaction of shock wave boundary layer [9]. Mainly focuses on the low-frequency unsteadiness of 2 D flow is experimented and it should be done on 3 D flow to study the unsteadiness. Experimented to study the properties of boundary layer interaction between glancing layer and turbulent layer and have future scope in the characteristics and properties should be known while conducting the experiments [10]. A change of boundary layer condition laminar to turbulent in transonic flow field has been done and need to determine the effects and properties of flow in the transonic flow field should be done [11]. Numerical solutions have been provided to shock wave boundary layer interaction problems and should consider as more scope find better solutions while using the CFD computerized methods to comparing the results and determining the accuracy of the solutions done in numerical and computed in laminar conditions [12]. Conducting the experiment to know different characteristics of three-dimensional flow has been proved and much more experiments should be done on different characteristics of flow and properties of the flow [13]. By using the blunted double cone experiment is conducted and blunted double wedge also expertise to done more experiments to achieve an accurate result [14].

5. Conclusion

The proper literature review has been done and shown in the paper. The concept of the shock wave, Boundary layer, types of flow fields, and Shock wave boundary layer interaction (SBLI) has mentioned clearly. Shock wave boundary layer interaction (SBLI) review based on different concepts of interaction in different boundary-layer conditions on different models and applications. Considering the numerical method to determine the concept of shock boundary layer interaction (SBLI) and comparing the result with CFD to provide the data that attain inefficient manner. And Conducting experiments through different wind tunnels (like LENS XX, transonic and supersonic wind tunnel) and comparing the result with CFD. Concluding that the CFD (Computational fluid dynamics) plays a major role in the different flow fields to determine the large eddy simulation and comparing the data with numerical analysis and experiment through different wind tunnels in different flow fields. And the shock boundary layer interaction future scope and observations are also clearly noted in this paper. Also concluding that the nose of the re-entry vehicle and rocket booster has been replaced with the blunted double cone or blunted double wedge to achieve a better interaction between shock wave and boundary layer to attain maximum efficiency.

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