Analysis of 2D Shock-Wave/Boundary-Layer Intraction Phenomena

M. Bordbar and Y.F. Yao

1 Introduction

Shock-wave/boundary-layer interactions (SWBLI) frequently occur in supersonic flows, the subject become increasingly important, especially for aerospace applications such as supersonic and hypersonic vehicles, which can have a significant influence on vehicles performance. The existence of shock wave boundary layer interaction can significantly alter the aerodynamic and thermodynamic loads in high speed flight. From the fundamental point of view, this problem involves the basic structure of shock interacting with spatially-developing boundary layer which represents one of the simplest occurrences of the strong viscous/inviscid interaction (see sketches in Fig. 1).

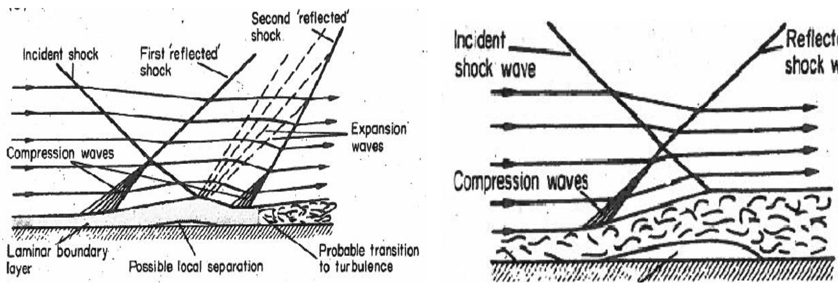

Fig. 1 Schematic view of shock-wave interacting with spatially-developing boundary-layer flow left: laminar boundary layer, right: turbulent boundary layer

The objective of this study is firstly to make some analysis of the shock wave and boundary layer interaction phenomena in two-dimensional laminar and turbulent flows and also boundary layer response to rapid pressure variation. The multi-layer structure or triple-deck boundary layer model has been considered to

M. Bordbar · Y.F. Yao
University of the West of England, Bristol BS16 1QY, UK

R. Bonazza and D. Ranjan (eds.), *29th International Symposium on Shock Waves 2*,
DOI: 10.1007/978-3-319-16838-8_66

analysis the penetration of incident shock-wave into non-viscous rotational layer. Finally several cases of shock/boundary layer interactions for flat plate have been selected to investigate the interaction characteristics between the oblique shock-wave and supersonic laminar/turbulent boundary layers, namely the wall pressure distribution and other factors such as shear force and wall heating that influencing an interaction.

2 Normal-Shock-Wave Interacting with Boundary Layer

There are three general possibilities when the near normal-shock-wave interacts with boundary layer. With a relatively weak shock, corresponding to an upstream Mach number just greater than unity, the diffused pressure rise may simply causes a gradual thickening of boundary layer ahead of shock with no transition and no separation. As seen in Fig. 2a, the gradual thickening causes a family of weak compression waves to develop ahead of the main shock. The compression waves join the main shock at some small distance from the surface, giving a diffused base to the shock. Immediately behind the shock, the boundary layer tends to thin out again and the local expansion takes place which bring a small region of mainstream up to slightly supersonic speed again and this is followed by another weak near-normal shock which develops in the same the initial shock. This process may be repeated several times before the main stream flow settle down to become entirely subsonic. At a slightly higher Mach number, the above pattern tends to change (see Fig 2b), the first shock becoming much stronger than the subsequent ones and all but one of the latter may well not occur at all for local upstream Mach numbers above 1.3. This is to be expected, because a strong first shock will produce a lower Mach number in the main stream behind it. This means that there is less likelihood of the stream regaining supersonic speed. Concurrently with this pattern change, the rate of thickening of the boundary layer, upstream of first major shock, become greater and the boundary layer at the base of the normal part of the shock will generally separate locally before reattaching. There is a considerable possibility that the transition to turbulence will occur behind the single subsidiary shock. With still greater local supersonic Mach numbers, the pressure rise at the shock may be sufficient to cause a separation of the laminar layer well ahead of the main shock position. This will result in the sharp change in the direction of the mainstream flow just outside the boundary layer and this will be accompanied by well-defined oblique shock which joins the main shock at some distance from the surface. This type of shock configuration is called a lambda shock-wave structure. It is unlikely that the boundary layer will reattach under these conditions, and the secondary shock which normally appears as result of re-attachment or boundary layer thinning, will not develop (see Fig. 2c). A turbulent boundary layer is far less susceptible to disturbance by an adverse pressure gradient than is a laminar layer, thus separation is not likely to occur for local mainstream Mach numbers, ahead of the shock, of less than 1.3 (this corresponds to a downstream to upstream pressure ratio of about 1.8). When no separation occurs, the thickening of the boundary layer ahead of the shock is rapid and the compression wavelets near the base of the main shock are much localized, so the base of the shock appears to be slightly diffused although no λ -shock formation is apparent. Pressure distribution for laminar and turbulent boundary layer along a flat plate shown in Figs. 3a and 3b respectively. It shows that the pressure distribution diffuses near the wall. Also that diffusion is much more pronounced for laminar than for turbulent boundary layer.

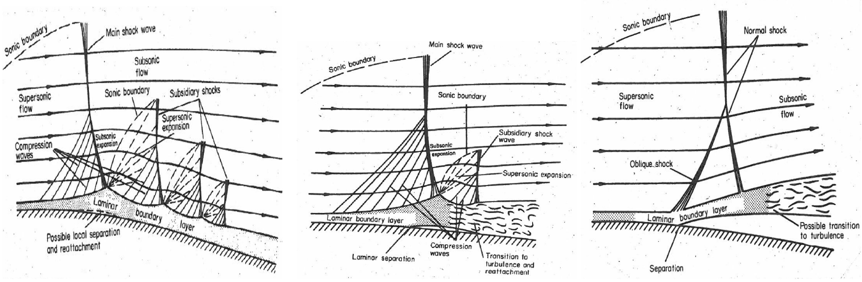

Fig. 2 Schematic view of near-normal shock interaction with laminar boundary layer

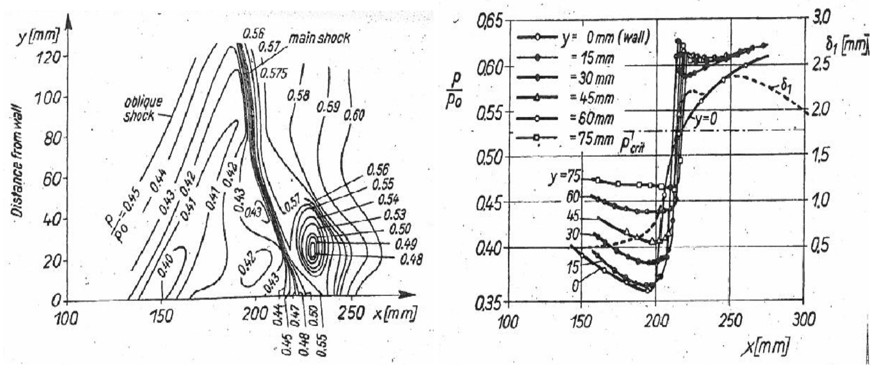

Fig. 3 Left: λ shock-wave laminar-boundary-layer interaction, right: pressure distribution at various distance from wall for shock-wave turbulent-boundary-layer interaction.

3 Oblique-Shock-Wave Interacting with Boundary Layer

In most practical cases, the shock is not perfectly normal; rather it is oblique shock which is produced if a supersonic flow encounters a wedge-type shape (e.g., at nozzle intake). Such shock will be at the angle, between upstream surface and itself, of considerably less than 90*o*. The general reaction of the boundary layer is similar to transonic flow, except that the oblique shock does not, in general, reduce the main stream flow to subsonic speed. If the boundary layer is turbulent, it appears to reflect the shock wave as a shock in much the same way as with the solid surface in the absence of boundary layer, although some thickening of the boundary layers occurs. There may also be local separation and reattachment, in which case the reflect shock originates just ahead of point of incidence. A laminar layer, however, thickens gradually up to the point of incidence, and may separate locally in this region, and then rapidly thins again, and the shock reflects as a fan of expansion waves, followed by diffused shock a little further downstream. There is additionally a set of the weak compression waves set up ahead of incident shock, due to the boundary layer thickening, but these do not usually set up a lambda configuration as with the near normal incident shock. Approximate representations of these cases are shown in Fig. 1. The pressure variation along a flat plate placed parallel to supersonic stream shown in Fig. 4a. The measurements were first performed by Liepmann et al. [1]. The pressure plots have been taken near the point on the flat plate where the oblique shock produced by a wedge interacted by the boundary layer. The pressure gradient is considerably steeper for turbulent than for the laminar boundary layer. The width of diffusion is equal to about 100δ in the case of interaction with a laminar boundary layer, but decrease to about 10δ for turbulent boundary layer (δ is boundary layer thickness). As the Reynolds number increases, the relative magnitude of viscous force decreases. Thus a laminar boundary layer is less resistant to the influence of the shock at high Reynolds numbers than at the lower values. In turbulent flows, the influence of shear forces is less obvious and the interaction depends generally only to a small degree, on the Reynolds number. This means that for a well-established turbulent regime, there is practically no influence of the Reynolds number in contrast to laminar interactions. In turbulent flow, the influence of Reynolds number is experienced mainly through the effect on the incompressible shape parameter (H); it determines the value of H at the interaction onset. The other important factor that influence the boundary layer within or upstream of the interaction region is the wall temperature T_W. The overall effect of wall cooling is to make the boundary layer more resistant, whereas wall heating makes it more fragile and provokes a dilation of interaction domain. Theoretical investigations of effect of wall heat transfer on laminar boundary layer and shock interaction were published in the paper by Curle [2]. This study contain an investigation of effect of heat transfer on effect of heat transfer on pressure rise on a flat plate as well as the description of approximate method of calculating the boundary layer for arbitrary wall temperatures and Prandtl numbers. This method takes advantage of experimental results of measurement performed by Gadd et al. [3]. Both theory and experiment lead to conclusion that the pressure ahead of the zone of separation is higher when the wall is heated than when it is adiabatic Fig. 4b.

4 Shock-Wave Penetration into Boundary Layer

The boundary layer triple deck terminology used to study the mechanism of penetration of shock wave into the boundary layer by Henderson [4]. The analysis was more pertinent to turbulent flow because in a turbulent boundary layer, the middle deck represents the greatest part of boundary layer even at the moderate Mach number, so that the behavior of an interaction can be described a perfect fluid model. Most of boundary layer behaves like an inviscid rotational flow, its entropy being constant along each streamline.

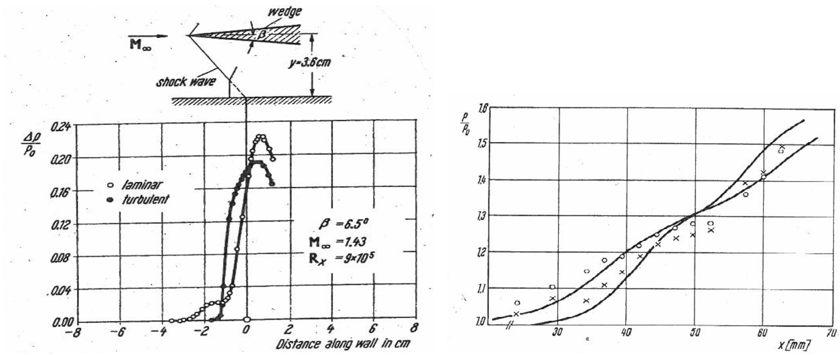

Fig. 4 Left: pressure distribution along a flat plate at supersonic velocity for laminar/turbulent boundary layers shock reflection, right: pressure distribution for laminar boundary layer on a flat plate at zero incident angle in supersonic flow at different wall temperatures T_W(circle: isothermal heated wall $T_W = 1.5T_\infty$; cross: adiabatic wall, lines are curve fitting using least square method)

Henderson split the boundary layer flow in to N-layers made of uniform and parallel constant pressure streams of different Mach numbers. Penetration of the incident shocks into the inviscid rotational layers results in a succession of transmitted shocks and expansion waves. As the upstream Mach number of layers decreases on approach to the wall, the angle of transmitted shock increases so that the incident shock bends and becomes steeper until it vanishes on approach to the sonic line Fig. 5a. The analytical formula proposed by Coles [5] represent a turbulent –boundary layer velocity profile considered for propagation of shock wave in a turbulent boundary layer illustrated by perfect-fluid calculations using the rotational method of characteristic. Calculation was made for turbulent velocity distribution with the outer Mach number equal to 4. The part of the boundary layer of which the Mach number is less than 1.8 was removed (in order to avoid singular shock reflection). The behavior of the viscous sub layer is neglected, which is justified for moderate shock strength at high Mach numbers. The calculation corresponds to the reflection on a rectilinear wall of the shock producing a downward deflection of -6 degrees in the outer irrotational stream. The characteristic mesh represented in Fig. 5b shows the bending of the shock through the rotational layer and the waves coming from the wall downstream of the reflection. The wall pressure distribution plotted in Fig. 5c. Shows that the pressure first jumps at the impact point to an intermediate value and then progressively reaches the constant level that corresponds to a shock reflection in a Mach 4 uniform flow. This behaviour, which is observed in a high-Mach –number flows, thus can be interpreted by inviscid arguments. At a lower Mach number, below 2.5, an overshoot is observed in the wall-pressure distribution, which cannot be explained simply by rotational effects. In these circumstances, the influence of the subsonic layer close to the wall as well as viscous inner layer can no longer be neglected, and purely inviscid analysis captures only part of the solution. The analysis are instructive because they do not consider the upstream transmission through the subsonic part of boundary layer with subsequent generation of compression waves, which coalesce to produce the reflected shock.

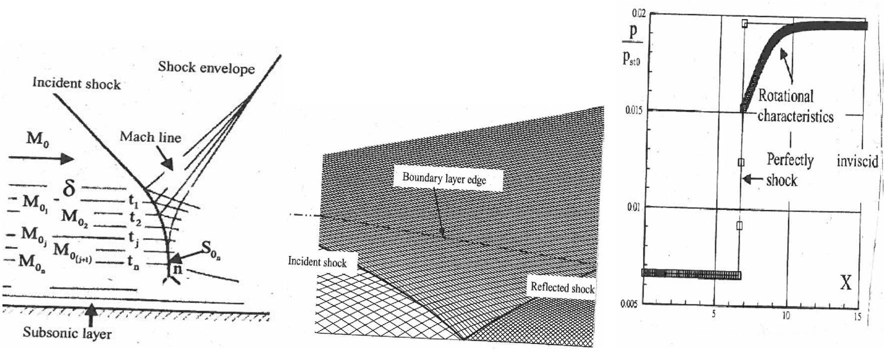

Fig. 5 Left: shock penetration into rotational layer, middle: characteristic mesh representation of characteristic calculation of shock reflection in rotational Shock reflection in rotational layer for turbulent layer, right: wall-pressure distribution using method of characteristic.

5 Conclusion

Schematic shock-wave/boundary-layer interactions involving normal and oblique shocks have been used to compare pressure distributions at various distance from the wall for laminar and turbulent flows with effects of different wall temperatures. Shock penetration in turbulent boundary layer is illustrated by perfect - fluid calculation using the rotational method of characteristics.

References

1. Liepmann, H.W., Roshko, A., Dhawam, S.: On reflection of shock waves from boundary layers. NACA Rep. 1100 (1952)
2. Curle, N.: The effect of heat transfer on the laminar boundary layer separation in supersonic flow. Aero.Quart. 12, 309–336 (1961)
3. Gadd, G.E., Holder, D.W., Regan, J.D.: An experimental investigation of the interaction between shock waves and boundary layers. Proc. Roy. Soc. A 226, 227–253 (1954)
4. Henderson, L.F.: The reflection of shock wave at a rigid will in the presence of a boundary layer. J. Fluid Mech. 30(4), 699–722 (1967)
5. Coles, D.E.: The law of the wake in turbulent boundary layer. J. Fluid Mech. 2, 191–226 (1956)