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ORIGINAL ARTICLE

Some physical aspects of shock wave/boundary layer interactions

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Abstract When the flow past a vehicle flying at high velocity becomes supersonic, shock waves form, caused either by a change in the slope of a surface, a downstream obstacle or a back pressure constraining the flow to become subsonic. In modern aerodynamics, one can cite a large number of circumstances where shock waves are present. The encounter of a shock wave with a boundary layer results in complex phenomena because of the rapid retardation of the boundary layer flow and the propagation of the shock in a multilayered structure. The consequence of shock wave/boundary layer interaction (SWBLI) are multiple and often critical for the vehicle or machine performance. The shock submits the boundary layer to an adverse pressure gradient which may strongly distort its velocity profile. At the same time, in turbulent flows, turbulence production is enhanced which amplifies the viscous dissipation leading to aggravated performance losses. In addition, shock-induced separation most often results in large unsteadiness which can damage the vehicle structure or, at least, severely limit its performance. The article first presents basic and well-established results on the physics of SWBLI corresponding to a description in terms of an average two-dimensional steady flow. Such a description allows apprehending the essential properties of SWBLIs and drawing the main features of the overall flow structure associated with SWBLI. Then, some emphasis is placed on unsteadiness in SWBLI which constitutes a salient feature of this phenomenon. In spite of their importance, fluctuations in SWBLI have been considered since a relatively recent date although they represent a domain which deserves a special attention because of its importance for a clear physical understanding of interactions and of its practical consequences as in aeroelasticity.

Keywords Shock wave/boundary layer interaction · Shock polar · Triple deck structure · Rotational flow · Separated flow · Shock-shock interference · Unsteadiness · Turbulence · Strouhal number

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List of symbols

(C)	Designates a shock
$E(f)$	Power spectral density
f	Frequency
h	Height of the separated bubble
L	Interaction length
M	Mach number
M_c	Convective Mach number
M_e	Mach number at the boundary layer outer edge
p	Pressure
p_{st}	Stagnation pressure
r	Density ratio
R	Designates the reattachment point
s	Velocity ratio
S	Designates the separation point
S_L	Strouhal number
T	Designates a triple point
U_D	Flow velocity on the separated flow dividing streamline
U_e	Flow velocity at the boundary layer outer edge
U_s	Shock displacement velocity

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X_0	Interaction origin
$\Phi(M_c)$	Normalized spreading rate of the mixing layer
φ	Shock induced deflection
δ	Boundary layer thickness
(Γ)	Designates a shock polar

1 General introduction

When the flow past a vehicle flying at high velocity becomes supersonic, shock waves inevitably form, caused either by a change in the slope of a surface, a downstream obstacle or a back pressure constraining the flow to become subsonic. In modern aerodynamics, one can cite a large number of circumstances where shock waves are present. On transport aircraft, a nearly normal shock terminates the supersonic region existing on the wing in certain flight conditions (see Fig. 1a). This transonic situation is also encountered in turbomachine cascades and on helicopter blades. Supersonic aircraft are much affected by shock waves which are of prime importance in air intakes whose purpose is to decelerate a supersonic incoming flow down to a low subsonic flow in the engine entrance section (see Fig. 1b). Intense shock phenomena also occur in over-expanded propulsive nozzles where a shock forms at the nozzle lip if the exit pressure is lower than the external pressure. For hypersonic vehicles, the high-temperature rise provoked by intense shocks influences the thermodynamic behaviour of air, causing the so-called real gas effects and their multiple repercussions on the vehicle aerodynamics (see Fig. 1c). In addition, strong interactions with the boundary layers are the origin of severe aero-heating problems if the shock is strong enough to provoke separation. Shocks are also met on missiles and aircraft afterbodies, as shown in Fig. 1d, as well as in space launcher nozzles and on projectiles of all kinds.

The encounter of a shock wave with a boundary layer results in complex phenomena because of the rapid retardation of the boundary layer flow and the propagation of the shock in a multilayered structure. The consequences of shock wave/boundary layer interaction (SWBLI) are multiple and often critical for the vehicle or machine performance. The shock submits the boundary layer to an adverse pressure gradient which may strongly distort its velocity profile. At the same time, in turbulent flows, turbulence production is enhanced which amplifies the viscous dissipation leading to aggravated efficiency loss in internal flow machines or substantial drag rise for profiles and wings. This interaction, felt through a coupling between the boundary layer flow and the contiguous inviscid stream, can greatly affect the flow past a transonic airfoil or inside an air-intake. The foregoing consequences are exacerbated when the shock is strong enough to separate the boundary layer. The consequence can be a dramatic change of the entire flow field structure with the formation of intense vortices and complex shock patterns replacing the simple purely inviscid flow structure. In addition, shock-induced separation most often results in unsteadiness, damaging the vehicle structure and limiting its performance.

In some respect, shock-induced separation can be viewed as the compressible facet of the ubiquitous separation phenomenon, the shock being an epiphenomenon. Indeed, the behaviour of the separating boundary layer is basically the same as in incompressible separation and the overall flow topology is identical. Perhaps, the most distinctive and salient feature of shock-separated flows is the accompanying shock patterns forming in the contiguous inviscid flow, whose existence may have major consequences on the entire flow field. It is difficult to completely separate SWBLI and phenomena induced by the crossing of shock waves, designated by the generic term shock-shock interferences.

Shock wave/boundary layer interaction is the result of a close coupling between the boundary layer, which is submitted at the shock foot or shock impact point to a sudden retardation and the outer, mostly inviscid supersonic flow. The clear understanding of this process necessitates a close analysis of both the inviscid flow and the boundary layer behaviours. A large number of studies have been devoted to SWBLI since the first investigation of transonic flows in the early 40s (for a review see Délery and Marvin [1]). The forthcoming sections are devoted to a reminder of some basic and well-established results on the physics of SWBLI corresponding to a description in terms of an average two-dimensional steady flow. Such a description allows apprehending the essential properties of SWBLIs and drawing the main features of the overall flow structure associated with SWBLI. In Sect. 6 emphasis is placed on unsteadiness in SWBLI which constitutes a salient feature of this phenomenon. In spite of their importance, fluctuations in SWBLI have been considered since a relatively recent date although they represent a domain which deserves a special attention because of its importance for a clear physical understanding of interactions and of its practical consequences as in aeroelasticity.

2 The basic shock wave/boundary layer interaction

What can be considered as the four basic interactions between a shock wave and a boundary layer, in two-dimensional flows, are the impinging–reflecting shock, the ramp flow, the normal shock, and the pressure jump.

In the oblique shock reflection, the incoming supersonic flow of Mach number M_1 undergoes a deflection φ_1 through the incident shock (C_1) and the necessity for the downstream flow to be again parallel to the wall (Euler type, or slip boundary condition for a non-viscous fluid) entails the formation of a reflected shock (C_2), the deflection φ_2 across (C_2) being such that $\varphi_2 = -\varphi_1$ (see Fig. 2a). Such a shock occurs inside

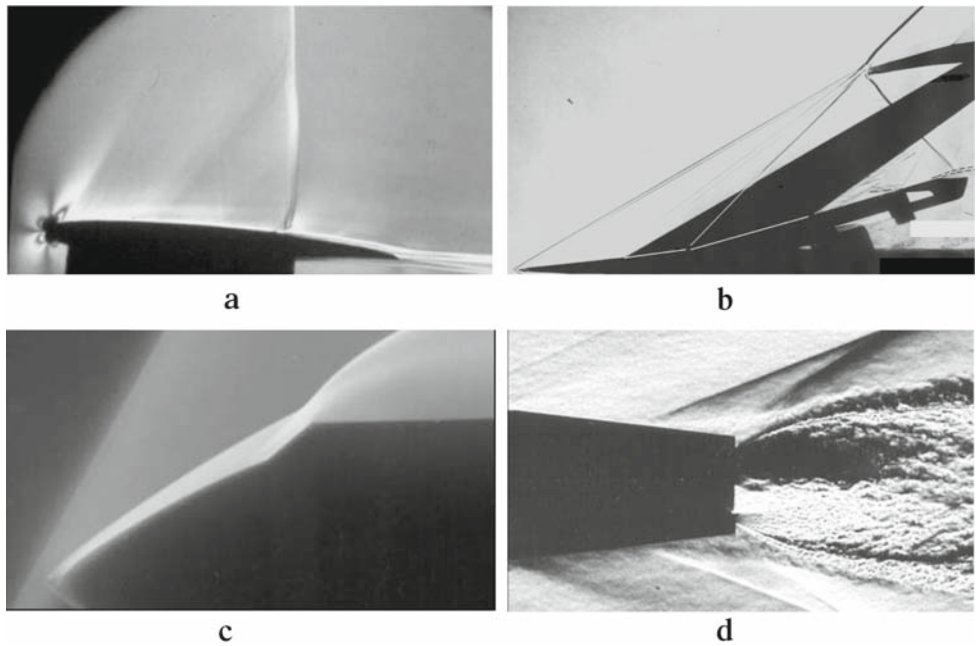


Fig. 1 Examples of shock wave formation in high speed flows (Onera documents). **a** transonic profile, **b** supersonic air-intake, **c** hypersonic vehicle, **d** afterbody with under-expanded nozzle

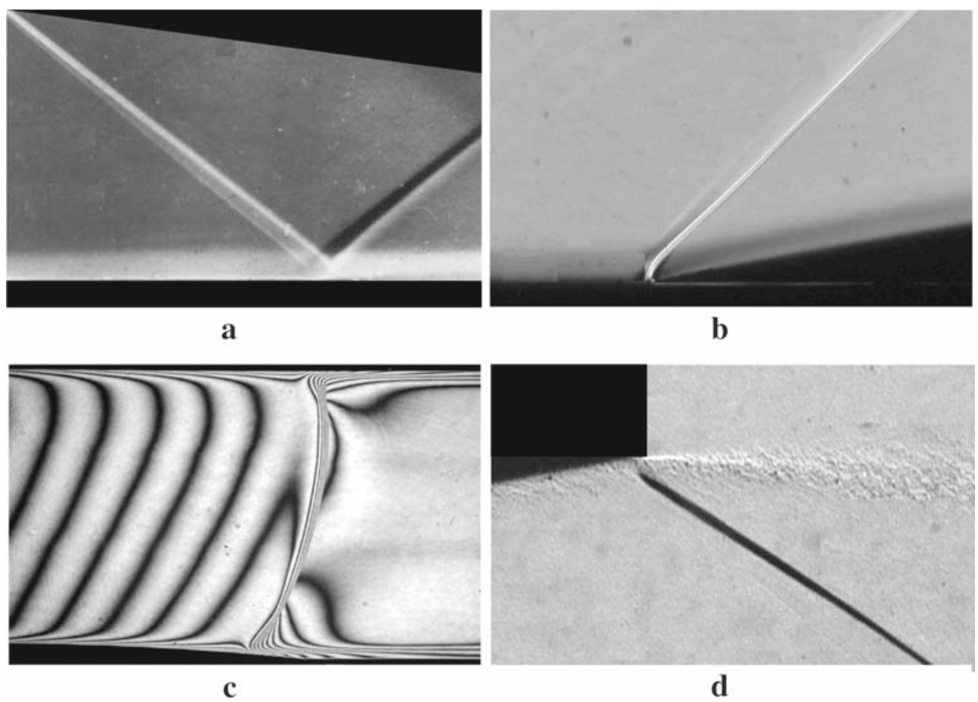


Fig. 2 Basic shock wave/boundary layer interactions (Onera documents). **a** oblique shock reflection, **b** ramp induced shock wave, **c** normal shock wave, **d** adaptation shock at a nozzle exit

a supersonic air-intake of the mixed compression type or at the impact of the shock generated by any obstacle on a nearby surface.

1. In the ramp flow, a discontinuous change in the wall inclination is the origin of a shock through which the incoming flow undergoes a deflection φ_1 equal to the wedge angle α (see Fig. 2b). Such a shock occurs at a supersonic air-intake compression ramp, at a control surface or at any change in the direction of a surface.
2. A normal shock wave is produced in a supersonic flow by a back pressure forcing the flow to become subsonic. In channel flow, a normal shock is also formed when a downstream choking necessitates a stagnation pressure loss in order to satisfy mass conservation (see Fig. 2c). The distinctive feature of a normal shock is to decelerate the flow without imparting a deflection to the velocity vector, the Mach number behind the shock being subsonic. However, in most practical cases, the shock is not perfectly normal, the situation corresponding to the

strong oblique shock solution to the Rankine-Hugoniot equations, even if the shock intensity is very weak as in transonic flows! In such situations, the velocity deflection through the shock is small so that the shock is said to be normal. Normal, or nearly normal, shocks are met in channel flows (turbomachine cascades, air intakes, supersonic diffusers), in shock tubes and over transonic profiles where a nearly normal shock terminates the supersonic pocket. Such interactions where the downstream flow is totally (or partly) subsonic are of special interest and lead to specific problems because of the possibility for downstream disturbances to influence the shock and initiate an interactive process at the origin of large-scale unsteadiness involving the whole flow, as in transonic buffeting or air-intake buzz.

3. An oblique shock is produced if a supersonic flow encounters a change in pressure as at the exit of an over-expanded nozzle (see Fig. 2d). In the present situation, the pressure discontinuity induces a flow deflection, whereas in cases 1 and 2, the pressure discontinuity is induced by a deflection. This is the mirror problem of the duality [deflection, pressure jump].

As far as the response of the boundary layer to the shock is concerned, there are no basic differences between the above situations, except perhaps case 4 where the interacting flow communicates with an atmosphere. So we will not distinguish different cases when discussing the viscous flow behaviour in the forthcoming sections. The major distinctions are between interactions without and with separation.

3 The boundary layer's response to a rapid pressure variation

The flow along a solid surface can be viewed as a structure composed of three layers (see the sketch in Fig. 3) as follows:

1. An outer inviscid layer which usually is irrotational (i.e., isentropic) and hence obeys the Euler equations or alternatives such as the potential equation. However, there are exceptions where this part of the flow is rotational as, for example, downstream of the curved shock formed ahead of a blunted leading edge, where what is referred to as an *entropy layer* is formed. A similar rotational layer can occur behind the near-normal but curved shock that forms on a transonic aerofoil.
2. Closer to the surface, and deeper within the boundary layer we come first to an outer portion where, over a streamwise distance of several boundary layer thicknesses, the flow can be considered as inviscid but rotational. In this part of the flow, viscosity contributes to create entropy and consequently vorticity, in agreement with Crocco's equation connecting the gradient of entropy $\overrightarrow{\mathrm{grad}}\, s$ with the rotational vector $\overrightarrow{\mathrm{rot}}\,\vec{V}$ in a steady non-viscous flow:

$$T\,\overrightarrow{\mathrm{grad}}\, s = -\vec{V} \times \overrightarrow{\mathrm{rot}}\,\vec{V}$$

More simply said, this layer is a region of variable stagnation pressure, the stagnation temperature being nearly constant. Although varying across a boundary layer, because of viscous effects, the stagnation enthalpy is almost constant for adiabatic walls, especially for turbulent boundary layers. This is even more true in the intermediate inviscid rotational layer. As the flow is considered inviscid, the stagnation conditions are constant along streamlines, since entropy is a transported quantity. The static pressure is constant across the boundary layer and hence the layer behaves like an inviscid flow through which the velocity, and hence Mach number, decreases steadily from the outer value M_e at the boundary layer edge ($y = \delta$) towards zero at the wall.

3. The third layer is in contact with the wall and is to insure the transition between the previous region and the surface; there, viscosity has again to play a role. This viscous layer must be introduced to avoid inconsistencies since it is not possible for a non-viscous flow to decrease its velocity without a rise in the static pressure and at the wall the stagnation pressure is equal to the static pressure (the velocity being equal to zero because of the no-slip condition).

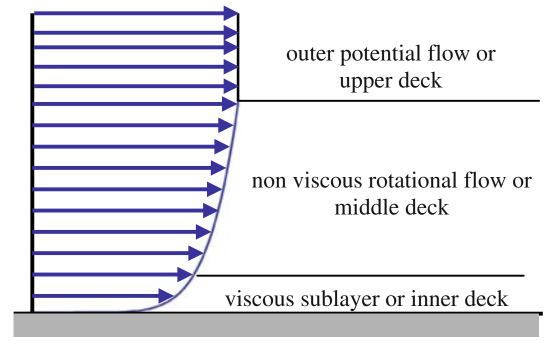


Fig. 3 The interacting flow multi-layer structure or triple deck (Lighthill, Stewartson–Williams)

The structure that is described above was first suggested by Lighthill [2]. A more formal justification was proposed in 1969 by Stewartson and Williams [3] for the case of a laminar boundary layer using an asymptotic expansion approach. They introduced the *triple-deck* terminology to designate such a structure. The outer deck is the outer irrotational flow, the middle deck the inviscid rotational layer, and the inner deck the viscous layer in contact with the wall. It should be made clear that such a representation is valid only if the viscous forces have not contributed to modify the entropy level of the boundary layer streamlines (except in the inner deck). This implies that the time scale of any phenomena considered with this approach is short compared to the time scale over which the viscous terms take affect. This is the case for SWBLIs where the shock imparts a sudden retardation to the flow. Such a model is also valid for a rapid acceleration as in the centred expansion wave that can occur at the base of a vehicle. In a turbulent boundary layer, the middle deck represents the greatest part of the boundary layer, even at moderate Mach number, so that the behaviour of an interaction can be for the most part described by considering a perfect fluid model. But, for the reasons cited above, such an inviscid model becomes inadequate close to the wall where viscosity has to be taken into account.

During the first part of an SWBLI, most of the flow, including a greater part of the boundary layer, behaves as an inviscid flow for which the pressure and inertia terms of the Navier-Stokes equations are predominant compared to the viscous terms. Thus, many aspects of the boundary layer response can be interpreted with perfect fluid arguments and by considering the boundary layer mean properties defined above. Such a description of the boundary layer behaviour calls upon the concept of rapid interaction. This is justified by the fact that in an SWBLI important changes occur over a short streamwise distance, the extent of the interaction being of the order of 10 times the boundary layer thickness δ for the laminar flow and much less in the turbulent case. This fact has several major consequences:

- Streamwise derivatives are comparable to derivatives in the direction normal to the wall, whereas in a classical boundary layer they are considered to be of lower order. This fact also influences the mechanism for turbulence production since the normal components of the Reynolds stress tensor may now play a role comparable to that of the turbulent shear stress, which, in general, is the only quantity considered.
- The turbulent Reynolds stresses do not react instantly to changes in the mean flow that are imparted via the pressure gradient. In the first phases of the interaction, there is a lag in the response of the turbulence and hence a disconnection occurs between the mean velocity and turbulent fields. Reciprocally, the velocity field is weakly affected by the shear stress, the action of viscosity being confined to a thin layer in contact with the wall. Thereafter, the turbulence level increases and can reach very high levels if separation occurs. This explains the difficulty in devising adequate turbulence models for SWBLIs, especially for the inception part of the process.

An inviscid fluid analysis provides a way to explain some basic features of an interaction, but it is not entirely correct, in the sense that viscous forces cannot be neglected over the entire extent of the interaction. Viscous terms must be retained in the region in contact with the wall; otherwise, one is confronted with inconsistency as said above. Nevertheless, the neglect of viscosity is justifiable in describing the penetration of the shock into the boundary layer. But there are a number of situations where the shock does not penetrate into the boundary layer, as in transonic interactions (except if the shock is very weak) or in shock-induced interaction (except at very high Mach number). In these cases, viscosity may have sufficient time to influence the flow behaviour even outside the near wall region. This is also true for interactions with large separated regions where the flow depends on its viscous properties to determine the longitudinal extent of the interaction.

In what follows, for conciseness we will concentrate our attention on the interaction induced by the impact of an oblique shock on a surface. It is clear that most of the conclusions could be applied *mutatis mutandis* to other kinds of shock interactions (see Fig. 2).

4 Interactions without separation—weakly interacting flows: the incident reflecting shock case

4.1 Overall flow organisation

The interaction resulting from the reflection of an oblique shock wave from a turbulent boundary layer is illustrated by the schlieren visualisation in Fig. 4. A similar structure would be seen for a laminar boundary layer, but the streamwise extent of the interaction domain would be greater. (The apparent thickening of the incident shock is due to its interaction with the boundary layers on the test section side windows;

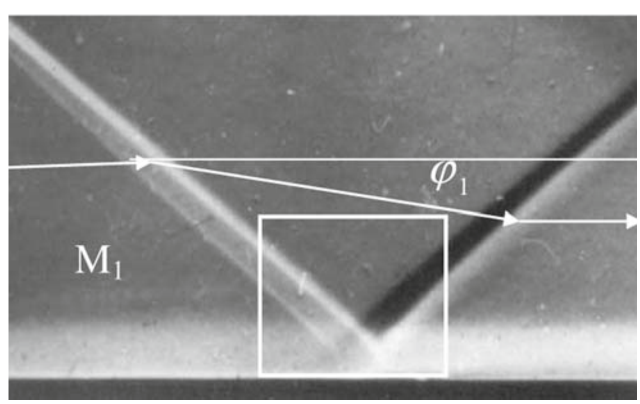


Fig. 4 Schlieren photograph of a shock reflection at Mach 1.95 (Onera document)

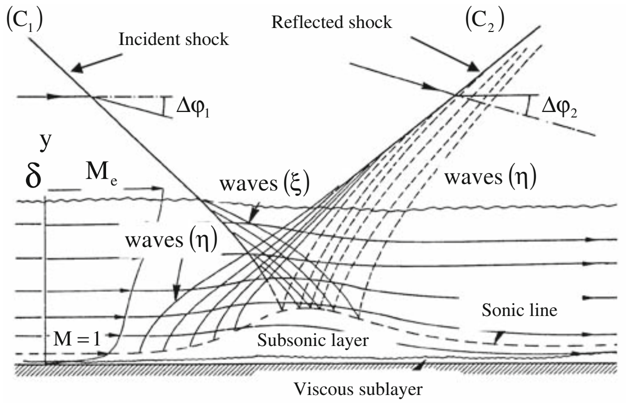


Fig. 5 A sketch of a turbulent shock reflection without boundary layer separation

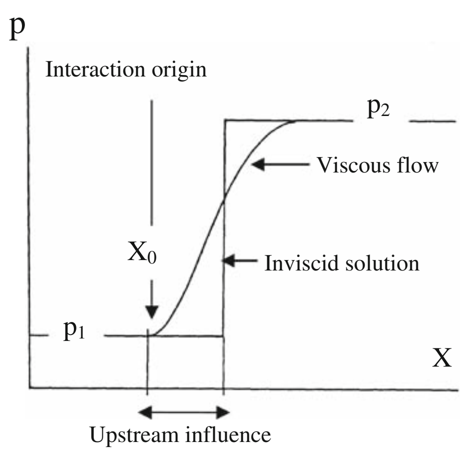


Fig. 6 The corresponding pressure distribution

its true location is indicated by the sharp deflection in the superimposed streamline).

The flow field organisation is sketched in Fig. 5. The incident shock (C_1) can be seen penetrating into the rotational inviscid part of the boundary layer where it progressively bends because of the local Mach number decrease. Correspondingly, its intensity weakens and it vanishes altogether when it reaches the boundary layer sonic line. At the same time, the pressure rise through (C_1) is felt upstream of where the incident shock would have impacted with the wall in the absence of a boundary layer. This upstream influence phenomenon is predominantly an inviscid mechanism, the pressure rise caused by the shock being transmitted upstream through the subsonic part of the boundary layer. This leads to a spreading of the wall pressure distribution over a distance of the order of the boundary layer thickness, compared with the purely inviscid flow solution. As shown in Fig. 6, the pressure starts to rise upstream of the inviscid pressure jump, after which it steadily increases and tends towards the downstream inviscid level. In this case, the viscous, or real, solution does not depart far from the purely inviscid solution. Accounting for the viscous effect would be a mere correction to a solution that is already close to reality. Such behaviour is said to be a *weak interaction* process in the sense that the flow is weakly affected by viscous effects. The dilatation of the boundary layer subsonic region is felt by the outer supersonic flow, which constitutes the major part of the boundary layer if the flow is turbulent. It acts like a ramp inducing compression waves (η) that coalesce to form the reflected shock (C_2). The thickness of the subsonic layer depends on the velocity distribution and hence a fuller profile, which has a thinner subsonic channel, also has a shorter upstream influence length. In addition, a boundary layer profile with a small velocity deficit has a higher momentum, hence a greater resistance to the retardation imparted by an adverse pressure gradient.

4.2 Shock penetration in a rotational layer

The propagation of a shock wave in a turbulent boundary layer is here illustrated by perfect fluid calculations using the rotational method of characteristic. This provides both high accuracy (the shock being fitted) and a picture of the wave's propagation in the supersonic flows. Calculations were made for a turbulent velocity distribution represented by the Coles [4] analytical expression, the outer Mach number being equal to 4. The part of the boundary layer whose Mach number is less than 1.8 has been removed (this cut-off distance from the wall was chosen to avoid singular shock reflection). The behaviour of the viscous sub layer is neglected, which is justified for moderate shock strengths at high Mach number. The calculation corresponds to the reflection on a rectilinear wall of a shock producing a downward deflection of $-6°$ in the outer irrotational stream. The characteristic mesh represented in Fig. 7 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. 8 shows that the pressure first jumps at the impact point to an intermediate value and then progressively reaches the constant level corresponding to shock reflection in a Mach 4 uniform flow. This behaviour, which is observed in high Mach number flows, can thus be interpreted by inviscid arguments. At lower Mach number, below 2.5, an overshoot is observed in the wall pressure distributions, which cannot be explained simply by rotational effects. In these circumstances, the influence of the subsonic layer close to the wall and also the viscous inner layer can no longer be neglected and a purely inviscid analysis captures only a part of the solution. The contours of Fig. 9 confirm that behind the shock, there is a static pressure decrease from the outer flow down to the wall. The inviscid analysis proposed by Henderson [5] and the method of characteristics calculations are instructive since they give a

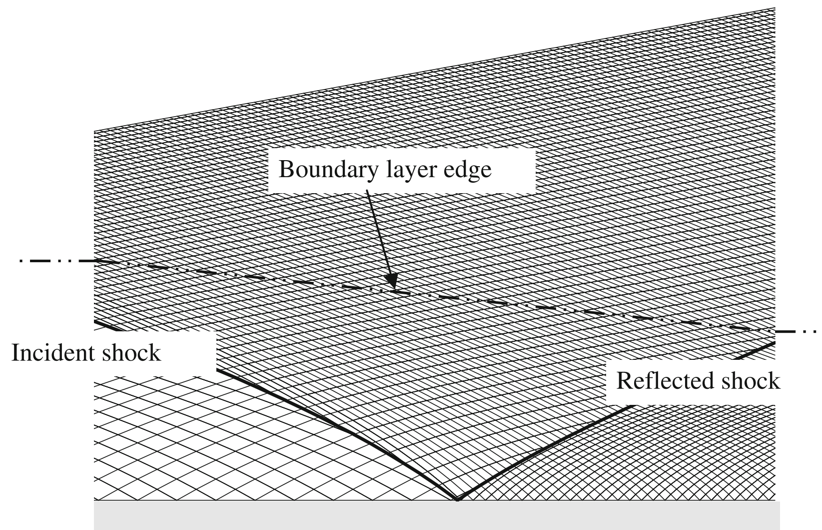


Fig. 7 Method of characteristic calculation of a shock reflection in a rotational layer. Wave system and shocks. Turbulent boundary layer profile (upstream Mach number 4, primary deflection −6°)

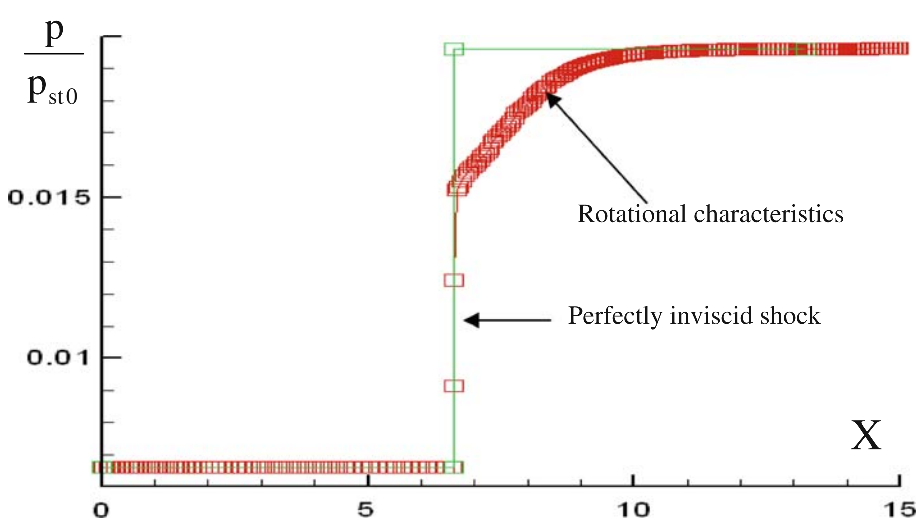


Fig. 8 Method of characteristic calculation of a shock reflection in a rotational layer. Entropy gradient effect on the wall pressure distribution (upstream Mach number 4- primary deflection 6°)

description of the complex wave pattern which is generated when a shock traverses a boundary layer considered as a rotational inviscid stream. However, the above scenario does not take into account the upstream transmission through the subsonic part of the boundary layer with the subsequent generation of compression waves, which coalesce to produce the reflected shock.

Fig. 9 Method of characteristic calculation of a shock reflection in a rotational layer. Static pressure contour. Turbulent boundary layer profile (Upstream Mach number 4, primary deflection −6°)

5 Interaction with separation—strongly interacting flows: the incident reflecting shock case

5.1 Overall flow organisation

A boundary layer is a flow within which the stagnation pressure decreases when approaching the wall and where, at least for short distances, it can be considered constant along each streamline. Neglecting compressibility (which is of course an over simplification) we can write the Bernoulli equation for each streamline:

$$p_{st} = p + \frac{\rho}{2}V^2$$

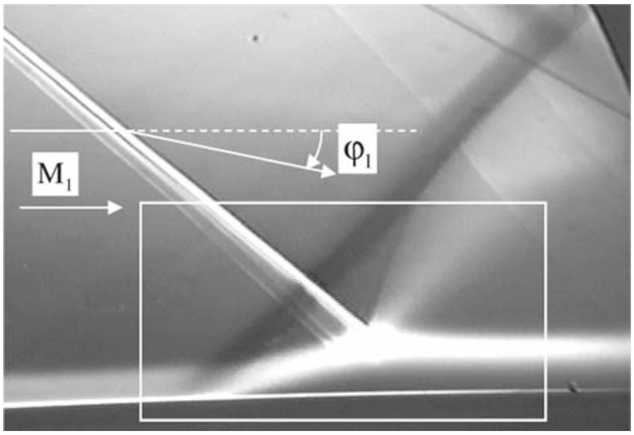


Fig. 10 Schlieren visualisation of an incident-reflecting shock at Mach 2 (Onera document)

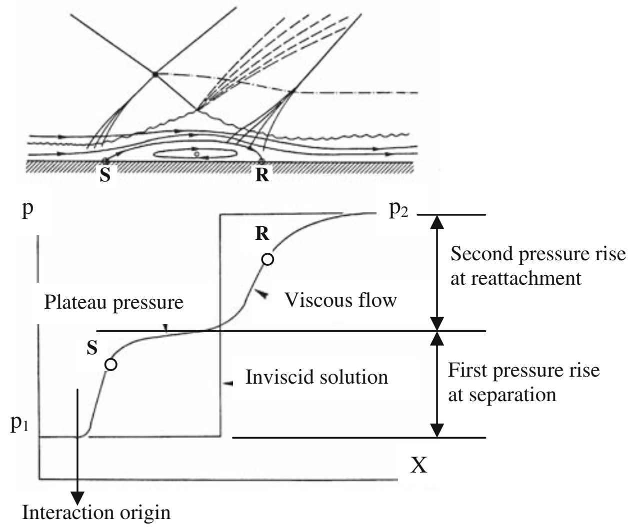
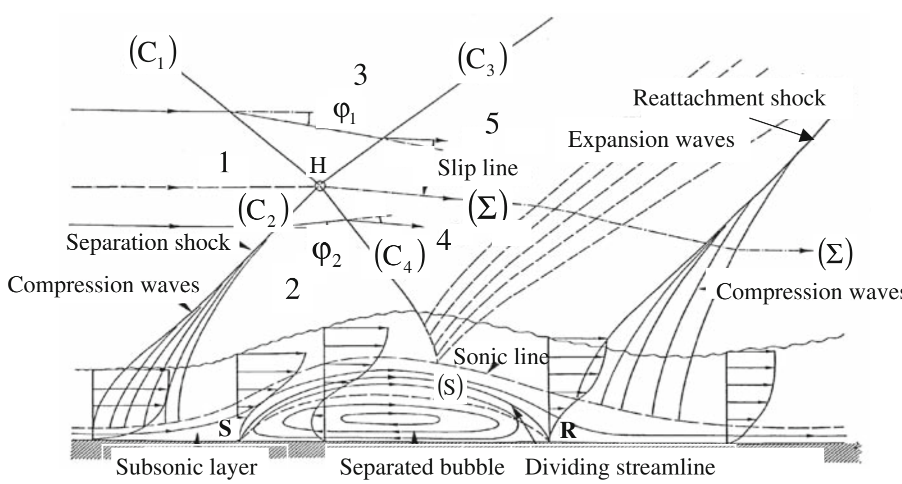


Fig. 12 Wall pressure distribution, in a shock separated flow

Thus any rise in p will provoke a greater retardation in regions where the stagnation pressure p_{st} is lowest, that is, in the boundary layer inner part. By imposing an adverse pressure gradient a situation can be reached where the flow adjacent to the wall is stagnated or reversed so that a separated region forms. An incident shock wave can readily induce separation this way as, for example, in the Mach 2 flow for which a schlieren picture is presented in Fig. 10 (The apparent thickness of the shock waves is due to the interactions taking place on the test section side windows). The structure of this flow is sketched in Fig. 11. Downstream of the separation point **S** there exists a recirculating 'bubble' flow bounded by a dividing streamline (S), which separates the recirculating flow from the flow streaming from upstream to downstream "infinity". The streamline (S) originates at the separation point **S** and ends at the reattachment point **R**. Due to the action of the strong mixing taking place in the detached shear layer emanating from **S**, a mechanical energy transfer takes place from the outer high-speed flow towards the separated region. As a consequence, the velocity U_D on the dividing streamline (S) steadily increases, until the deceleration associated with the reattachment process starts.

The transmitted shock (C_4) penetrates into the separated viscous flow where it is reflected as an expansion wave because there is near constant pressure level within the bubble. This causes a deflection of the shear layer towards the wall where it eventually reattaches at **R**. At this point, the separation bubble vanishes and the flow on (S) is decelerated until it stagnates at **R**. This process is accompanied by a sequence of compression waves that coalesce into a reattachment shock in the outer stream. As shown in Fig. 12, the wall pressure distribution exhibits initially a steep rise, associated with separation, followed by a plateau typical of separated flows. A second more progressive pressure rise takes place during reattachment. In this situation, the flow field structure is markedly different from what it would be for

Fig. 11 Sketch of the flow induced by a shock reflection with separation



the purely inviscid case and the shock reflection is said to be a *strong viscous-inviscid interaction*. This means that the viscous effects have to be taken fully into account when predicting the flow. They are no longer a simple adjustment to an already nearly-correct inviscid solution, but they play a central role in the establishment of the solution. It is evident that there has been a hierarchy reversal.

The *free interaction theory* [6] establishes that the pressure rise at separation and the extent of the first part of the interaction depend only on the flow properties at the interaction onset and not on the downstream conditions, in particular the shock intensity. During the first part of the interaction, the flow is a consequence of the reciprocal and mutual influence, or coupling, between the local boundary layer and the inviscid contiguous stream, and not the further development of the interaction. This important result, well verified by experiment, explains many features of interactions with shock induced separation.

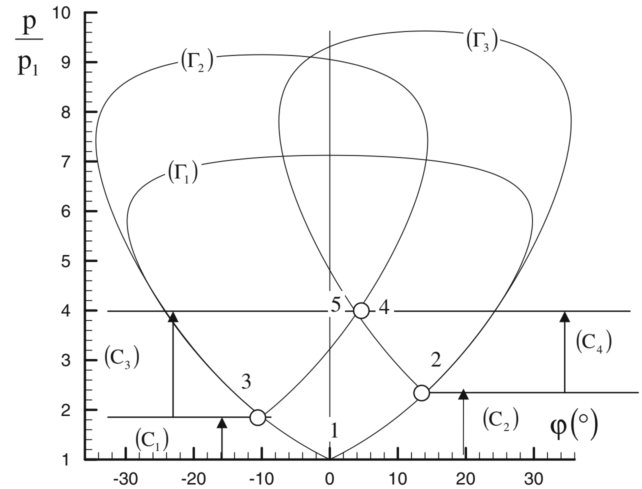


Fig. 13 Shock pattern interpretation in the shock polar diagram. Upstream Mach number 2. Separation shock deflection 14°; incident shock deflection −10°

5.2 The outer inviscid flow structure

The pressure rise at separation generates compression waves which coalesce to constitute the separation shock (C_2). This shock intersects the incident shock (C_1) at point **H** where (C_1) undergoes a deflection (refraction) to become the shock (C_4); the separation shock (C_2) becomes in a similar way the shock (C_3). The shock (C_4) meets the separated region boundary at point **I**. There, to insure continuity of pressure, the pressure rise produced by (C_4) must be compensated for by a centred expansion emanating from **I**. This expansion provokes a deflection of the separated region boundary which is turned towards the wall such that it impacts with it at the reattachment point **R**. There a new deflection occurs with formation of the reattachment shock (C_5). In addition, a slip line emanates from the intersection point **H**. For this case, the two-shock system of the perfect-fluid oblique shock reflection, that comprises simply an incident plus reflected shock, is replaced by a pattern involving five shock waves.

The pattern, made by the shocks (C_1), (C_2), (C_3) and (C_4) is a Type I shock/shock interference according to Edney's classification [7], which can be best understood by considering the shock polar representation in Fig. 13. The figure corresponds to an incoming uniform flow of Mach number 2.5. The separation shock deflection is given by an adequate turbulent separation criterion, which fixes the Mach number behind the separation shock [8]. This angle, which is here around 14°, does not depend on the intensity of the shock having caused the separation. The polar (Γ_1) is associated with the upstream uniform state 1 and represents any shock forming in 1, in particular, the incident shock (C_1). The image of the downstream flow 3 is the point 3 on (Γ_1), the deflection imparted by (C_1) being negative (the velocity is deflected towards the wall). The separation shock (C_2) is also represented by (Γ_1) since the upstream state is 1. The image of the downstream flow 2 is at point 2 on (Γ_1), the deflection φ_2 being upward. The situation downstream of **H** is at the intersection of the polars (Γ_3) and (Γ_2) attached to the states 3 and 2, respectively. Their intersection is the image of two states 4 and 5 having the same pressure ($p_4 = p_5$) and the same direction ($\varphi_4 = \varphi_5$), hence compatible with the Rankine-Hugoniot equations. The set of successive shocks $(C_1) + (C_3)$ is different from the set $(C_2) + (C_4)$ and hence the flows that have traversed each set have undergone different entropy rises. Thus a slip line (Σ) is formed separating flows 4 and 5 which have different velocities, densities, temperatures, and Mach numbers (but identical pressures). In a real flow a shear layer develops along (Σ) insuring a continuous variation of the flow properties between states 4 and 5. The fluid that flows along a streamline passing under the point **H**, and belonging to the inviscid part of the field, crosses three shock waves: (C_2) and (C_4), plus the reattachment shock (C_5). Thus, its final entropy level is lower than for the entirely inviscid case where the fluid would have only traversed the incident plus reflected shocks. This is also the case for an interaction without separation, which is close to the inviscid model at some distance from the wall. The conclusion is that, in an interaction with shock induced separation, entropy production is smaller than in a non-separated interaction, or in the limiting case of the inviscid model. This result is exploited by control techniques aiming at reducing wing drag or efficiency losses in internal flows [9].

If, for a fixed upstream Mach number, the strength of the incident shock is increased, a situation is reached where the two polars (Γ_2) and (Γ_3) do not intersect. Then a Edney Type II interference occurs at the crossing of shocks (C_1)

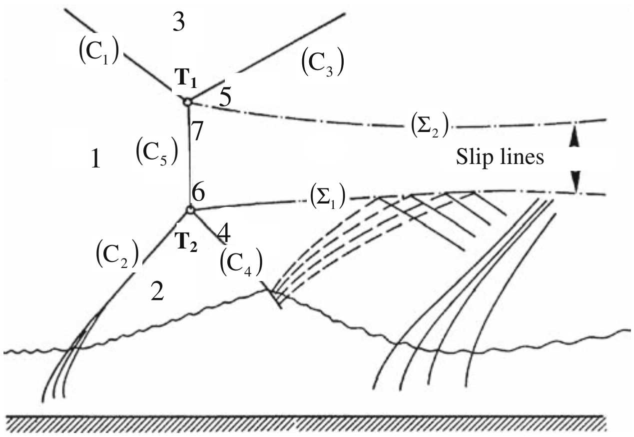


Fig. 14 Shock reflection with singular shock intersection or Mach phenomenon: Schematic view of the situation in the physical plane

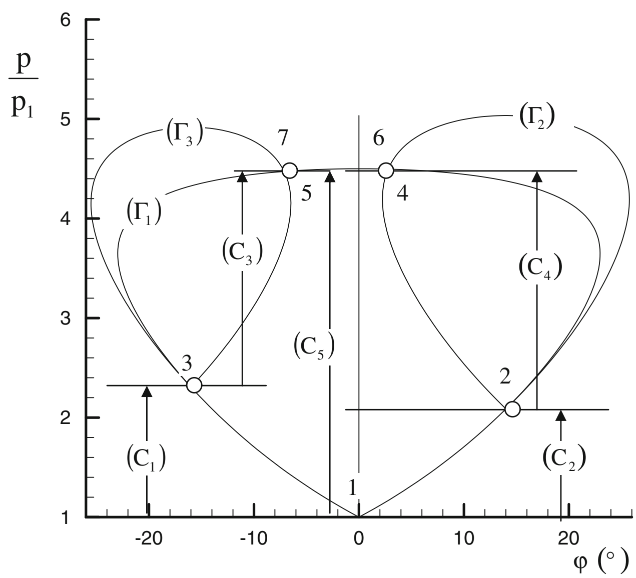


Fig. 15 Shock reflection with singular shock intersection or Mach phenomenon: Situation in the shock polar plane ($M_1 = 2, \varphi_1 = 14°$, $\varphi_2 = -16°$)

and (C_2) and a nearly normal shock, or Mach stem is formed between the two triple points $\mathbf{T}_1$ and $\mathbf{T}_2$ as shown in Fig. 14. The singular shock interaction of Fig. 15 is for an upstream Mach number of 2.5, the separation shock deflection of 14° resulting from a separation criterion [8]. The Mach reflection is obtained by increasing the incident shock deflection. The downstream states 4 and 6 located at the intersection of the polars (Γ_1) and (Γ_2) are separated in the physical plane by the slip line (Σ_1), whereas the downstream states 5 and 7 at the intersection of (Γ_1) and (Γ_3) are separated by the slip line (Σ_2). The subsonic channel downstream of the Mach stem (C_5) is accelerated under the influence of the contiguous supersonic flows, so that a sonic throat appears after which the flow is supersonic (see Fig. 14). In this case, the interaction produces a completely different outer flow structure with the formation of a complex shock pattern replacing the simple purely inviscid flow solution. Occurrence of a Mach phenomenon can be very detrimental in hypersonic air-intakes as the stagnation pressure loss behind the normal shock is much greater than behind the oblique shocks.

Similar shock patterns are encountered in over-expanded nozzle when the adaptation shock forming at the nozzle exit is strong enough to separate the nozzle boundary layer [10]. As shown in Fig. 16a which is relative to a two-dimensional nozzle, separation takes place inside the nozzle, the separation shocks forming on each wall crossing to form a Type I shock pattern. The situation in the polar plane is depicted in Fig. 16b for the following conditions: Mach number 2, shock induced deflections −10° et 10°. If separation progresses inside the nozzle, the Mach number at separation origin decreases and a situation will be reached where the intersection of the separation shocks leads to a Type II interference with formation of a Mach phenomenon (a Mach disc for an axisymmetric flow), as shown in Fig. 16c. The corresponding situation in the polar plane is shown in Fig. 16d (situation corresponding to Mach number 1.8, shock induced deflections −12° et 12°).

In some circumstances, the Type II solution of Fig. 17a is not possible, and the shock pattern sketch in Fig. 17b is observed, to which corresponds the shock polar diagram of Fig. 17c. Then, a Mach phenomenon occurs but in such a way than the flow angles after the triple points have the opposite signs as compared to the situation in Fig. 17a. In this case, the area of the subsonic channel downstream of the Mach stem (disc) increases and a situation is generally met where the subsonic flow "breaks down", a recirculation bubble forming as shown in Fig. 17b. Such a situation is sometimes called an inverse Mach reflection.

6 Shock wave unsteadiness

6.1 Introduction

The previous sections have given the elements of organization of steady interactions, showing how the shocks and most of the perfect fluid flow properties can be connected. This was essentially a steady picture. However, when the shock is strong enough to induce the separation of a turbulent boundary layer, it is known that unsteadiness appears. In general, this produces strong flow oscillations which are felt far downstream of the interaction, and can damage airframes or engines. They are generally called "unsteadiness" or "breathing", because they involve very low frequencies, typically at least at two orders of magnitude below energetic eddies of

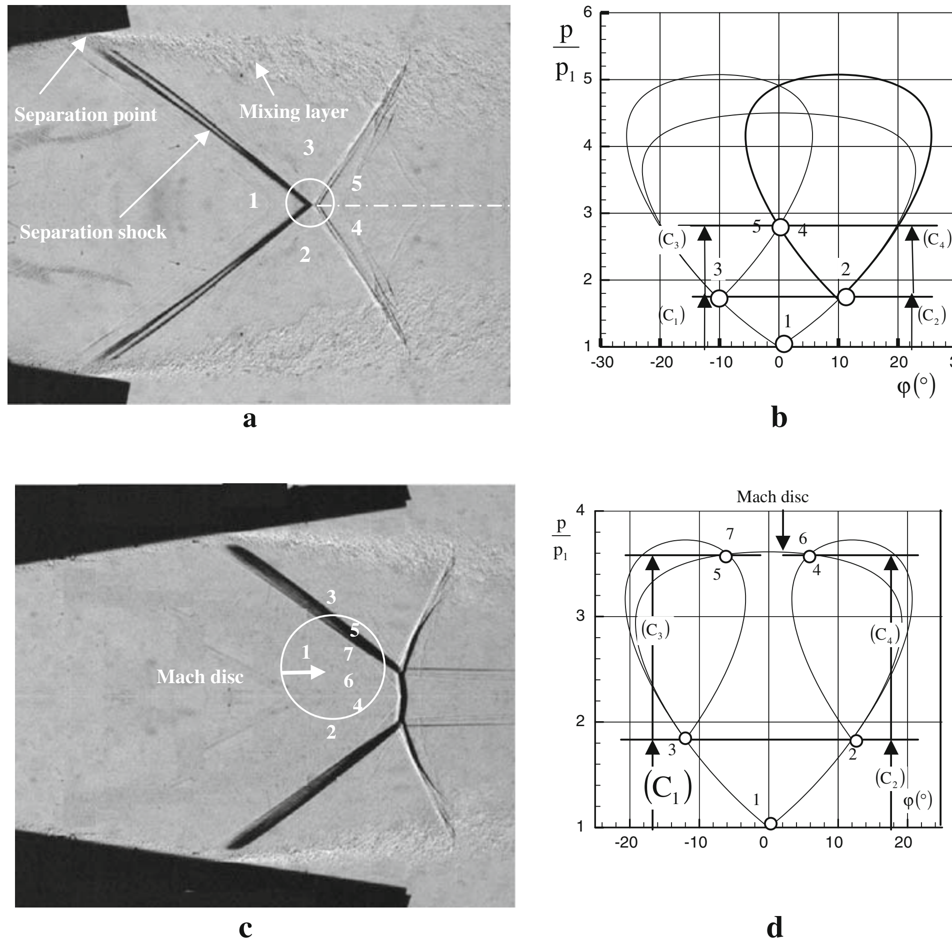


Fig. 16 Separation in an over-expanded nozzle (Onera documents). **a** separation in a nozzle with Type I interference, **b** polar plane representation for Type I interference, **c** separation in a nozzle with Type II interference, **d** polar plane representation for Type II interference

the incoming boundary layer. The origin of these oscillations raises several types of questions. What is their cause, and is there a general way to understand them? Such interactions can be produced in several ways depending on geometry, producing different sorts of geometry. An attempt of interpretation was proposed in [11,12], under the form of a diagram reproduced in Fig. 18.

The organization of the diagram needs a comment. The shock wave separates two parts of the flow: the upstream and the downstream layers. Therefore, the shock wave may be considered as an interface between upstream and downstream conditions, and its position and its motion will vary accordingly. This implies that the shock motion may be two-folded. A first class of motions can include the cases where upstream and/or downstream conditions vary, making the shock move. The other class of motions comes from the propagation of a perturbation, for example, at the foot of the shock, which propagates along the shock sheet. Having these elements in mind, the shock motion should be analysed from the point of view of the upstream and downstream conditions, and the rest of the present analysis will consist in commenting the phenomena or the flow organisation related to the different branches of the diagram.

6.2 The upper branch, local, and long distance influence

In the upper branch, we consider the evolution of the turbulent field. Turbulence is subjected to a shock wave. It is distorted and generally amplified by its passage through the shock; its anisotropy is modified in the distortion. If this passage is fast enough, the non-linear effects can be neglected, and the evolution of turbulence can be described by a (linear) theory of rapid distortion or in the linear theory of Ribner [13]. Further evolution of such distorted turbulence depends of course on non-linear effects. Further downstream, this non-equilibrium turbulence contributes to form a new boundary layer. This is represented in the diagram by the upper branch, which is relevant in all cases, separated or non-separated. This upper branch may contribute to the shock motion in two ways: first, as it represents cases where eddies fly through the shock, and therefore distort them, there is a motion due to local flow variations. Note that such a motion is taken in account in Ribner's linear theory. Second, distorted turbulence contributes to form a non-equilibrium boundary layer downstream of the interaction. If long-distance coupling can exist, as will be discussed in the next paragraph, this distorted turbulence can have an indirect influence on shock motion.

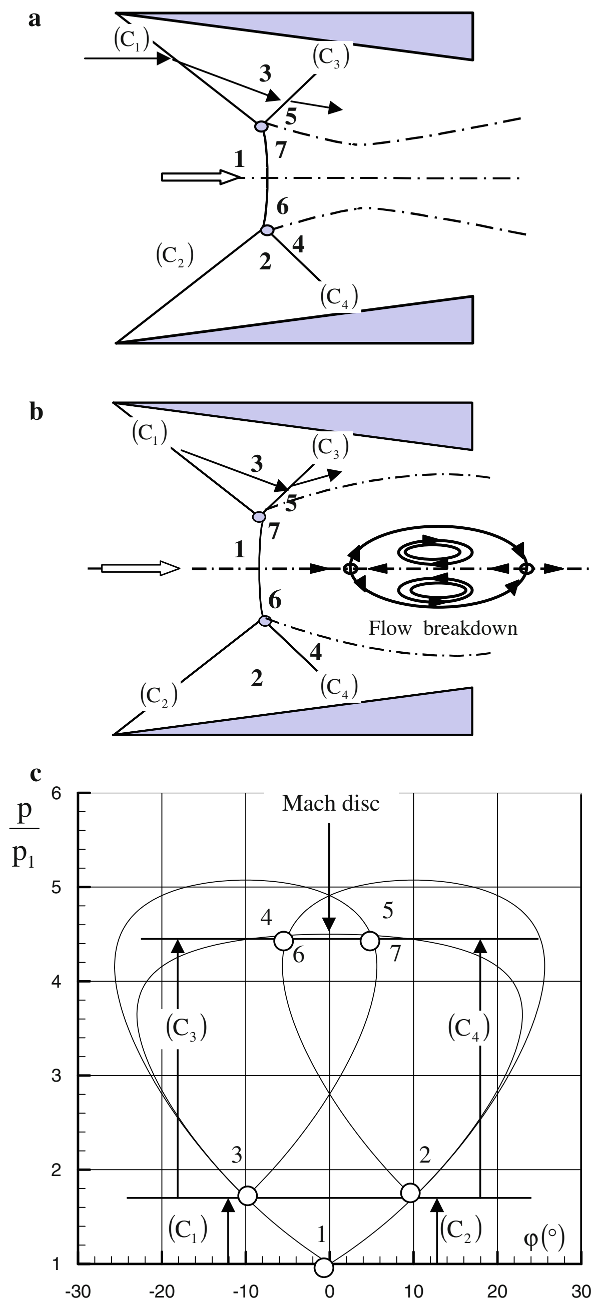


Fig. 17 Direct and inverse Mach phenomenon. **a** direct Mach phenomenon, **b** inverse Mach phenomenon, **c** inverse Mach phenomenon: Situation in the shock polar plane ($M_1 = 2, \varphi_1 = 10°, \varphi_2 = -10°$)

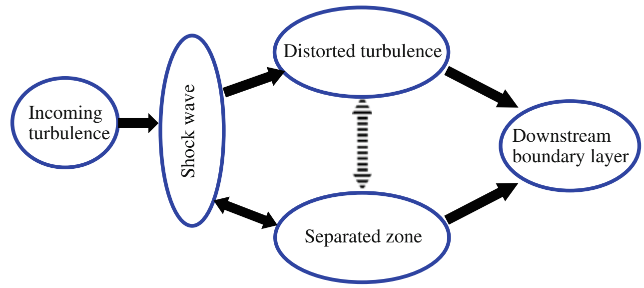


Fig. 18 A diagrammatic representation of shock boundary layer interactions

6.3 The lower branch, separated flows

Now, situations in which the shock wave is strong enough to make the boundary layer separate are considered. The separation can be incipient or well developed. We will term a separation as well developed if there is an entire zone of space experiencing reverse velocity during amounts of time long enough to produce an average separated bubble. The cases where some isolated and intermittent spots contain fluid with negative velocity and producing no average separation will be supposed to belong to incipient separation. When an average separation zone exists, it can impose its specific characteristic scales of time and space as downstream conditions for the shock wave. The two cases, incipient and well-developed produce probably different shock dynamics. Finally, the flow exiting from the separated region is shed into the downstream boundary layer and will contribute to form a new boundary layer together with the turbulence coming from upstream and distorted by the shock system.

A first remark is that the shock wave may have a particular frequency response. This depends on the shape of the shock wave and on the flow around the shock wave [14,15]. In some particular cases, as in the transonic experiments of Sajben and Kroutil [16], the shock can be frequency selective. In general, this transfer function is not known and depends on the flow, but if they are not frequency selective, the overall trend seems that the shock waves behave rather like low-pass filters, and therefore they may be expected to be more sensitive to low frequencies.

Coming back to the diagram in Fig. 18 it may be noticed that the lower branch is made of two-sided arrows. This mimics the fact that the downstream flow may control the motion of the shock, and therefore that couplings between the downstream layer and the shock wave may exist. Such a situation can happen if the flow downstream of the shock is mostly subsonic, like in plane shock interactions in channels, or in interactions on profiles in the transonic regime. This occurs, for example, when the shock motion depends on the turbulent flow far downstream of the interaction. This is probably the case of transonic buffeting for which, according to classical interpretations, there is an acoustic feedback between the flow at the trailing edge of the profile and the shock wave [17]. Note that more recent interpretations [18] suggest that this could also be linked to global instability properties. A consequence of such a far field influence is that it is possible to generate shock motion by imposing far downstream conditions. This is classically achieved in wind tunnel experiments by using a rotating cam to produce shock motion like in [19,20]. Another consequence for wind tunnel experiments

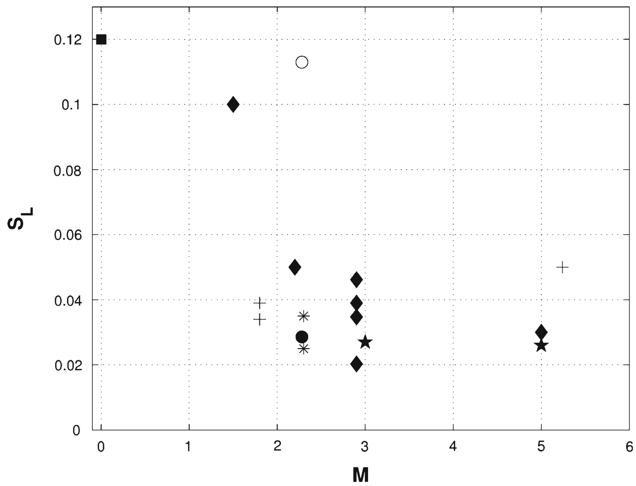


Fig. 19 Dominant frequency in supersonic shock/boundary layer interactions (from [26])

in the transonic regime is that the fluctuations in the nozzle diffuser, if strong enough, may contribute significantly to the shock motion. This represents a very particular class of flows.

If we consider now situations in which there is no feedback with the far downstream flow, different possibilities can be found. In a first one, the flow downstream of the shock wave does not impose particular conditions. Therefore, the shock motion will be specified by the incoming turbulence. This corresponds to the upper branch of the diagram in Fig. 18. This is the case of non-separated interactions, in which turbulence is just amplified through the shock, but no turbulent structures with a particular dynamics are formed just downstream of the shock wave. This is also found in shock turbulence interactions [21–23] and in compressions by turning as studied by Poggie and Smits [24]. This latter flow is made by the reattachment of a free shear layer on an inclined flat plate. The fluid can flow freely downstream, and the measurements show very clearly that the resulting mean pressure gradient and the spectra of pressure fluctuations scale with the size of the incoming turbulent structures.

The question is quite different when an obstacle or a large separation zone is present. In this case, the downstream conditions often control predominantly the shock dynamics. This was shown unambiguously by the experiments [25] on the interaction produced by a cylinder normal to a plate. They found that the dominant frequency of the shock unsteadiness varies like the inverse of the cylinder diameter. Similar results were found in interactions produced by oblique shock reflections. More precisely, a compilation of results in different sorts of interactions has been proposed in [26], and is recalled in Fig. 19.

In this figure, the dominant frequency of the shock unsteadiness has been represented. It is defined as the maximum of the premultiplied power spectrum $fE(f)$ (f is the frequency and $E(f)$ the power spectral density) of pressure fluctuations at the foot of the shock. It is normalized by the length of interaction and the external velocity downstream of the leading shock, leading to the Strouhal number:

$$S_L = \frac{fL}{U_e}$$

The collapse of the data versus Mach number for the different flows (shock reflection, compression ramps, blunt bodies, channel flows) was not excellent. The scatter was about ±20%. However, no particular trend was found versus Mach number, excepted for the compression ramp flow of Thomas et al. [27], which was found a little higher than the other data. This collapse, even if partial, is surprising, since the dominant frequency may be expected dependent on the flow geometry. This result, however, suggests that the different flow cases share some common features for the origin of the unsteadiness. The scatter was not totally satisfactory, indicating that some details were not well represented. However, it was clear that all the experiments had generally a Strouhal number around 0.03–0.04. As for well-developed interactions, L is much larger than the boundary layer thickness δ; it appears without any ambiguity that the unsteadiness is several orders of magnitude below the frequencies $\frac{U_e}{\delta}$ produced by the energetic eddies of the incoming layer, of typical size δ. Another property of the shock motion can be derived in this case. As noticed in [28] and recalled by [12], a velocity scale

$$U_s = Lf$$

can be deduced for the shock motion. Since $S_L << 1$, this implies that $U_s << U_e$. In supersonic conditions, this implies that the shock wave behaves in a quasi-static way, i.e., its intensity does not depend on its velocity, as often found in the transonic regime.

The point of the influence of the incoming turbulence may be discussed in deeper details. Coming back to the diagram in Fig. 18, it is possible for upstream turbulence to have an influence on the shock motion. Structures of very long size have been put in evidence by Adrian et al. [29], Ganapathisubramani et al. [30] in subsonic boundary layers, and by Ganapathisubramani et al. [31] in supersonic flows. If we consider the superstructures which can exist in the upstream flow, it is possible to determine a frequency scale they can generate and check if they can correspond to the observed Strouhal numbers. Such structures are supposed to be 30δ long, with a speed of convection of $0.75U_e$ The resulting frequency is therefore $f = \frac{0.75U_e}{30\delta}$ and the corresponding Strouhal number is

$$S_L = 0.025\frac{L}{\delta}$$

It turns out that such structures can generate fluctuations with the right Strouhal number for interactions such that $\frac{L}{\delta} \approx$

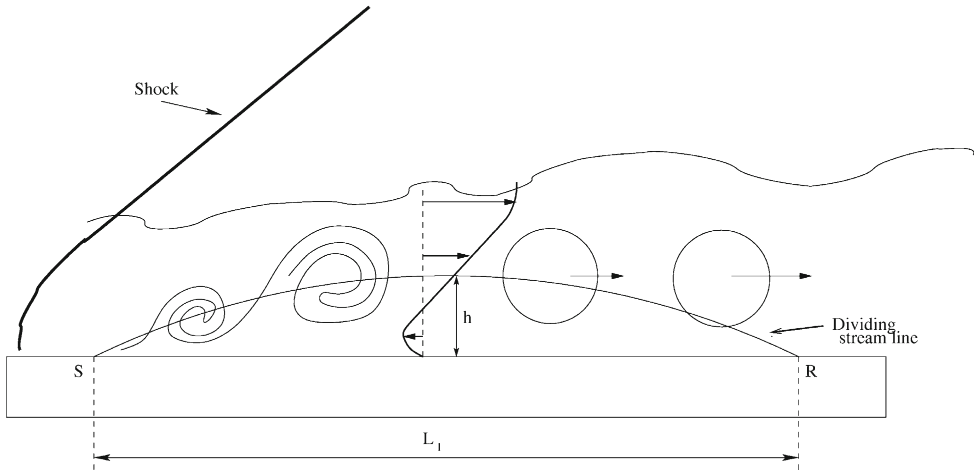


Fig. 20 Sketch of the separated zone in impinging oblique shock interactions

1. Interactions like the oblique shock reflection studied in [12] have ratios $\frac{L}{\delta}$ about 5 or 7 for Strouhal numbers about 0.03–0.04. Therefore, such low frequencies cannot explain the unsteadiness observed in the shock reflection, and do not provide a general answer to the question of the origin of the shock wave unsteadiness in separated flows.

Piponniau et al. [32] have proposed an analysis for the origin of this unsteadiness. Their conditional measurements of the size of the separated bubble showed that this zone is strongly intermittent, with a few events during which backwards flow of strong intensity engulfs into the separation pocket. These events are connected with shock motions of large amplitude. This has led to the assessment that the large shock pulsations are closely related to the flapping of the mixing layer formed at the edge of the separated bubble as shown in Fig. 20. Therefore, they proposed an explanation based on considerations on air entrainment by this mixing layer.

Their objective was to find the parametric dependence of the shock motion frequency rather than a complete theoretical description. They considered that air entrainment drained air from the separated zone. They evaluated the amount of mass contained in the bubble and the rate of mass entrained. The ratio of these two quantities provides a time scale which represents the time necessary to drain a significant amount of mass from the separated zone. The inverse of this time gives a frequency scale. They assumed that the dependence of the spreading rate of the mixing layer on density and velocity ratios and on convective Mach number is the same as in canonical mixing layers. The analysis provides a Strouhal number of the form:

$$S_L = \Phi(M_c) g(r, s) \frac{L}{h}$$

in which $\Phi(M_c)$ in normalized spreading rate of the mixing layer, $g(r, s)$ is a weak function of the velocity and density ratios r and s; h is the height of the separated bubble. Essentially, this correlation suggests that for the same aspect ratio $\frac{L}{h}$, the Strouhal number varies with the convective Mach number like the spreading rate of the mixing layer. This is supported by the existing data. It should be remarked that in most of the interactions under investigation, the convective Mach number of the large structures in the mixing layer is close to 1. This corresponds to values of $\Phi(M_c)$ in the range 0.2–0.3 and suggests that the aspect ratio of the considered separations is about 5 or 6. This is consistent with what is known from the details of the geometry of these interactions. Therefore, it seems that this simple model provides a more general representation of the unsteadiness. Of course, this scheme is limited to two-dimensional situations in which a reattachment point exists, as in Restricted Shock Separation found in nozzle flows; for other types of interactions, the reader may be referred to [33]. The simple model proposed here gives, however, indications on the leading elements for analysing other situations, and on the way to control them.

7 Concluding remarks

The structure of shock wave/boundary layer interactions is predominantly a consequence of the response of the boundary layer to the sudden local compression imparted by the shock and it reacts as a non-uniform flow in which viscous and inertial terms combine in an intricate manner. The most significant result of this is the spreading of the pressure discontinuity caused by the shock so that its influence is felt well upstream of where this would have been located in an inviscid fluid model. When the shock is strong enough to separate the boundary layer, the interaction has dramatic consequences for the development of the boundary layer and for the contiguous inviscid flow field. Complex shock patterns are then formed which involve shock/shock interferences whose nature depends on the Mach number and on the way the primary shock is produced (whether by shock reflection, ramp or normal shock). In these circumstances, the most salient feature of shock-induced separation is the pattern of shocks produced even though this is a secondary phenomenon associated with the process. The boundary layer behaves more or less as it would for any other ordinary separation and essentially the same as in subsonic flows. It obeys the specific laws mainly dictated by the intensity of the overall pressure rise imparted by the shock, regardless of the way in which this is generated. A very striking feature of these interactions is the overwhelming repercussion that the shock has on the contiguous inviscid supersonic stream, which can be spectacular for internal flows. Although their basic flow topology is the same, laminar and turbulent interactions have distinctly different properties that stem from the far greater resistance of a turbulent boundary layer to flow retardation and hence separation.

One of the most detrimental consequences of SWBLI is the occurrence of flow unsteadiness. This can be of high intensity when the shock is strong enough to induce separation. Such unsteadiness can occur at high frequencies when associated with turbulent fluctuations and to a lesser extent with separated bubble instabilities. In other cases, unsteadiness occurs at very low frequency when the fluctuating motions involve the whole aerodynamic field. This corresponds to large length scales, as, for example, in transonic buffeting or in air-intake buzz. Such large-scale unsteadiness seems to be a special feature of transonic interactions where the downstream subsonic flow allows a forward transmission of perturbations that excite the shock wave. In fully supersonic interactions, the higher Mach number of the outer flow field tends to isolate the interaction domain from downstream perturbations; the perturbations however remain at frequencies much lower than the energetic eddies of the incoming boundary layer. This can be related to the differences of mass entrainment in the mixing layer of the separated bubble. A consequence is that the frequency range involved by the fluctuations of the separated bubble and by the shock wave decreases with Mach number like the spreading rate of the supersonic mixing layer; this implies a reduction of frequency with respect to the subsonic situation. These pictures are consistent with a two-dimensional situation, and the influence of three-dimensional geometries and/or of mass bleeding, together with the free separation in which separation is not followed by reattachment remain issues to lead to the determination of unsteadiness in the general case.

In spite of a large numbers of studies over more than half a century, the complex phenomena resulting from the interaction of a shock and a boundary layer remain a major concern for high speed aerodynamicists. Important and challenging aspects of the phenomena have not been considered in this brief review article such as the 3D character of SWBLIs. In reality, most of the configurations are three-dimensional leading to complex flow topology and shock structure. Furthermore, even in nearly 2D situations (channel flow or nozzle flow for example), the flow inevitably adopts a 3D organisation, whose influence on the overall flow is ill known. This aspect, which has been frequently ignored, has received only recently some attention. Also, as already said, the unsteadiness in SWBLIs are considered since a relatively short time and it can be anticipated that this aspect (also true for any separated flow) is of crucial importance in domains such as aeroacoustics, aeroelasticity, combustion and others.

Thus, there remains a large field of basic research on SWBLI to establish a still more realistic and precise physical description of the flow structure. On the other hand, reliable modelling of shock induced separation, including the prediction of unsteadiness, is still largely an open question…but this is another story.

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