1 Introduction

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Shock wave–boundary-layer interactions (SBLIs) occur when a shock wave and a boundary layer converge and, since both can be found in almost every supersonic flow, these interactions are commonplace. The most obvious way for them to arise is for an externally generated shock wave to impinge onto a surface on which there is a boundary layer. However, these interactions also can be produced if the slope of the body surface changes in such a way as to produce a sharp compression of the flow near the surface – as occurs, for example, at the beginning of a ramp or a flare, or in front of an isolated object attached to a surface such as a vertical fin. If the flow is supersonic, a compression of this sort usually produces a shock wave that has its origin within the boundary layer. This has the same affect on the viscous flow as an impinging wave coming from an external source. In the transonic regime, shock waves are formed at the downstream edge of an embedded supersonic region; where these shocks come close to the surface, an SBLI is produced.

In any SBLI, the shock imposes an intense adverse pressure gradient on the boundary layer, which causes it to thicken and possibly also to separate. In either case, this increases the viscous dissipation within the flow. Frequently, SBLIs are also the cause of flow unsteadiness. Thus, the consequences of their occurrence almost invariably are detrimental in some respect. On transonic wings, they increase the drag and they have the potential to cause flow unsteadiness and buffet. They increase blade losses in gas-turbine engines, and complicated boundary-layer control systems must be installed in supersonic intakes to minimize the losses that they cause either directly by reducing the intake efficiency or indirectly because of the disruption they cause to the flow entering the compressor. These systems add weight to an aircraft and absorb energy. In hypersonic flight, SBLIs can be disastrous because at high Mach numbers, they have the potential to cause intense localized heating that can be severe enough to destroy a vehicle. In the design of scramjet engines, the SBLIs that occur in the intake and in the internal flows pose such critical issues that they significantly can limit the range over which vehicles using this form of propulsion can be deployed successfully. This list of examples is by no means exhaustive.

Our aim in writing this book is to establish a general understanding of the aerodynamic processes that occur in and around SBLIs, concentrating as much as possible on the physics of these flows. We seek to explain which factors determine their structure under a variety of circumstances and also show how they impact on other parts of their flowfield, influencing parameters such as the drag, the surface-flux distributions, and the overall body flow. Our intention is to develop an understanding of which circumstances lead to their formation, how to estimate their effect, and how to manage them if they do occur. We demonstrate how the present state of our understanding has resulted through contributions from experiments, computational fluid dynamics (CFD), and analytical methods. Because of their significance for many practical applications, SBLIs are the focus of numerous studies spanning several decades. Hence, there is a considerable body of literature on the subject. We do not attempt to review all of it in this book but we aim to distill from it the information necessary to fulfill our aims.

1.1 Structure of the Book

The first chapter of the book explains the fundamental aerodynamic concepts relevant to all SBLIs. Subsequent chapters examine in more detail the interactions in specific Mach-number regimes, beginning with transonic flows, followed by supersonic flows, and finally hypersonic and rarefied flows. Throughout the chapter, examples are cited that demonstrate how the nature of the interaction varies with these changes. Because of the wide range of knowledge and disciplines involved, we do not attempt to do this entirely alone; we have enlisted several prominent internationally recognized experts in the field who very generously contributed to the preparation of the book. They were asked specifically to give their perspective on critical experimental, computational, and analytical issues associated with SBLIs in their particular area. In all, six chapters are contributions from other authors and we gratefully acknowledge their assistance. Although we edited the material provided, we do not attempt to unify the writing style but instead seek to retain the flavor of individual contributions as much as possible.

Chapter 2 explains the fundamental aerodynamic concepts relevant to SBLIs throughout the Mach-number range. This chapter was written by Professor Jean Délery, the former Head of Aerodynamics at ONERA in France. Although it is not our intention to produce a conventional textbook, this chapter comes close in that it provides a wide-ranging overview of the background aerodynamics relevant to SBLIs. In writing this chapter, Professor Délery emphasized the explanation of the underlying physics of the flows, and his contribution is an invaluable platform for subsequent chapters.

Chapter 3 addresses *transonic SBLIs*. This topic is of particular relevance to the super-critical wings that are used widely on many current aircraft and to gas-turbine-blade design. We wrote this chapter in conjunction with Professor Délery and it includes new results on SBLIs in this range. Again, the emphasis is on establishing an understanding of the physical processes taking place within these interactions because this is considered a necessary prerequisite for devising effective control strategies to minimize detrimental effects.

Chapters 4 and 5 are devoted to *supersonic interactions and their numerical modeling*. Chapter 4 concentrates on two-dimensional interactions and is followed by a discussion of three-dimensional SBLIs in Chapter 5. Both chapters were written by Professor Doyle D. Knight from Rutgers University in New Jersey, USA, and Professor Alexander A. Zheltovodov from the Khristianovich Institute of Theoretical and Applied Mechanics in Novosibirsk, Russia. They chose to describe in detail a number of fundamental flowfields to explain how the SBLI structure changes across the parameter range. More complex flowfields can be understood as combinations of one or more of these fundamental elements. In particular, three-dimensional interactions are explained from a basis of comparison with equivalent two-dimensional flow cases described in Chapter 4. The capabilities of CFD to predict these complex supersonic flows also are assessed in this part of the book.

The next three chapters comprise a section devoted to *hypersonic SBLIs*. Predicting when and how they develop in this speed range is especially important because of the impact on vehicle design. Chapter 6 is written by Dr. Michael Holden from CUBRC in Buffalo, New York, USA. For several decades, he has been acknowledged as the leading experimentalist in this area. He presents a wide range of results from which he develops a detailed insight into the impact that SBLIs make on vehicle aerodynamics at high Mach numbers and to what extent the outcome can be predicted.

Chapters 7 and 8 focus on numerical simulation, including the influence of real-gas effects, rarefaction, and chemical reactions on the interactions. Professor Graham V. Candler of the University of Minnesota, Minneapolis, MN, USA is the author of Chapter 7. He shows that despite the success of current advanced computational methods in predicting hypersonic flows, accurate simulation of SBLIs remains a major challenge. He discusses the physics of hypersonic SBLI flows and emphasizes how understanding this bears on the effective numerical simulation of them. Chapter 8 addresses the way in which the very low ambient density that occurs in the upper atmosphere impacts vehicle flows involving SBLIs. Under these circumstances, the conventional Navier-Stokes methods fail and particle-simulation methods, specifically Direct Simulation Monte Carlo (DSMC), must be used as an effective alternative predictive tool. Chapter 8 cites results for hypersonic flows from which the influence that rarefaction and chemical reactions have on SBLIs can be assessed.

The book concludes with two chapters that address the specialised topics of *flow unsteadiness* associated with SBLIs and the use of *analytical treatments*. Chapter 9 was written by Dr. Jean-Paul Dussauge in collaboration with Drs. P. Dupont and J. F. Debiève from the Institut Universitaire des Systèmes Thermiques Industriels, Université d'Aix-Marseille in France. In their chapter, they consider turbulent interactions in the transonic and lower supersonic range and explore how flow structures in the SBLI and external stimuli (e.g., upstream turbulence and acoustic disturbances) lead to flow unsteadiness and downstream disturbances.

1.1.1 George Inger

Very sadly, Professor George Inger, who produced the final chapter, died on November 6, 2010 before this book was published.[1] His contribution is written from

[1] When George Inger died he was serving as a Visiting Professor at Virginia Polytechnic Institute and State University in Blacksburg, Virginia, where he had previously taught in the 1970s. Before that he occupied the Glenn Murphy Chair of Engineering at Iowa State University where he had been a researcher, teacher, and consultant in the field of aero-thermodynamics for more than 30 years.

a very personal perspective and describes those areas where he considered analytical methods could be applied effectively to the SBLI problem. We consider it a fitting tribute to him that we are able to include his material in its entirety as, in some measure, it summarizes a significant portion of his life's work, that of applying asymptotic expansion methods to fluid mechanics. Building on the legacy of Lighthill, Stewartson, and Neiland who developed the so-called triple deck method in the late 1960s, George extended their concepts well beyond their early achievements to, for example, SBLIs including turbulent interactions; a section devoted to this is included in his chapter as hitherto unpublished work. He was a great enthusiast for analytical methods and he firmly believed that they complemented and were an essential adjunct to experiments and numerical methods, being an effective means of enhancing our insight into the physics of a flow. While acknowledging that for complex flows such as SBLIs these methods have proved to have had significant limitations, they nevertheless continue to provide us with valuable interpretations of observed phenomena and predict flow behavior over a wide range of conditions. For this reason we were delighted to be able to have his contribution to this book.

1.2 Intended Audience

This book is targeted to technologists, research workers, and advanced-level students working in industry, research establishments, and universities. It is our intention to provide a single source that presents an informed overview of all aspects of SBLIs. In preparing the book, we endeavored to explain clearly the relevant fluid mechanics and ensure that the material is accessible to as wide an audience as possible. However, we assumed that readers have a good working knowledge of basic fluid mechanics and compressible flow. This is the first book solely devoted to the subject and it incorporates the latest developments, including material not previously publicized.

Before entering academia, he had worked in the aerospace industry with McDonnell-Douglas, Bell Aircraft, and the GM Research Laboratories. Over his career, he published extensively and became a pioneer in the basic theory of high temperature chemically reacting gas flows and propulsion in space.

2 Physical Introduction

Jean Délery

2.1 Shock Wave–Boundary-Layer Interactions: Why They Are Important

The repercussions of a shock wave–boundary layer interaction (SBLI) occurring within a flow are numerous and frequently can be a critical factor in determining the performance of a vehicle or a propulsion system. SBLIs occur on external or internal surfaces, and their structure is inevitably complex. On the one hand, the boundary layer is subjected to an intense adverse pressure gradient that is imposed by the shock. On the other hand, the shock must propagate through a multilayered viscous and inviscid flow structure. If the flow is not laminar, the production of turbulence is enhanced, which amplifies the viscous dissipation and leads to a substantial rise in the drag of wings or – if it occurs in an engine – a drop in efficiency due to degrading the performance of the blades and increasing the internal flow losses. The adverse pressure gradient distorts the boundary-layer velocity profile, causing it to become less full (i.e., the shape parameter increases). This produces an increase in the displacement effect that influences the neighbouring inviscid flow. The interaction, experienced through a viscous-inviscid coupling, can greatly affect the flow past a transonic airfoil or inside an air-intake. These consequences are exacerbated when the shock is strong enough to separate the boundary layer, which can lead to dramatic changes in the entire flowfield structure with the formation of intense vortices or complex shock patterns that replace a relatively simple, predominantly inviscid, unseparated flow structure. In addition, shock-induced separation may trigger large-scale unsteadiness, leading to buffeting on wings, buzz for air-intakes, or unsteady side loads in nozzles. All of these conditions are likely to limit a vehicle's performance and, if they are strong enough, can cause structural damage.

In one respect, shock-induced separation can be viewed as a compressible manifestation of the ubiquitous flow-separation phenomenon: The shock is simply an associated secondary artefact. From the perspective of viscous-flow, the behaviour of the separating boundary layer is basically the same as in incompressible flow, and the overall topology is identical. Nevertheless, the most distinctive and salient features of shock-separated flows are linked to the accompanying shock patterns formed in the contiguous inviscid outer flow. The existence of these shocks may have major consequences for the entire flowfield; in practice, it is difficult to completely separate SBLIs from the phenomena that arise due to the intersection of shock waves – usually referred to by the generic term *shock-shock interference*. SBLIs can occur at any Mach number ranging from transonic to hypersonic, but it is in the latter category that the shocks have particularly dramatic consequences due to their greater intensity.

It is not inevitable that SBLIs or, more generally, shock wave/shear layer interactions have entirely negative consequences. The increase in the fluctuation level they cause can be used to enhance fuel-air mixing in scramjet combustion chambers or to accelerate the disorganisation of hazardous flows, such as wing-trailing vortices. Also, because interactions in which separation occurs can lead to smearing or splitting of the shock system, the phenomenon can be used to decrease the wave drag associated with the shock. This last point illustrates a subtle physical aspect of the behaviour of SBLIs. Shock waves also form in unsteady compressible flows by focusing compression waves, as seen in the nonlinear acoustic effects in rocket combustion chambers or the compression caused by a high-speed train entering a tunnel. Extreme cases are associated with explosions or detonations in which interactions occur in the boundary layer that develops on the ground or the surface behind the propagating blast wave.

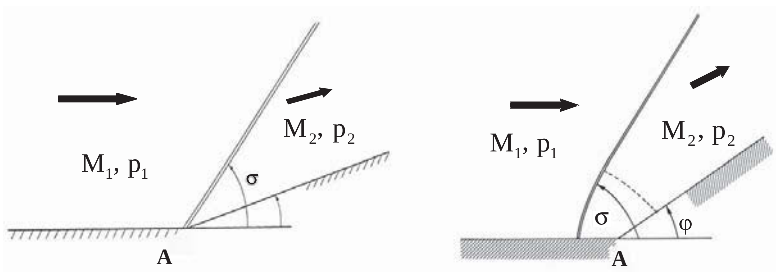
SBLIs are a consequence of the close coupling between the boundary layer – which is subjected to a sudden retardation at the shock-impact point – and the outer, mostly inviscid, supersonic flow. The flow can be influenced strongly by the thickening of the boundary layer due to this retardation. Although in many instances these flows can be computed effectively with modern computational fluid dynamics (CFD), the methods are certainly not infallible, especially if the flow is separated. For this reason, it is necessary to clearly understand the physical processes that control these phenomena. With this understanding, good designs for aerodynamic devices can be produced while avoiding the unwanted consequences of these interactions or, more challengingly, exploiting the possible benefits. An effective analysis of both the inviscid flow and boundary layer must be obtained to achieve this understanding. Therefore, the next section summarizes the basic results from shock-wave theory and describes the relevant properties of boundary-layer flows and shock-shock interference phenomena.

2.2 Discontinuities in Supersonic Flows

2.2.1 Shock Waves

The discontinuities that can occur within supersonic flow take several forms, including shear layers and slip lines as well as shock waves. The governing equations are presented in Appendix A of this chapter and include the Rankine-Hugoniot equations that govern shock waves and other discontinuities. From these equations, we can establish the following results, which have direct relevance when considering aerodynamic applications involving SBLIs. When the flow crosses a shock wave, it entails the following:

1. A discontinuity in flow velocity, which suddenly decreases.
2. An abrupt increase in pressure, which has several major practical consequences including:



a- attached shock wave (φ < limit deflection) b- detached shock wave (φ > limit deflection)

Figure 2.1. Attached and detached planar shock wave.

   i. A boundary layer at a surface that is hit by a shock suffers a strong adverse pressure gradient and therefore will thicken and may separate.
   ii. The structure of the vehicle is submitted to high local loads, which can fluctuate if the shock oscillates.
3. A rise in flow temperature, which is considerable at high Mach numbers, so that:
   i. The vehicle surface is exposed to localised high-heat transfer.
   ii. At hypersonic speeds, this heating is so intense that the fluid can dissociate, become chemically reactive, and possibly ionise downstream of the shock.
4. A rise in entropy or, equivalently, a decrease in the stagnation pressure. This is a significant source of drag and causes a drop in efficiency (i.e., the maximum recovery pressure diminishes).

The Rankine-Hugoniot conservation equations provide an inviscid description of a discontinuity, whether a shock wave or a slip line. This analysis conceals the fact that such phenomena, in reality, are dominated by viscosity at work either along the slip line or inside the shock wave. This is a region of rapid variation of flow properties but of finite thickness, roughly 10 to 20 times the incident flow molecular mean free path. This fact explains why there is an entropy rise through a shock wave: In an adiabatic and nonreacting flow, the only source of entropy is viscosity.

2.2.2 The Shock-Polar Representation

Valuable physical insight about how the shock patterns associated with SBLIs develop can be gained by considering the so-called shock polar [1], which provides a graphical representation of the solution to the Rankine-Hugoniot equations for oblique shocks. Consider a uniform supersonic flow with Mach number M_1 and pressure p_1 flowing along a rectilinear wall with direction φ_1 (by convention, we assume $\varphi_1 = 0$), as shown in Fig. 2.1a. At **A**, the wall exhibits a change of direction, $\Delta\varphi = \varphi - \varphi_1 = \varphi$. As long as this deflection is not too large, **A** will be the origin of a plane-oblique shock wave that separates upstream flow (1) from downstream state (2), with states (2) and (1) connected by the Rankine-Hugoniot equations. Shock polar (Γ) is the locus of the states connected to upstream state (1) by a shock wave; the shape of (Γ) depends on the upstream Mach number M_1 and the value of the specific heat ratio γ. There are several such representations; the most convenient form is a plot of the shock-pressure rise (or the pressure ratio p_2/p_1) versus the velocity deflection φ through the shock. Shock polars defined this way are

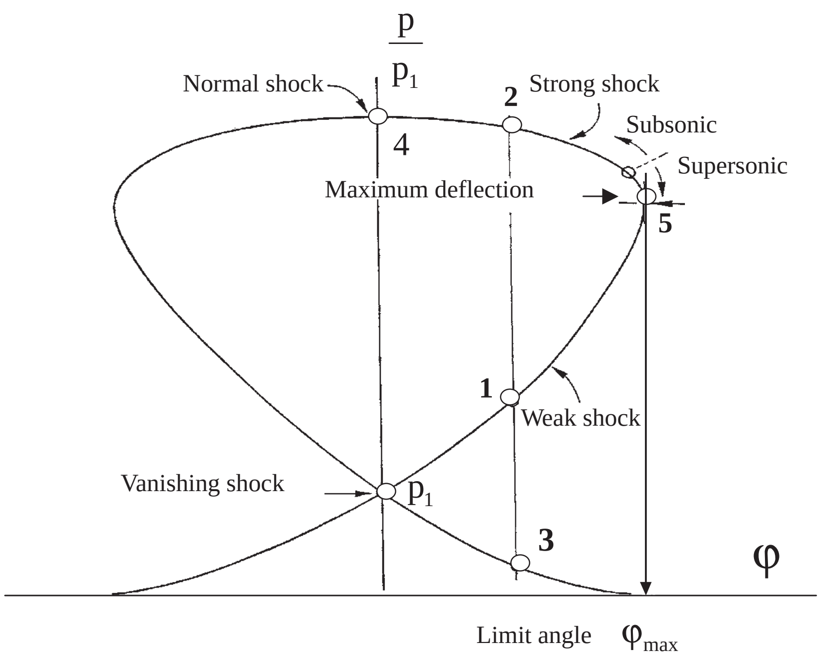


Figure 2.2. The shock-polar representation in term of flow deflection and pressure jump.

closed curves that are symmetrical with respect to the axis $\varphi = 0$ (if φ_1 is assumed to be equal to zero), as shown in the example plotted in Fig. 2.2. At the origin, the polar has a double point corresponding to a vanishing shock (i.e., Mach wave). For a given value of the deflection angle φ, there are two admissible solutions, (1) and (2). (A third solution for which the pressure through the shock decreases is rejected because it fails to satisfy the Second Law of Thermodynamics.) Solution (1), which leads to the smaller pressure jump, is called the *weak solution*; the second is the *strong solution*. For $\varphi = 0$, the strong solution is the normal shock – that is, point (4) on the shock polar.

There is a maximum deflection φ_{max} beyond which an attached shock at **A** is no longer possible. If the deflection φ imparted by the ramp is greater than φ_{max}, then a detached shock is formed starting from the wall upstream of **A** (see Fig. 2.1b). In this case, the flow downstream of the shock does not have a unique image point on the polar but instead follows an arc extending from the normal shock image (i.e., for the shock foot at the wall) to the image corresponding to the shock away from the wall. Another particular point about the polar is the image of the shock for which the downstream flow is sonic. This point is slightly below the maximum deflection location and it separates shocks with supersonic downstream conditions from those with subsonic downstream conditions. A shock polar exists for every upstream Mach number; the shape of the curves for several examples is illustrated in Fig. 2.3. The slope of shock polar $(dp/d\varphi)_0$ at the origin passes through a minimum $(dp/d\varphi)_{min}$ for an upstream Mach number $M_1 = \sqrt{2} = 1.414$, with $(dp/d\varphi)_0 > (dp/d\varphi)_{min}$ for $M_1 > \sqrt{2}$ and $(dp/d\varphi)_0 > (dp/d\varphi)_{min}$ for $M_1 < \sqrt{2}$. Thus, when $M_1 > \sqrt{2}$, each successive polar is above the previous one as the Mach number increases. The order

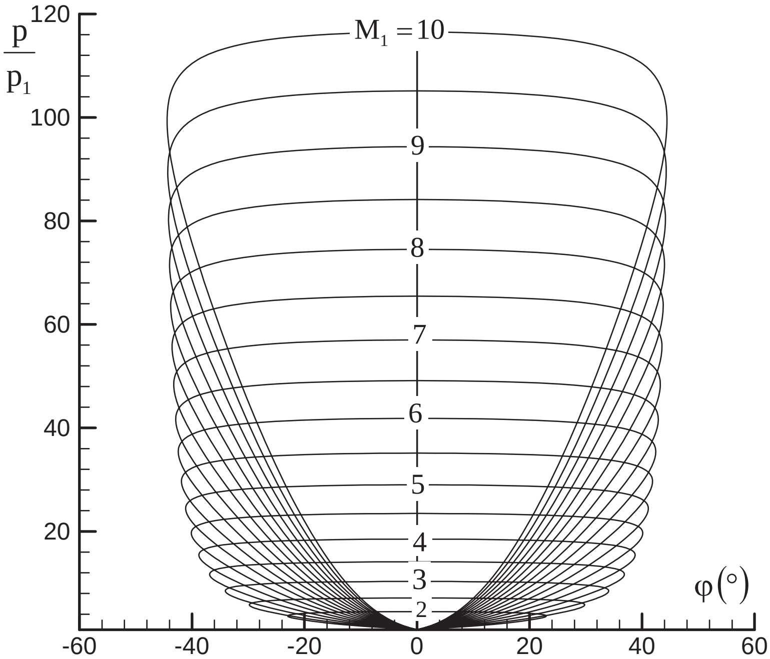


Figure 2.3. Shock polars for varying upstream flow Mach numbers ($\gamma = 1.4$).

reverses for $M_1 < \sqrt{2}$, with the polar for a lower Mach number now above the previous polar (Fig. 2.4). This fact is significant when considering the shock penetration of a boundary layer (see Section 2.5).

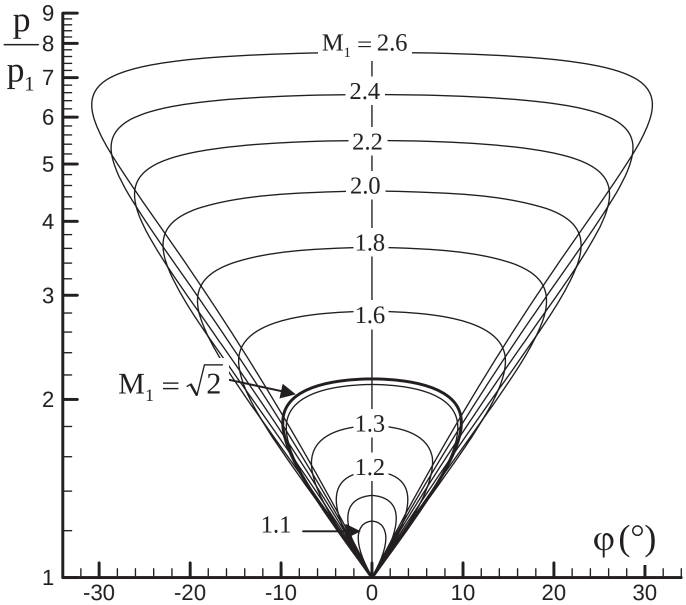


Figure 2.4. Relative positions of the shock polars ($\gamma = 1.4$).

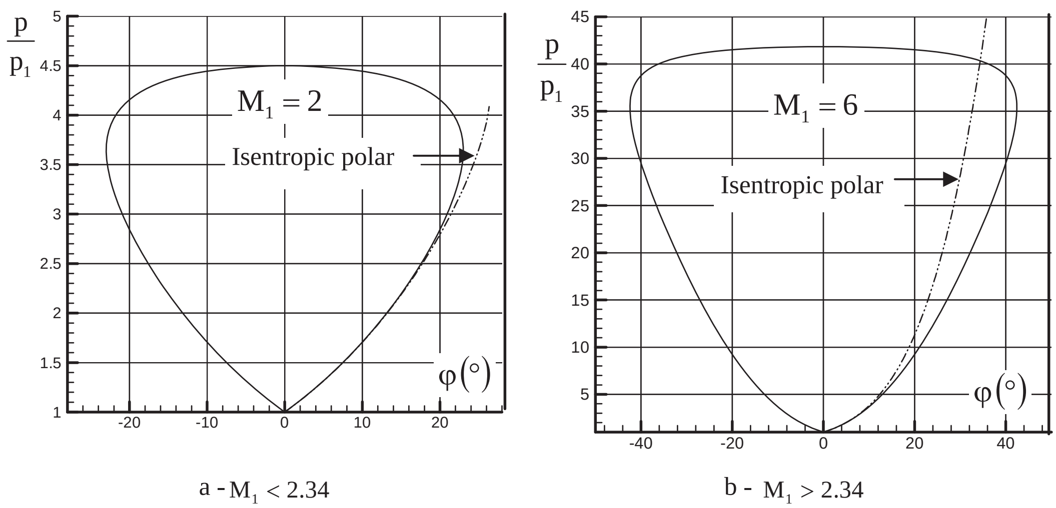


Figure 2.5. Relative location of the shock polar and the isentropic polar ($\gamma = 1.4$).

Any flow of gas in equilibrium undergoing isentropic changes from known stagnation conditions is completely defined by two independent variables: pressure p and direction φ. This flow has a unique image point in the plane $[\varphi, p]$, which is considered a *hodographic plane*. Passing through a shock wave entails a change in entropy of the fluid, resulting in a jump of its image in the $[\varphi, p]$ plane. The new point must lie on the shock polar attached to the upstream state. An interesting property of the hodographic representation $[\varphi, p]$ is that two contiguous flows separated by a slip line have coincident images because – according to the Rankine-Hugoniot equations – the condition for the flows to be compatible is that they have the same pressure and direction. Similarly, a simple isentropic expansion or compression can be represented by an isentropic polar in the $[\varphi, p]$ plane. For a planar two-dimensional flow of a calorically perfect gas, such a curve is defined by the following characteristic equation:

$$\omega(M, \gamma) \pm \varphi = \text{constant},$$

where $\omega(M, \gamma)$ is the Prandtl-Mayer function:

$$\omega(M, \gamma) = \sqrt{\frac{\gamma+1}{\gamma-1}} \tan^{-1} \sqrt{\frac{\gamma-1}{\gamma+1}(M^2-1)} - \tan^{-1}\sqrt{M^2-1}$$

The polar representing an isentropic compression from the same Mach number is plotted with the shock polar in Fig. 2.5. It can be demonstrated that at the origin, the two curves have a third-order contact, so they remain very close until relatively high deflection angles. At moderate Mach numbers, weak-type shock solutions can be considered as almost isentropic. The isentropic polar is below the shock polar for upstream Mach numbers that are less than about 2.34 and it passes above for higher Mach numbers.