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# Shock tunnel experiments on control of shock induced large separation bubble using boundary layer bleed

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ABSTRACT

Shock–Boundary Layer Interaction (SBLI) often occurs in supersonic/hypersonic flow fields. Especially when accompanied by separation (termed strong interaction), the SBLI phenomena largely affect the performance of the systems where they occur, such as scramjet intakes, thus often demanding the control of the interaction. Experiments on the strong interaction between impinging shock wave and boundary layer on a flat plate at Mach 5.96 are carried out in IISc hypersonic shock tunnel HST-2. The experiments are performed at moderate flow total enthalpy of 1.3 MJ/kg and freestream Reynolds number of 4 million/m. The strong shock generated by a wedge (or shock generator) of large angle 30.96° to the freestream is made to impinge on the flat plate at 95 mm (inviscid estimate) from the leading edge, due to which a *large* separation bubble of length (75 mm) comparable to the distance of shock impingement from the leading edge is generated. The experimental simulation of such large separation bubble with separation occurring close to the leading edge, and its control using boundary layer bleed (suction and tangential blowing) at the location of separation, are demonstrated within the short test time of the shock tunnel (∼600 μs) from time resolved schlieren flow visualizations and surface pressure measurements. By means of suction – with mass flow rate one order less than the mass flow defect in boundary layer – a reduction in separation length by 13.33% was observed. By the injection of an array of (nearly) tangential jets in the direction of mainstream (from the bottom of the plate) at the location of separation – with momentum flow rate one order less than the boundary layer momentum flow defect – 20% reduction in separation length was observed, although the flow field was apparently unsteady.


## 1. Introduction

Interaction of a shock of sufficient strength with a boundary layer effects in flow separation [5]. A shock impinging on a flat plate boundary layer is a fundamental case of the interaction, which is especially observed in high speed intakes at off-design condition (at higher operating Mach numbers than the design conditions). Particularly, at hypersonic speeds, the occurrence of such interactions in scramjet intakes, when accompanied by flow separation, affects the performance of the entire system drastically, thus calling for the control of the interaction at off-design. In such a case the separation occurs close to (or at) the leading edge (where the boundary layers are insignificantly thin); the separation bubble is of length, comparable to the distance of shock impingement from the leading edge – termed *large* separation bubble in the present study.

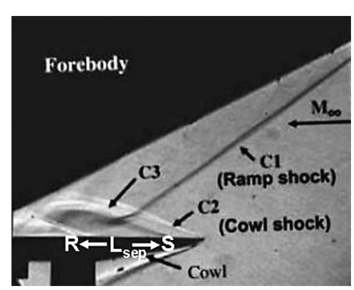


**Fig. 1.** Schlieren image of intake flow field at off-design Mach number of 8 (Mahapatra and Jagadeesh [10]).

A schlieren image of such separation occurring in hypersonic intake (cowl plate) at off-design Mach number of 8 was presented by Mahapatra and Jagadeesh [10]; the schlieren image is shown in Fig. 1, with the indication of location of separation S, reattachment R and the distance between them $L_{sep}$ called separation length.

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There are very few (and only recent) investigations on the separation at leading edge termed 'separation with zero initial boundary layer thickness' at hypersonic (high enthalpy) flow conditions [14] and to the best of our knowledge there are no reported works on the control of such flow fields at hypersonic speeds in the literature.

The interactions and their control (of especially separation, using various techniques) at supersonic speeds are well understood [4,15,22]. The understanding of complex 3-dimensional and unsteady aspects of the interaction at supersonic speeds has grown substantially with the advent of advanced flow diagnostics (like PIV and supersonic anemometry) and computations (especially LES and DNS). However, experimentation on the interaction at hypersonic flow condition requires the simulation of associated moderate to high flow total enthalpies. While hypersonic wind tunnels do simulate the high Mach numbers, impulse facilities like shock tunnels are required to simulate the required total enthalpy, but only for short test times (∼1 ms). There are however, only few shock tunnel studies on shock wave boundary layer interactions reported in the literature [2,3,11], due to the concerns regarding the evolution of separated flow within the short run time; simulation and investigation of control of the interaction within the short run time is still challenging.

There are two ways by which the interaction may be controlled – one is by the manipulation of the boundary layer ahead of the interaction; the other is by adopting control mechanisms in the interaction zone itself (such as passive control, or boundary layer bleed at shock foot itself) [4]. Since the boundary layer's resistance to the imposed adverse pressure is crucial in determining the interaction, the increase of boundary layer momentum (or favourably modifying the boundary layer profile) upstream of the interaction would increase the boundary layer resistance, thereby tending to reduce the size of separation bubble or even prevent separation. Techniques of this type are thus appropriate for strong interactions (with separation, especially large separation bubbles). They include vortex generators which enhance the momentum of boundary layer flow by mixing with the outer flow [1,12,18,21]; boundary layer bleed through slots or holes which include suction and blowing upstream of the interaction, manipulating the boundary layer profile [6,7,17,19,20,23]; and also wall cooling (including film and transpiration cooling) [2,8,9].

Boundary layer bleed is a classical technique for separation control, but there are challenges in implementing the technique to shock boundary layer interactions, due to such concerns as bleed location relative to the interaction and bleed slot/hole geometry which can have effects on shock patterns. Especially, the bleed through the holes (rather than slot) is 3-dimensional and understanding the flow control requires the study of the complex 3-D flow field; this has largely been addressed by computational studies in the literature [7,17,20]. Bleed through micro-holes has also been used as micro-air-vortex generators upstream of the interaction [1,21]. Recently, boundary layer bleed has also been implemented to control SBLI phenomena of a shock train [23]. Many of the flow control studies reported in the literature are at supersonic speeds; while the above mentioned studies addressing the wall cooling are among the experimental investigations at hypersonic speeds (including the data from among the few reported shock tunnel experiments). There are few wind tunnel studies on the control of hypersonic SBLI using bleed. Schulte et al. [19] reported an extensive (hypersonic) wind tunnel study on the control of impinging shock wave boundary layer interaction using bleed (both suction and blowing) at Mach 6 (total temperature of 500 K). Optimal location of bleed relative to shock impingement location, bleed slot width and bleed slot angle, resulting in maximum reduction in separation length, were reported from among the experiments in the study. Haberle and Gulhan [6] also reported hypersonic (Mach 6) wind tunnel studies of scramjet inlet flow field in the presence of bleed for the control of lip shock induced separation on the ramp. Though these experiments were reported at high Mach numbers, shock tunnel experiments are required to understand the control of the interaction at higher total enthalpies associated with hypersonic flows. It is with this backdrop that shock tunnel experiments were initiated in order to understand the hypersonic impinging shock wave boundary layer interaction with *large* separation bubble and its control. The present paper reports the shock tunnel demonstration of reduction in separation length using bleed (both suction and blowing) at the location of separation occurring close to the leading edge, at same freestream Mach number and Reynolds number reported by Schulte et al. [19], but at a relatively higher total enthalpy of 1.3 MJ/kg.

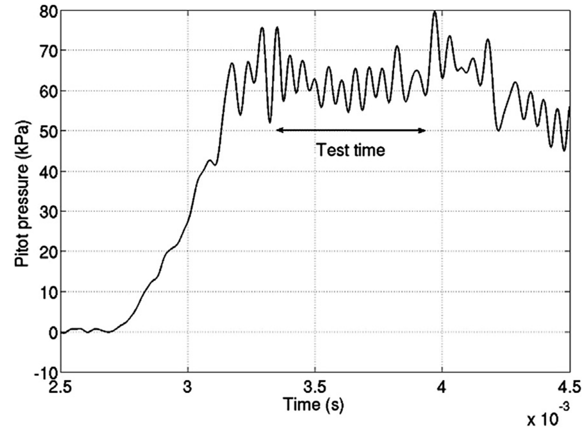


**Fig. 2.** A typical pitot signal in HST-2 (at Mach 5.96).

## 2. Experimental facility and test model

Experiments are performed in IISc hypersonic shock tunnel HST-2 [10]; the shock tube is 50 mm in diameter, with 2 m driver and 5.12 m driven sections, and the test section is of 300 mm × 300 mm cross section and 450 mm length. It is a conventional shock tunnel where the reflection of a propagating shock at the end of the shock tube compresses the test gas to high pressure and temperature (thus simulating the required enthalpy). The compressed test gas is expanded through a nozzle (whose area ratio simulates the required Mach number) into the test section of the tunnel. Thus the hypersonic flow of required flow enthalpy is simulated in impulsive fashion, i.e. for a short run time. The end of the shock tube is equipped with two fast response PCB pressure sensors from which the shock speed and the reservoir pressure after the shock reflection are measured. The total temperature after shock reflection is estimated from the shock speed. The total pressure after the normal shock inside the test section is measured using a pitot probe; the pitot pressure is measured by a fast response PCB sensor and all the pressure signals (including the surface pressure signals) are recorded by the data acquisition system NI PXI-6133 at a rate of 1 mega sample per second. A typical pitot pressure signal is shown in Fig. 2.

The pitot signal also serves to indicate the test time of the tunnel. It may be seen that the pitot signal rises from 2.7 ms to 3.3 ms (called rise time), after which it remains steady for ∼600 μs, till 3.9 ms. The time for which the pitot remains steady is the test time of the shock tunnel, during which the time averaged pressure gives the pressure behind a normal shock for the simulated freestream conditions (the steady pitot pressure). From the measured shock tube conditions and the pitot pressure freestream conditions are estimated using normal shock relations and are given in Table 1. The uncertainty in freestream Mach number $M_\infty$ is ±3.2%,