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Technical Notes

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Wind-Tunnel Setup for Investigations of Normal Shock Wave/Boundary-Layer Interaction Control

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Nomenclature

M	=	local Mach number
M_∞	=	freestream Mach number
p	=	surface pressure
p_{01}	=	total pressure upstream of control
p_{02}	=	total pressure downstream of control
x	=	streamwise coordinate
y	=	vertical coordinate
Δp_0	=	total pressure loss across control, $p_{01} - p_{02}$
Φ	=	overall pressure ratio (wind-tunnel pressure ratio), $p_{01} - p_{02}$

I. Introduction

THE control of transonic shock wave/boundary-layer interactions (SBLIs) has been studied widely because of its potential for transonic wings and supersonic engine inlets [1–5]. The expected benefits of control are wave drag reduction on wings and improved total pressure recovery in supersonic inlets. Successful control of a transonic normal SBLI smears the shock foot by changing the shock structure into a number of shock or compression waves without incurring additional viscous losses. For some types of control, the boundary-layer flow can also be improved, for example, through the introduction of streamwise vorticity.

However, to date, there is no single control device that can fulfill all objectives and fundamental research is necessary to understand the complex interactions that occur in the presence of shock control. Such fundamental studies are too expensive to perform on complete configurations or wings; for this reason, most researchers investigate shock control in transonic channel flow experiments where a normal (or nearly normal) shock wave interacts with the boundary layer formed along a wall of the working section, such as seen in Fig. 1. The results from such experiments are directly applicable to supersonic inlets and recent research in Cambridge [6] has demonstrated how predictions for wave drag reduction on transonic wings can also be obtained from such simple studies. Potential experimental setups can be divided into those where the shock wave is in a flow with strong Mach number gradients, such as in a diverging duct or on the rear of a transonic bump, and those where the shock is in a nominally uniform flow (parallel duct). The former allow for easier positioning of the shock wave but at the cost of introducing nonuniformities and a considerable variation of shock strength with streamwise location. The latter experiments are attractive because of their simplicity and, more importantly, because they allow a fundamental study of shock control with fewer dependent variables.

However, parallel duct experiments can experience difficulties in controlling the shock position relative to a control device [7]. Here we elucidate the cause of these difficulties and propose a novel wind-tunnel setup to overcome the problem.

II. Experimental Arrangement

A. Shock Instability Problem

A schematic diagram of a conventional setup to study normal SBLIs is shown in Fig. 1. Here the shock structure is positioned through manual control of the back pressure by using the streamwise gradient of total pressure due to the growth of boundary layers and geometry variations in flow direction. The constant area working section ensures uniform flow conditions and constant shock strength with streamwise location. However, Holden [7] reported that shock waves became increasingly unstable when various methods of shock control were tested. Similar unsteadiness of a shock structure in constant area working sections is reported in [8,9]. In fact, it can easily be seen that a control technique that is successful in reducing overall losses must introduce a shock instability when applied to such channel flow experiments. Although a similar shock instability is well known for supersonic converging ducts, it appears to have been overlooked in experimental studies of shock control.

To illustrate the problem, Fig. 1 also shows a schematic correlation between the streamwise shock position and the total pressure loss $\Delta p_0 \equiv p_{01} - p_{02}$ in the presence of beneficial control. Supposing a shock structure is located at point **a**, the shock moves downstream with a decrease of the back pressure hence an increase in the overall pressure ratio across the working section Φ. When it reaches point **b**, however, any further increase of Φ causes the shock to jump to point **c** and then continues towards point **d**. When moving a shock upstream through an increase in the back pressure (a decrease in Φ), it will jump from point **e** to point **f**. Therefore, due to this mechanism, it is impossible to hold a shock structure in a region where the total pressure gradient is negative. By definition, any successful control method is likely to introduce such regions, leading to shock unsteadiness.

B. New Working Section

To overcome the shock instability and allow the investigation of shock control in parallel channels, a new setup has been contrived to position shock waves in the presence of beneficial control devices. A somewhat similar concept for shock positioning was employed by Seddon [10] and Kooi [11]. The basic idea is to divide the incoming airflow into two parts, namely, an upper unchoked and a lower choked passage, as seen in Fig. 2. This causes a bow shock to form just ahead of the choked channel whose standoff distance is adjusted by the choking flap. Thus, this setup enables an accurate and stable positioning of a shock in the presence of control by moving the shock holding plate and adjusting the choking flap.

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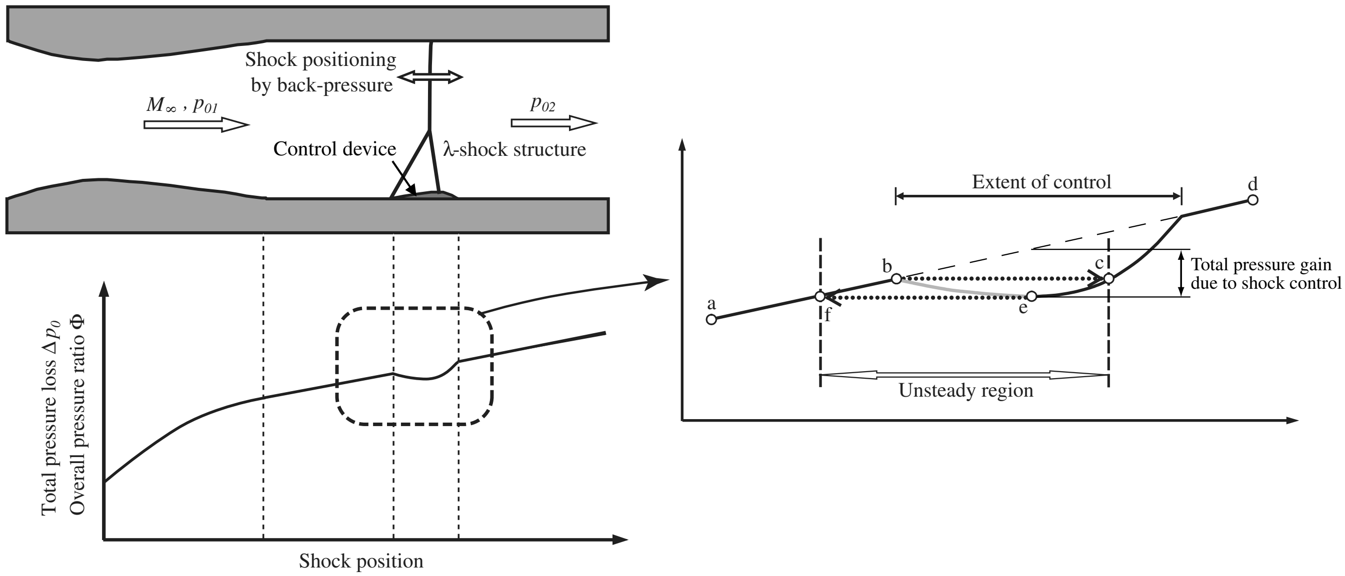

Fig. 1 Conventional working section (top left); correlation between shock position and total pressure loss (bottom left); correlation between shock position and total pressure loss near control region with beneficial control (right).

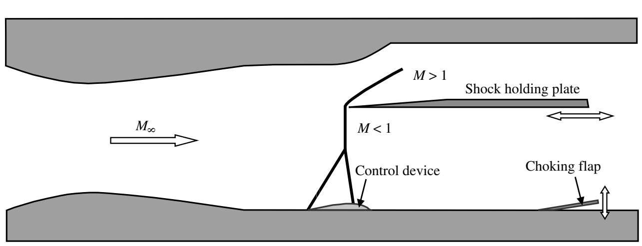

Fig. 2 Schematic diagram of new working section.

III. Results

Wind-tunnel experiments have been performed in the blowdown-type supersonic wind tunnel of the University of Cambridge. The rectangular working section is 114 mm wide and 178 mm high at the straight section downstream of the nozzle. The incoming airflow is partitioned by a plate of 6 mm thickness. The height of the upper and lower passages is 91 and 122 mm, respectively. The freestream total pressure is 1.7×10^5 Pa and the freestream total temperature is 297 K at a Mach number $M_\infty = 1.5$. The incoming boundary-layer thickness is 5.7 mm and the Reynolds number based on boundary-layer displacement thickness is approximately 25,000. The origin of the x coordinate is the nominal shock location, which was set to be 86 mm downstream from the nozzle end for the experiments described here.

A. Uncontrolled SBLI

Figure 3 composes schlieren photos of uncontrolled SBLIs with the conventional and the new working section. It can be seen that the essential flow features are identical in both cases. A small λ-shock structure can be observed at the foot of the normal shock wave, indicating shock-induced boundary-layer separation. Downstream of the triple point lies a shear layer, similar to that observed by Seddon[10]. Faint oblique lines represent Mach waves stemming from joints of parts or small disturbances on the tunnel walls, which have negligible effects on the flow. The apparent downward deflection of the floor is an optical effect of the schlieren system.

Figure 4 shows a comparison of the surface pressure distributions along the centerline for both configurations, which shows good agreement. Note that the pressure jump across the shock wave is better captured by the new working section (which is equipped with more closely spaced pressure tapping points); otherwise, the agreement is excellent. Plotted on the right of Fig. 4 are total pressure profiles measured behind the interaction. The small discrepancy stems from the difference of streamwise measurement location. The traverse was performed more downstream with the old setup, where the boundary layer is slightly thicker and has recovered somewhat.

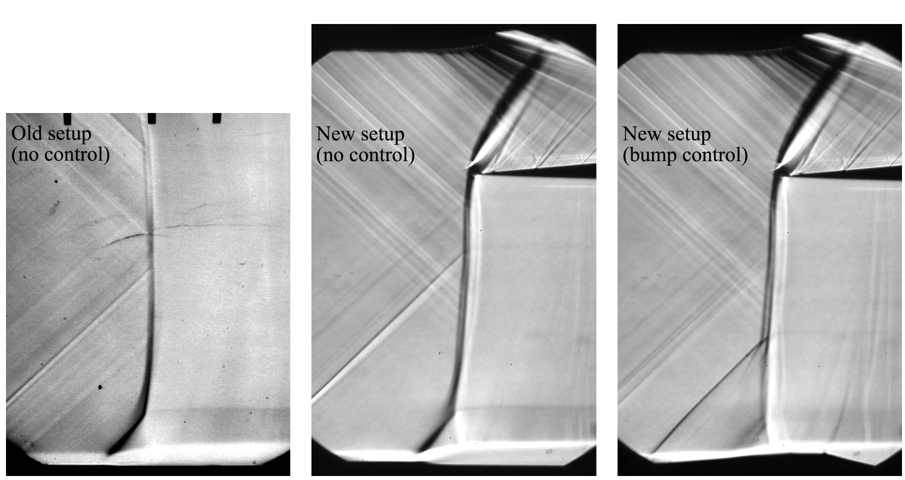

Fig. 3 Schlieren photos of uncontrolled and bump-controlled SBLIs.

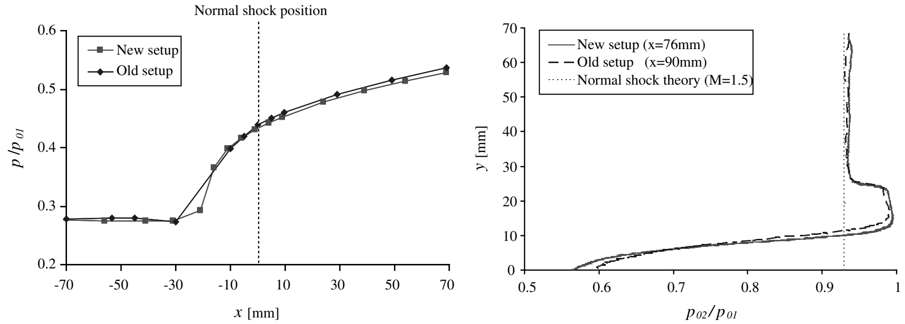

Fig. 4 Surface pressure distributions of uncontrolled SBLIs (left); total pressure distributions of uncontrolled SBLIs (right).

Taking all the results into consideration, it can be concluded that the new setup does not fundamentally alter the nature of the flowfield in the SBLI region.

B. Controlled SBLI

To investigate the effect of the new setup on shock stability in the presence of control, a 3-D bump section was mounted under the shock. This configuration had previously suffered from severe shock instability, making accurate measurements impossible [7].

As seen on the right of Fig. 3, it has been possible to locate the shock above the 3-D bump with the new setup. Now it is possible to observe a large λ-shock structure whose front shock leg starts at the onset of control. A somewhat smeared smaller λ-shock structure is present near the foot of the rear shock leg. Further flow details can be seen at the rear edge of the bump crest due to reexpansion. The remarkable improvement of the shock stability enabled measurements and visualizations that were impossible with conventional setups [12].

IV. Conclusions

A novel wind-tunnel setup has been proposed to test various types of shock control at positions where conventional setups are unable to hold a shock system due to the shock instability problem, which is particularly crucial in the presence of beneficial control. A new working section has been demonstrated to successfully overcome the instability problem without changing the nature of flow in the presence of 3-D bump control. The new setup is of particular use in the experimental investigation of beneficial shock controls capable of reducing overall drag.

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