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Measurement of Shock-Wave/Boundary-Layer Interaction in a Free-Piston Shock Tunnel

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The short test times available in free-piston shock tunnels necessitate careful examination of experimental techniques. Measurements of shock-wave/boundary-layer interaction are presented to demonstrate the effectiveness of these hypervelocity facilities in studies of a class of flows relevant to scramjet inlets. A two-dimensional laminar boundary layer with a freestream Mach number of 5.6, a static pressure of 4 kPa, and a unit Reynolds number of 1.14×10^6/m was used as the incoming flow condition on a cold flat plate. A shock wave was generated by a wedge normal to the plate and with a face at 2.84 deg to the freestream direction. It is found that, for this weak interaction, steady flow conditions can be obtained, although the test time is of the order of only 1 ms.

Introduction

THE present high level of interest in aerospace planes and orbital transfer vehicles has focused attention on a number of hypersonic fluid flow problems. One such problem is the unavoidable interaction between shock waves and boundary layers, both in the external flow over a hypersonic vehicle and in the inlet to an air-breathing propulsion system.

Shock-wave/boundary-layer interaction is extremely important in the inlet of the airframe integrated scramjet engine under consideration for the next generation of space launch vehicle. The shock waves that are necessary to compress the inlet flow must glance across the boundary layer developed on the forebody of the vehicle. Similar interactions will occur farther inside the scramjet. The sudden pressure rise through a shock wave causes thickening of the boundary layer in the region of the shock wave and can cause the flow to separate. In closed ducts such as scramjet inlets the extreme result is choking of the flow, with a shock wave standing outside the entrance to the inlet and flow spilling around the sides. Interactions can also increase the rate of shear locally in the flow, with hot spots developing on surfaces.

These phenomena in scramjet engines are examples of an important class of shock-wave/boundary-layer interactions in which sweepback of the shock wave is a key element. The flow in such interactions is then three dimensional. The configuration of interest in this paper is shown in Fig. 1. A planar shock wave generated by a wedge is swept back across a two-dimensional boundary layer on a flat plate. The leading edge of the wedge is normal to the plate. If the deflection angle of the wedge is small (less a few degrees), large regions of separated boundary layer generally do not occur and the interaction is regarded as weak. This three-dimensional interaction has been studied extensively[1-3] at freestream Mach numbers of about 3, but investigators disagree over some basic aspects of the flow. There is a dearth of experimental data at hypersonic Mach numbers, particularly at test conditions similar to those experienced in upper atmospheric powered flight and in re-entry.

A large hypervelocity test facility has been operating at the University of Queensland, Australia, since 1987. Its unique ability to reproduce realistic flight conditions is achieved at the expense of test run time. Typical steady flow test times are about 0.5–2 ms. The aim of the work reported here was to demonstrate that weak shock-wave/boundary-layer interaction studies can be performed in the facility. This involved showing that the test time is long enough for steady flow to develop and that the region of upstream influence can be determined with acceptable accuracy by surface pressure measurements.

Experimental Procedures

Shock Tunnel

Experiments were performed in the T4 free-piston shock tunnel in the Department of Mechanical Engineering at the University of Queensland, Australia. This is one of a series of free-piston shock tunnels designed by R. J. Stalker, the first three being constructed at the Australian National University, Canberra, Australia, in the 1960s. The latest in the series, T5, designed by WBM-Stalker Pty Ltd., is at the California Institute of Technology, Pasadena, California. The principle of operation has been described by Stalker.[4] The key feature is the heating of the driver gas (which drives the shock wave along the shock tube) by compression through a single stroke of a free piston in a large compression tube. In T4, the free-piston driver is 26 m long and 230 mm in diameter. The piston mass is 92 kg. The shock tube is 10 m long and 76 mm

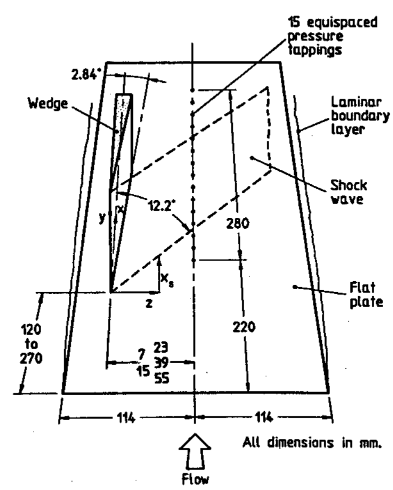

Fig. 1 Experimental configuration.

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Table 1 Shock tunnel test section conditions

Mach number	M_∞	$=5.6$
Flow velocity	u_∞	$=3850$ m/s
Stagnation enthalpy	H_o	$=9.6$ MJ/kg
Nozzle reservoir pressure	p_o	$=12.0$ MPa
Nozzle reservoir temperature	T_o	$=5590$ K
Freestream pressure	p_∞	$=4.0$ kPa
Freestream density	ρ_∞	$=0.012$ kg/m^3
Freestream temperature	T_∞	$=1210$ K
Wall temperature	T_w	$=300$ K
Ratio of specific heats	γ	$=1.34$

in diameter. A contoured nozzle was used, with a length of 861 mm, a throat diameter of 25 mm, and an area ratio of 110.

Model

A flat plate with a sharp leading edge (Fig. 1) was used to generate the boundary layer. The shock wave was generated by a wedge mounted normal to the plate. The angle of the face of the wedge to the freestream was 2.84 deg, this angle being kept small with the intention of ensuring that the interaction would be weak and without major flow separation. The leading edge of the wedge was unswept (i.e., normal to the plate).

A single row of 15 pressure tappings with 20-mm spacing (parallel to the x direction in Fig. 1) was machined into the plate. Finer spatial resolution of pressure distributions was obtained by moving the wedge in the x and z directions (while maintaining the wedge angle to the freestream). Checks were made to ensure that pressure tapping locations were clear of shock tunnel nozzle and model edge effects.

Test Conditions

The test section conditions were calculated from measurements in the shock tube of the filling pressure (the test gas was air), the primary shock velocity, and the time history of stagnation pressure immediately upstream of the nozzle. One-dimensional equilibrium shock tube calculations provided stagnation conditions upstream of the nozzle. From these, the test section conditions were obtained using one-dimensional, nonequilibrium nozzle flow calculations based on the code NENZF.[5] The test conditions are listed in Table 1.

The ability of NENZF to predict Mach number was confirmed, within the accuracy of the technique, by luminosity photography of the shock-wave angle on a 9.3-deg wedge in Mach 5.5 flow. The shock-wave angle to the freestream from the 2.84-deg test wedge was predicted to be 12.2 deg at Mach 5.6, assuming a perfect gas with $\gamma=1.34$.

These conditions gave a zero-pressure-gradient boundary layer with a unit Reynolds number of 1.14×10^6/m. For similar flow conditions, experiments[6] in T4 with heat transfer gauges indicate that transition occurs at a Reynolds number of between 1.1×10^6 and 1.5×10^6. This puts the expected transition location over twice as far downstream as the last transducer location. There is, of course, a distinct possibility that the disturbance created by the shock wave can initiate transition earlier than this. It is most likely, however, that the boundary layer is laminar, at least up to the shock location. The thickness of the undisturbed boundary layer was calculated using the code CHARNAL[7] written for two-dimensional parabolic flows. At 350 mm from the leading edge of the flat plate (a typical shock-wave location), the boundary-layer thickness was 3.5 mm. Here, the boundary-layer thickness is defined as the distance from the surface to the location where the local velocity is 99% of the freestream velocity.

Techniques and Instrumentation

Fifteen piezoelectric pressure transducers with built-in preamplifiers were used. Typical rise and decay time constants were 2 μs and 1 s, respectively. A small cavity above each transducer diaphragm was connected to the flow over the plate by a short 1.6-mm-diam hole. This configuration permits adequately fast transducer response, as demonstrated by the fast rise in plate static pressure during test flow starting (Fig. 2).

A digital event recorder was used to take 2048 samples of each pressure signal, with 2.5 μs between samples. The recording was triggered by the stagnation pressure signal in the shock tube upstream of the nozzle. After each firing of the shock tunnel, the data were transferred to a microcomputer for analysis.

Measurements were made with and without the wedge on the plate, the flat plate results being used to correct for flow nonuniformities as discussed in the following.

Results

Figure 2 contains typical time histories of the stagnation pressure p_o immediately upstream of the nozzle and a static pressure influenced by the shock wave (for the same shock tunnel run). The noise on the static pressure trace is due

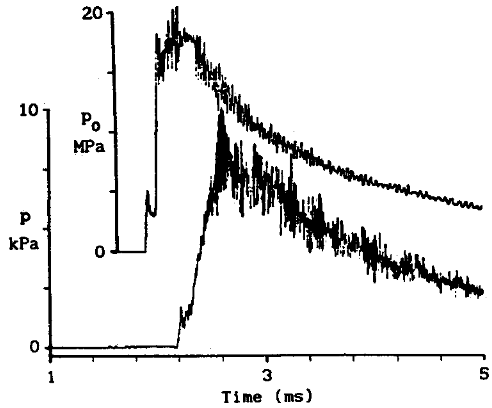


Fig. 2 Typical time histories of nozzle stagnation pressure p_o and flat plate static pressure p.

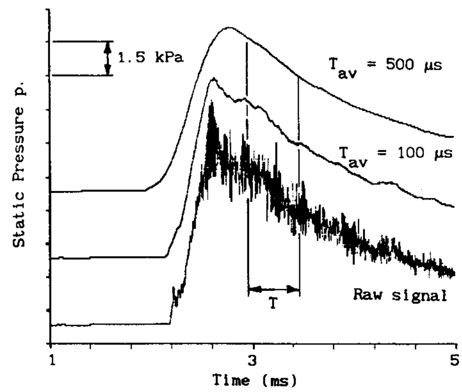


Fig. 3 Typical static pressure trace time averaged using two different averaging periods T_{av}.

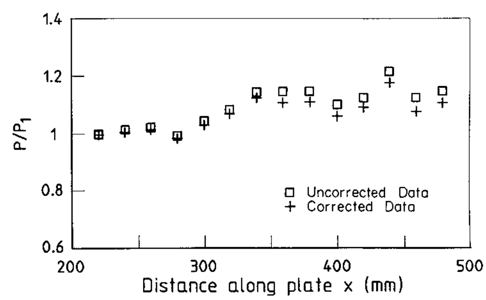


Fig. 4 Streamwise pressure distribution $p(x)$ on the plate without shock-wave generating wedge.

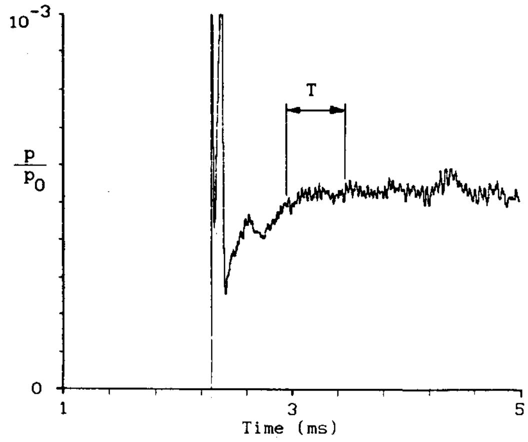


Fig. 5 Typical time history of static pressure p on the plate normalized by stagnation pressure p_o.

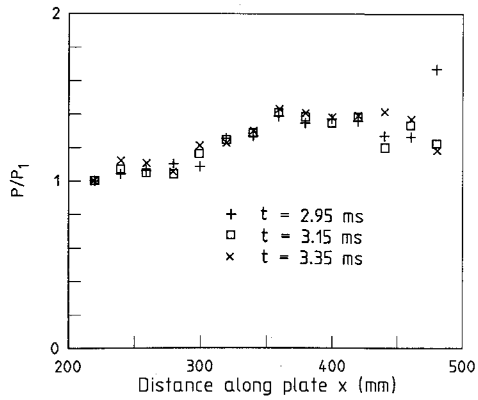


Fig. 6 Streamwise pressure distributions during the steady flow period T with the wedge present.

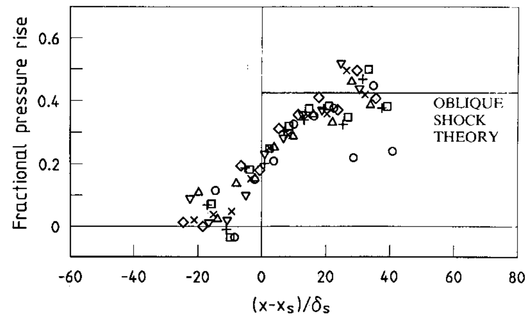
largely to pressure transducer response to stress waves in the model and has been reduced in Fig. 3 by digital filtering. The transient starting of the flow is apparent.

Although the pressures vary with time, there is a time interval when steady flow is established and data are reliable. It is marked as T on Fig. 3 and is 500 μs long.

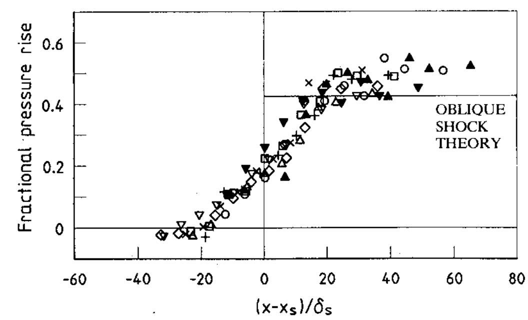
There are four justifications for calling the flow steady over the interval T:

1) The change in pressure of about 20% over this interval, and the associated changes in density, temperature, velocity, and viscosity, result in a change in Reynolds number of about 15% and negligible change in Mach number. Thus, the nature of the flow is not expected to change significantly.

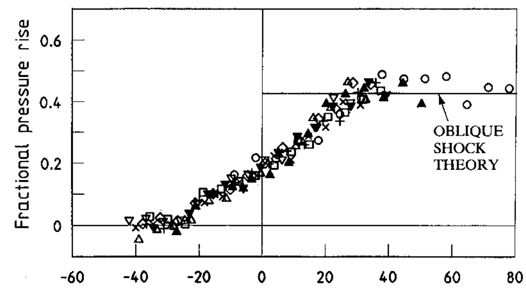
2) Over the time interval T, the test gas travels nearly 2 m (about four model lengths). A similar length of flow has passed the model before the start of T. This is considered sufficient to develop fully the boundary layer and the weak shock-wave/boundary-layer interaction. Strong interactions are characterized by large separated flow regions and might take longer to become steady due to the time required to feed the separation bubble to full size. Another effect of the high velocity of the flow (3850 m/s) is that the streamwise pressure gradient caused by falling nozzle stagnation pressure is low. The falling stagnation pressure results in a pressure rise of about 5% along the length of the model without a wedge. This is shown in Fig. 4 where a correction has been made to the raw data for the effect that transport delay and falling stagnation pressure have on the distribution. Pressures have been normalized by the value p_1 from the leading transducer. Given the hypersonic nature of the flow and the thickness of the boundary layer, it is considered adequate to compensate for this rise using the principle of superposition.



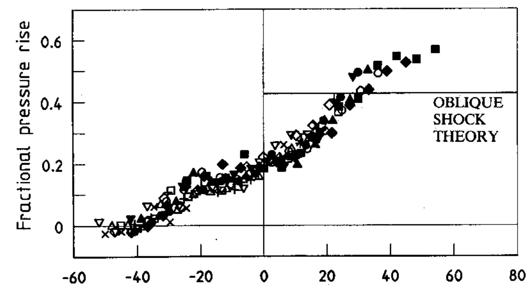
a) $Z = 7$ mm from wedge



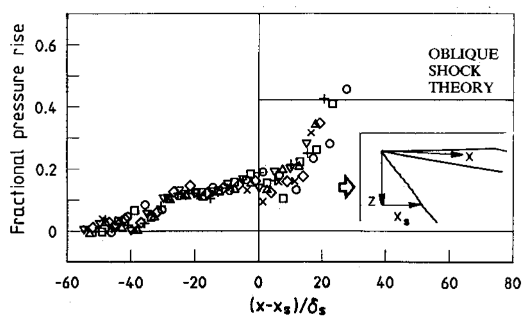
b) $Z = 15$ mm from wedge



c) $Z = 23$ mm from wedge



d) $Z = 39$ mm from wedge



e) $Z = 55$ mm from wedge

Fig. 7 Streamwise pressure perturbation distributions through the shock wave.

3) Although the test section static pressure falls with falling stagnation pressure, the ratio of the two pressures stays reasonably constant. This is an indication that a steady flow pattern has been established. Figure 5 is a typical time history of this ratio for a transducer that is affected by the shock wave (the data were time averaged over 100 μs before normalization). The initial spike is due to the shorter transport time of the starting flow and is not meaningful. The test gas is driven through the nozzle by expanding helium driver gas. Studies by Stalker and Morgan[8] indicate that contamination of the test flow by helium does not occur until after the time interval T. Helium contamination results in a lowering of the static to stagnation pressure ratio. No such effect is evident in Fig. 5.

4) The spatial pressure distribution pattern is nearly constant over the time interval T. In the data analysis process, a novel computer animation of the development of pressure distribution is used to identify when steady flow begins and ends. Figure 6 shows measured pressure distributions for one run at three different times t: the beginning, middle, and end of the interval T in Figs. 3 and 5. All three distributions are time averages over 100 μs and are normalized by the value p_1 from the leading transducer.

The steady flow pressure distributions were obtained by averaging the static pressures over all of the useful steady flow interval T. To allow for small variations between runs, the time-averaged static pressures from each run were first normalized by the instantaneous stagnation pressure for that run, with allowance for a calculated flow transport time of 400 μs. The pressure rise due to the shock was then taken as the difference between these averaged pressures and the corresponding ones from runs with no shock generating wedge. This difference, reduced to a fractional rise by dividing by the same "no wedge" data, is the ordinate of Fig. 7. Each symbol represents data from a separate test.

Distance along the plate in Fig. 7 is measured from the position of the shock wave and is normalized by the computed boundary-layer thickness δ_s at the shock-wave location. The location of the shock wave was determined from the oblique shock-wave relations for a perfect gas with a value of 1.34 for the ratio of its specific heats. Inherent in this is the assumption that the partially dissociated flow in the test section remains frozen through the weak shock wave so that a plane shock wave results. The 42% pressure rise measured across the shock (see Figs. 7b and 7c) is in reasonable agreement with that predicted by oblique shock theory with frozen chemistry. Error bars are not used in Fig. 7, as the scatter of the data gives an adequate indication of its reliability.

Upstream pressure influence due to the shock-wave/boundary-layer interaction is clearly demonstrated in Fig. 7. Close to the wedge, the extent of upstream influence is seen to increase with distance z from the wedge; whereas farther from the wedge, this rate of growth is small or even zero. This is consistent with the concept of an inception region close to the wedge in which the interaction develops with distance from the leading edge of the wedge. Figures 7c–7e give evidence of a region of almost constant pressure immediately upstream of the shock location. This region appears to grow in size with distance from the shock generator and may indicate the presence of a small separation bubble. Such incipient separation has been observed by others when studying weak interactions.[9]

Conclusions

It has been shown that hypersonic, laminar, weak, glancing shock-wave/boundary-layer interactions can be studied in free-piston shock tunnels, even though test times are of the order of only 1 ms. The quality of the data is adequate for evaluation of theories. The results also demonstrate that free-piston shock tunnels can be used to determine the wedge angle at which significant flow separation occurs.

The findings that shock-wave/boundary-layer interactions can be successfully investigated in a shock tunnel is particularly significant in that shock tunnels produce high enthalpy flow over relatively cold model surfaces, a condition expected to occur in engines of aerospace planes.

Acknowledgments

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